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APOLLO LUNAR LANDING MISSION SYMPOSIUM

JUNE 25-27, 1966

Manned Spacecraft Center
Houston, Texas

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APOLLO
LUNAR LANDING MISSION
SYMPOSIUM

PROCEEDINGS
AND
COMPILATION OF PAPERS

Approved:

Owen E. Maynard, Chief
Mission Operations Division
Apollo Spacecraft Program Office

INTRODUCTION

The Apollo Lunar Landing Mission Symposium was held at the NASA Manned Spacecraft Center on June 25, 26 and 27, 1966. The papers presented during the three days covered all aspects of the lunar landing mission, with primary emphasis on the lunar operations phases of the mission.

The purpose of the Symposium was to present the current lunar landing mission plan, and to subject the plan to a critical review by the body of experts who composed the audience.

To accomplish this objective, the papers were necessarily very detailed. Questions and comments were solicited from the audience after each paper, and this participation produced some excellent results. However, due to the volume and detail of material presented, the audience could not be expected to provide a critical appraisal from a single look at the material. For this reason, then, the papers and proceedings have been published and transmitted to each attendee. The attendees are urged to review the material and submit comments to Mr. Owen Maynard at the Manned Spacecraft Center.

The Symposium material has been published in three volumes, generally in the order that the papers were presented. The questions and comments from the audience follow the particular paper at which they were directed.

Volume I of the Symposium material contains, in addition to the formal papers presented on the first day, the introductory remarks by Dr. Gilruth, General Phillips, and Dr. Shea; Volume I also contains Dr. Shea's concluding remarks from the final day of the Symposium.

Due to time limitations during the Symposium, there were several topics of interest for which presentations had been prepared but were not formally discussed. These topics were the following:

a. Control of Lunar Surface Contamination and Back Contamination

b. Thermodynamic Constraints on Lunar Mission Capability

c. Service Module Reaction Control System Propellant Management.

This material has been included in the published version of the Symposium.

APOLLO LUNAR LANDING MISSION SYMPOSIUM

JUNE 25 - 27, 1966

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Saturday, June 25

WELCOME	Dr. Robert R. Gilruth
INTRODUCTORY REMARKS	Maj. Gen. Samuel C. Phillips
SYMPOSIUM INTRODUCTION	Dr. Joseph F. Shea
GENERAL MISSION SUMMARY AND CONFIGURATION DESCRIPTION	Owen E. Maynard
NAVIGATION, GUIDANCE AND CONTROL	Dr. Robert C. Duncan
LAUNCH THROUGH LUNAR ORBIT	M. P. Frank
LUNAR EXCURSION MODULE DESCENT	Donald C. Cheatham

Sunday, June 26

LUNAR SURFACE EXPERIMENTS	Robert E. Vale
LUNAR EXCURSION MODULE ASCENT AND RENDEZVOUS	Morris V. Jenkins
TRANSEARTH INJECTION THROUGH ENTRY	M. P. Frank
EARTH RETURN ABORT CAPABILITIES	Ronald L. Berry
SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES	Morris V. Jenkins

Monday, June 27

COMMUNICATIONS	Douglas R. Broome
CREW TASKS AND TRAINING	Joseph P. Loftus
EXTRAVEHICULAR MOBILITY UNIT OPERATIONS	William C. Kincaid
TOUCHDOWN DYNAMICS	Hugh Scott
SITE SELECTION CRITERIA	A. V. Bernard
SITE SELECTION DATA SOURCES AND INTERPRETATION	John E. Dornbach
CONCLUDING REMARKS	Dr. Joseph F. Shea

Additional papers included in the proceedings, but not presented at the Symposium:

CONTROL OF LUNAR SURFACE CONTAMINATION AND BACK CONTAMINATION	Richard H. Kohrs
THERMODYNAMIC CONSTRAINTS ON LUNAR MISSION CAPABILITY	Jerry W. Craig
SERVICE MODULE REACTION CONTROL SYSTEM PROPELLANT MANAGEMENT	Owen E. Maynard

INTRODUCTORY REMARKS

DR. ROBERT GILRUTH - OPENING REMARKS

Dr. Gilruth opened the symposium by welcoming the attendees to the Manned Spacecraft Center. He discussed the fact that this was not the usual kind of symposium in that it was designed more for the people who were conducting it, rather than for the audience. It could be considered as a working session in which NASA expected to get a hard-core review of the lunar mission.

He urged the participation and comments of the audience throughout the sessions, and then introduced General Phillips.

GENERAL SAMUEL PHILLIPS - OPENING REMARKS

General Phillips discussed the fact that the Apollo Program is committed to a set of technical requirements and configurations which are considered adequate. All the elements of the overall system either have been or are scheduled to be fully evaluated and qualified by tests before flight.

He stated that the purpose of the symposium is to focus attention on the lunar mission and to insure that the right things are being done in planning and preparing for the execution of the mission. Also, the symposium would serve to clearly identify any additional actions or any redirections of actions or efforts that this critical review of lunar mission planning might indicate.

General Phillips then turned the symposium over to Dr. Shea.

DR. JOSEPH SHEA - INTRODUCTION

Dr. Shea opened the discussions by explaining that the symposium covers the first lunar mission only and that the details of the earth orbital missions and the ground test program will not be discussed.

The current mission approach will be presented, and it is not claimed that this approach is necessarily correct in every sense. It may change between now and the time when it is actually accomplished, and it is not claimed to be unique. The question is more "Will this way work; is it adequate; then, is it the best?" Dr. Shea then went on to summarize several major considerations in the design of the first lunar landing mission. Detailed discussions of these points will, for the most part, make up the body of the symposium.

1. The first mission will be "open ended", that is, there will be discrete sets of decision points and the decision to continue, stop, or modify the mission will be made at these points along the way. The general concept for the mission is to keep it going as long as

possible; in effect, continue the philosophy in the test program of capitalizing on success. If everything continues to operate properly, the first lunar mission would be a landing mission. If there are problems of any significance along the way, depending on what the problem is and where it occurs, the mission will be changed or brought back.

2. The various limitations and constraints imposed on a lunar landing mission limit launch to roughly three days in any given month. Due to recycle time associated with the launch vehicle there must be an intervening day between each of these three days.

3. A major launch date constraint is that of lighting at the moon at the time of arrival. Lighting conditions are limited to the sun being in the region of 7° to 20° above the horizon and behind the astronaut as he approaches the site.

4. Due to the considerations for recycle time and the lighting constraints the mission will have to be prepared to go to any of three selected sites.

5. Data on the landing sites will come from three major sources: earth-based information, Orbiter A information, and Orbiter B information. Surveyor will be used to confirm this information and to tell in general what the lunar surface is like rather than to certify an individual site. We are not proposing to land, for the first mission, at an actual Surveyor site.

6. The spacecraft will be loaded with the maximum propellant and consumables that are consistent with the launch vehicle capability at the time of the mission.

7. Attitude variations will be used to control the spacecraft thermal extremes. This has resulted from design trade-offs which ease hardware design problems and save weight.

8. The Manned Space Flight Network will be used as the prime navigation source; however, onboard navigation capability is provided. The normal mission will be designed to conserve RCS consumables so that attitude maneuvers are to be minimized.

9. The LM descent engine will be used as a valid backup to the SM engine through lunar orbit insertion.

10. A free return trajectory will be used through lunar orbit insertion.

11. A limited number of landmark sightings, made by the onboard systems, will be used to reduce altitude uncertainties and effectively tie the Manned Space Flight Network to the moon.

12. There are three types of areas in which a landing could be made:

    a. A general area, possibly ten miles in length, in which any particular point could be an acceptable landing site.

    b. A specific area determined by the guidance dispersion. This could be an area about one and one-half miles in length.

    c. A specific point which would require considerable maneuvering to reach.

The first landing mission will probably use the specific area type of site. This would result in a saving of fuel (over the specific point site) which could be used for the hover and touchdown phase. The main point is that the capability exists to reach all three types of sites.

13. The crew will be used integrally throughout the mission, particularly in site selection and during the landing phase to avoid local obstacles and to provide visibility during touchdown. If increased knowledge indicates minimal dust problems from engine exhaust interaction and we can presume visibility through touchdown, then some of the mechanization of the landing and touchdown operation can be simplified.

14. The first mission will have an 18-hour surface stay and two joint excursions by the astronauts.

15. The concentric flight plan will be used for ascent from the lunar surface to rendezvous with the CSM. The LM Rendezvous Radar, the CSM optical system, and the MSFN tracking of both spacecraft will combine in the navigation and checking of the maneuvers.

16. The CSM has the inherent capability to rescue a LM from a low lunar orbit. Providing for this capability is one of the major contingency considerations for Service Module reaction control reactants.

17. The mission is planned for a water landing with the prime recovery zone in the Pacific Ocean.

18. There is essentially continuous abort capability throughout the mission. This includes LM descent up to and including touchdown. There is also the ability to stage the LM after impact if the impact dynamics could cause the LM to tip over.

19. There are several accumulators in the mission flight plan. These are places where the spacecraft could essentially mark time to get ahead or behind mission planning in the event of any unforeseen problems. These accumulators are as follows:

    a. Number of earth orbits.

    b. Number of lunar orbits before landing.

    c. Surface stay time.

    d. Number of orbits before rendezvous.

    e. Number of orbits after rendezvous.

20. The nominal and backup modes and redundancy provided for systems operation are significant, but will not be covered for all systems, such as environmental control and electrical power.

21. There is essentially a continuous communication capability except while behind the moon and occasionally during coast when the spacecraft is in a thermal roll condition. These intermittent losses of spacecraft to ground communications are not considered to be serious.

22. There are a reasonable set of precautions against contamination of both the lunar surface and the earth. It is recognized that as long as men are involved, there are biological products generated and there is a lower level of contamination at the moon which is essentially unavoidable.

23. There are some concerns of which the major ones are listed:

    a. Environmental effects can cause unexpected problems but these are not considered to be large.

    b. The calibration of the Guidance and Navigation system is a more significant point. Experience has shown that when a system is operated for the first time it can cause problems. The concern is that the first lunar mission is the first time that the G&N system will be used at lunar distances.

    c. The lunar landing is naturally of some concern. It has been simulated on earth, but lunar conditions cannot be completely duplicated.

    d. Crew tasks must be carefully watched to keep from overloading the astronauts.

CONCLUDING REMARKS

CONCLUDING REMARKS - DR. SHEA

Dr. Shea concluded the symposium by emphasizing that there are going to be many decisions to be made over the next 18 months, and that we will be in a position to make them as necessary. However, there is not much time for gross changes.

He brought out the fact that the software is at least as critical as the hardware, and that we will never have as much confidence in the qualification of the software as we have in the qualification of the hardware. This is because the software comes late, and because it contains so many multiple paths of operation that it is almost impossible to run enough simulations to check every possible combination.

He then summarized the status of the hardware:

- The Saturn I-B is already flying as a launch vehicle.
- The checkout cycle of what will be the first manned CSM is already far along and the flight will occur next year.
- Almost all of the stages of the Saturn V have been delivered. It will fly next year.
- The LM and the first Block II CSM should fly next year.
- The capability to do the lunar mission should be available to us very soon.

Dr. Shea closed his remarks by suggesting that all attendees have an obligation to review the results of the symposium and send comments to Mr. Owen Maynard at the Manned Spacecraft Center.

GENERAL MISSION SUMMARY
AND
CONFIGURATION DESCRIPTION

by

Owen E. Maynard

and

John R. Sevier

PLATEAUS AND GROSS MISSION DESCRIPTION

This section of the Symposium will describe the first Apollo lunar landing mission in its entirety--from lift off to recovery. It is intended to provide the general framework for a proper understanding of the subsequent presentations which will explore particular aspects of the mission in considerably greater detail.

It is useful to think of the lunar landing mission as being planned in a series of steps (or decision points) separated by mission "plateaus" (Figure 1). The decision to continue to the next plateau is made only after an assessment of the spacecraft's present status and its ability to function properly on the next plateau. If, after such as assessment, it is determined that the spacecraft will not be able to function properly, then the decision may be made to proceed with an alternate mission. Alternate missions, therefore, will be planned essentially for each plateau. Similarly, on certain of the plateaus, including lunar stay, the decision may be made to delay proceeding in the mission for a period of time. In this respect, the mission is open-ended and considerable flexibility exists. This flexibility will be discussed in detail throughout the symposium.

It will be convenient, for purposes of overall mission description, to quickly go through the mission plateaus and decision points. Following this gross description, the operations for each plateau will be examined in greater detail.

The end points of these plateaus representing major "commit" points in the lunar landing mission are characterized by propulsive maneuvers resulting in major changes in the spacecraft energy. These commit points and mission plateaus can both be represented schematically on a single chart as shown in Figure 2.

Figure 2 illustrates the major maneuvers during the lunar landing mission in terms of both delta V (on the left) and pounds of propellant (on the right). These maneuvers represent the "commit" points, and the space in between represents the plateaus. A pictorial representation of the mission is illustrated in Figure 3, in which the Earth, the Moon, and their relative movement throughout the mission are shown to scale in an earth centered coordinate system. The spacecraft's orbits about the earth and moon are, of course, not to scale.

The first plateau, pre-launch, terminates at launch from the Complex 39 facility at Merritt Island. The launch to earth orbit is performed with the first two stages and a partial burn of the third stage of the Saturn V launch vehicle. As shown in Figure 2, the earth ascent phase represents the major expenditure of propellant for the lunar landing mission, approximately $5\frac{1}{2}$ million pounds of propellant has been expended to place the approximately 300,000 pound payload in earth orbit.

Referring to Figure 3, the earth ascent phase is shown schematically as it might be seen from some distant point in space looking down on the earth-moon plane. The moon's position at launch is shown in the lower right hand corner, and its daily movement, as as the mission continues, is shown at successive points.

Following the ascent, the spacecraft reaches the second mission plateau, earth parking orbit. During each parking orbit, which can last up to $4\frac{1}{2}$ hours, spacecraft systems are checked out and made ready for the next major maneuver, translunar injection. As shown in Figure 2, translunar injection represents a considerable change in spacecraft energy; the velocity is increased by some 10,000 ft/sec with a propellant expenditure of about 150,000 lbs. from the second burn of the launch vehicle's third stage.

Following translunar injection, the spacecraft is on the next plateau, translunar coast. An initial period of ground tracking is performed to confirm that the spacecraft is on a satisfactory trajectory, and following this confirmation, the transposition and docking operation is performed. This operation involves the Command and Service Modules (CSM) separating from the rest of the configuration, turning around and docking on the Lunar Module (LM), which is still attached to the S-IVB, and continuing the translunar coast. Sufficient separation velocity is applied by the Service Module Reaction Control System (RCS) to assure that there is no possibility of subsequent recontact with the S-IVB.

The spacecraft continues to coast on the translunar leg of the trip for approximately the next three days. Two or three midcourse corrections will be made by the Service Propulsion System (SPS) during the translunar coast phase to assure that the spacecraft arrives at the correct location for its next major burn, lunar orbit insertion.

The lunar orbit insertion maneuver occurs behind the moon after the spacecraft has passed out of line of sight to earth. The maneuver is performed with the SPS and requires approximately

3500 ft/sec dalta V and 25,000 lbs. of propellant. If the lunar orbit insertion maneuver is not performed for some reason, then the spacecraft merely circumnavigates the moon and returns to safe earth entry conditions on the free return trajectory with no SPS engine burns required.

The successful execution of the lunar orbit insertion maneuver, however, will have placed the spacecraft in a circular orbit about the moon at an 80 anutical mile altitude. This is the next plateau. After at least three revolutions in lunar orbit, the LM is separated from the CSM and the two man crew begins their descent to the lunar surface, leaving a single crewman behind in the Command Module (CM).

The initial deboost of the LM from its 80 nautical mile orbit is made behind the moon and is performed by a small impulse from the LM descent engine. Following a coast period of about one hour, during which the LM has slowly descended to 50,000 ft. altitude, the descent engine is again ignited and the main braking maneuver is initiated. From this point, the descent to the surface requires about 10 minutes, the latter portion of which is under manual control of the crew. As noted in Figure 2, the fuel expenditure for this maneuver has been over 15,000 lbs., with an equivalent delta V of about 6500 ft/sec.

Following the lunar landing, the crew will secure the LM, don their extravehicular life support equipment, and exit to the lunar surface. During the 18 hour lunar stay period, there will be two exploration periods of three hours each performed by both crewmen. The extravehicular activity will consist of sample collection, emplacement of the experiments package for long term operation, photography, and general geological observation. Following the return from the last exploration period, the crew performs the pre-launch checkout of the LM systems and prepares for launch. At the proper time, with the CSM approximately 10 degrees ahead of LM, the ascent engine is ignited, and the LM ascent stage lifts off from the moon, leaving the descent stage on the surface. Although the ascent trajectory involves several maneuvers from lift-off until rendezvous with the CSM is accomplished, the most significant of these is the main powered ascent which involves a continuous engine burn from the surface to burnout at 50,000 ft. altitude. From Figure 2, it may be seen that about 5000 lbs. of propellant is expended with an equivalent delta V of about 6000 ft/sec. The conclusion of the main ascent burn at 50,000 ft. is such that the LM is placed on a safe coasting trajectory which will not impact the moon, even if the subsequent rendezvous maneuvers were not made for some reason. This leaves the LM in a relatively stable situation from which rescue by the CSM could be made, if necessary. However, the planned ascent phase continues with a series of small impulses provided by the LM RCS, and rendezvous occurs with the CSM about two hours after lift-off.

After the LM has docked on the CSM, the LM crew is transferred to the CSM, along with the data and samples collected on the lunar surface. The LM is then jettisoned, and preparations are made for the next major maneuver, transearth injection is slightly less than for lunar orbit insertion, and the propellant expenditure for the SPS is considerably less, about 8000 lbs. since the spacecraft is considerably lighter.

It is of interest to nate that up until the LM began its descent operations, a backup for transearth injection was available in the LM descent engine. This is considered a particularly useful capability, since it allows one to guard against an SPS failure during its first major burn for lunar orbit insertion. For this reason, then, the propellant requirement for transearth injection is also shown in terms of what would be required from the LM descent propulsion system, approximately 14,000 lbs.

Following the transearth injection, the spacecraft is on a plateau much like the outgoing leg of the trajectory to the moon. It is targeted to arrive at safe entry conditions at the proper time to allow it to reach its primary recovery area in the Pacific Ocean. Small midcourse corrections during the transearth coast assure that these conditions are reached. Shortly before arrival at the entry point, the Service Module is jettisoned, and the CM is oriented for entry. Entry range varies between 1500 and 2500 nautical miles and is controlled by rolling the CM during the entry phase. At 25,000 ft. altitude, the drogue parachutes are deployed and followed a short time later by the main parachutes which slow the CM to safe touchdown conditions.

Recovery is soon effected, and with the CM and crew safe aboard ship, the mission is completed.

Having completed this gross description of the total mission, it will be of interest to devote some attention to a few basic mission planning considerations before proceeding with the more detailed description of the mission.

PLANNING FOR LAUNCH ATTEMPTS, LUNAR LIGHTING AND SITE SELECTION

This section will consider launch attempts at the earth, the lighting conditions for landing at the moon, and any intervening accumulators that could contribute to a balanced planning scheme for a first manned landing attempt. Consider first the question of launch attempts at the earth: very briefly, it is highly advantageous to allow at least 48 hours between each of 2 or 3 scrubs. Certainly more than one $2\frac{1}{2}$ hour launch window per month is required because of the reasonably high probability of scrub against a particular scheduled time, and the resulting impact on the following program.

The causes of scrubs and holds could be due to launch vehicle, spacecraft, launch complex or MSFN systems problems, or possibly weather.

Probability of scrub has historically, and for good reason, increased markedly after the time it becomes necessary to recycle in the event of a scrub.

Recycle time is controlled by "fix" time, holding limits, servicing cycle, weather, launch and control team recycle and flight crew change time. It is interesting to note here that although there are multiple shifts involved there are no complete backup teams except in the case of the flight crew. Historically, and again for good reason, recycle times are most frequently in excess of 24 hours and usually more like 48 hours.

The probability of a scrub against a particular scheduled launch time is reasonably high: approximately 1/3. The probability of launch, therefore, increases markedly as multiple recycles are allowed: starting at about .67 for no recycle, .89 for one recycle, .96 for two recycles, and .99 for three recycles. Since the recycle time is usually in the neighborhood of 48 hours, then planning launch opportunities for consecutive days will not signficantly increase the probability of launch over that which considers only alternate days as opportunities. From these considerations, then, it is highly advantageous to provide some accumulator time in the system to permit at least two and possibly three recycles of at least 48 hours each (Figure 4).

Before considering the implications of this, another basic constraint will be examined; namely, the lighting requirements at the moon. Present understanding of the nature of the photometric function at the moon, and more direct observation as well, leads to the conclusion that there probably exists a small range of sun elevation angles where a flight crew's ability to select an acceptable touchdown point is markedly improved over angles outside that range. Again, present understanding of the nature of the lunar surface leads to the conclusion that it is necessary to provide a high capability in this area in order to enhance mission success and crew safety. The first lunar landing mission is now being planned for sun elevations between 7° and 20° to take maximum advantage of the crew's visibility in the landing operation (Figure 5).

Since there is only a range of 13° in the planned sun elevation angle, and since sun elevation angle changes at the rate of 26° for a single 48 hour recycle, 52° for two recycles, and 78° for three recycles, then this obviously leads one to look at multiple landing sites.

Some slight flexibility in launch to a single site could be realized if accumulators were used as built-in holds. That is, plan to launch early after a successful countdown, and wait at some point in the mission for the planned landing site to catch up to the correct lighting condition. For example, additional earth orbits and lunar orbits could be used as built-in holds at the rate of 1/2° per hour (3/4° per earth orbit and 1° per lunar orbit). Similarly, translunar transit time could be used as an accumulator at this same rate. Since there is a limitation of only three earth orbits (from S-IVB consumables considerations), and since the free return trajectory requirement restricts translunar transit time to a narrow range, then about the only significant flexibility item is in the number of lunar orbits prior to LM descent. However, to make full use of this, one would sometimes launch so early that more than 70 hours in lunar orbit would be required to rectify the lighting conditions at the landing site. Consumables and systems limitations would then become a problem.

Therefore, the multiple landing philosophy becomes an inherent feature in lunar mission planning.

Apollo mission planning personnel have, of course, been involved in Lunar Orbiter and Surveyor site selection. The sites shown on Figure 6 are from the Orbiter A and Orbiter B missions as planned to be flown later this year. These particular sites have been identified at this point in time as most probable to contain acceptable touchdown points in a large dispersion ellipse and radar approach terrain.

Considering now the question of spacecraft performance, Figure 7 shows the dates in 1968 when certain of these sites can be achieved with the 7° to 20° sun elevation. In generating the performance scan, a 95,000-pound spacecraft was assumed loaded with 37,500 pounds of SPS propellant of which 1% was retained for reserve.

It is apparent that changing the magnitude of sun elevation would shift the days that a particular site could be reached. Increasing the range of sun elevation would increase opportunities and decreasing the range of sun elevation would obviously decrease opportunities.

Figure 7 (for Pacific injections) and Figure 8 (for Atlantic injections) show that availability of a considerable number of potential sites generally clustered in three groups which are separated in longitude such that a 48-hour recycle back at earth launch could be accommodated as planning is shifted westerly from one group to the next.

It is apparent from Figures 7 and 8 that Pacific injections (with daylight launches) are available during most of the year. During the latter part of the year, Atlantic injections would probably be used, and night launches would be necessary.

The distributions for 1967 and 1969 are not markedly different from 1968. Figures 9 and 10 illustrate the landing site accessibility for 1969.

Figures 11 and 12 illustrate this mission planning concept for typical launch dates in 1968 and 1969 to three of the candidate sites. These figures show how the earth, moon, and sun cooperate to give a reasonable mission concept allowing two 48-hour recycle times at the launch pad and near optimum lighting at the moon. The precise targeting points would not have to be determined until about 6 months before the mission.

Data for site selection starts with earth-based photography, radar, IR, and other observations, which, at this point in time, have lead to the selection of a relatively large number of contender sites for Apollo landing. This number will decrease as Orbiter and Surveyor data lead to conclusions as to which are the better sites in the three groups. By 6 months before the mission it would be highly desirable to have narrowed these sites down to one in the east, one in the central portion, and one in the west.

The scientific objectives of at least the early mission have been developed so that they may be achieved virtually independent of the site location.

Having established these basic mission considerations of launch opportunities, lunar lighting, and site selection, we will now return to the mission itself and examine it in more detail.

DETAILED MISSION DESCRIPTION

Figure 13 illustrates the orientation of the space vehicle on the launch pad. The spacecraft's -Z axis (direction of crew's heads), and the launch vehicle's Position I are pointing east. In the event of a pad abort at this point, the orientation is such that the trajectory of the launch escape vehicle would take the Command Module over the water.

The inner gimbal of both the spacecraft's and launch vehicle's Inertial Measurement Unit (IMU) are aligned normal to the upcoming orbit plane.

EARTH ASCENT

Lift off occurs after a 6-second hold down period following engine ignition (Figures 14 and 15). During this time, the thrust of the first stage, which is powered by five F-1 engines, has built up to its rated value of $7\frac{1}{2}$ million pounds. The thrust-to-weight ratio of the vehicle at this point is 1.25, so its initial ascent acceleration is relatively small.

The space vehicle rises vertically from the launch pad until the mobile launcher is cleared. It then performs a roll maneuver to align the launch vehicle Position I along the desired launch azimuth, which can vary between 72° and 108°. The orientation on the pad was such that Position I was pointing east.

Following the vertical rise period, which lasts 12 seconds, a programmed pitch is commanded which will continue throughout the first stage burn. Maximum dynamic pressure (700 psf) is reached at about 84 seconds at an altitude of 43,000 feet. The inboard engine cutoff of the first stage will occur about 155 seconds after liftoff and will be followed by the outboard engines' cutoff four seconds later. During this first stage operation, the spacecraft will have attained an altitude of about 200,000 feet and will be about 65 nautical miles down range. Maximum acceleration will have occurred at this point and amounts to about $4\frac{1}{2}$ g's. Tracking and communications will have been continuous during this period with the ground-based facilities in the Cape area and with the facilities at Grand Bahama during the latter portion of the burn.

Since the launch vehicle operations during this period are automatic, the crew has been functioning in mainly a capacity of monitoring launch events and communicating the occurence of these events to the ground. Critical spacecraft systems are being checked and abort readiness maintained. An Emergency Detection System is provided onboard for sensing various malfunctions of the launch vehicle and displaying this information to the crew, who can then initiate an abort, if necessary. Automatic abort capability is maintained to nearly the end of first stage operations to allow for extremely time critical situations.

Following cutoff of the outboard engines of the first stage, thrust decay to 10% occurs in about one-half second, at which time the second stage ullage rockets and first stage retrorockets are fired, and S-IC/S-II separation occurs (Figure 16).

The S-II stage is powered by five J-2 engines, each having 200,000 pounds of thrust. Thrust buildup to the rated value occurs rapidly and at 163 seconds after liftoff, the second stage has reached full thrust.

At this point, the switchover is made from the programmed-pitch guidance scheme, used during first stage operations, to a path-adaptive scheme used during second and third stage operations. All guidance equipment for the launch vehicle is contained in the Instrument Unit located between the S-IVB stage and the spacecraft adapter. As stated previously, all guidance operations during ascent are performed automatically by the launch vehicle. However, from this point on in the ascent, the capability exists onboard the spacecraft to take over the guidance function in the event of a failure of the launch vehicle inertial platform.

Approximately 25 seconds after S-II ignition, the S-IC/S-II forward interstage is jettisoned, and this is followed by the Launch Escape System jettison five seconds later (Figure 17). Up until this point, the Launch Escape System has been the means of safely removing the CM and crew away from a malfunctioning vehicle in the event an abort was necessary. The high thrust and acceleration capability of the Launch Escape System motors was required to accomplish a safe abort during the atmospheric portion of the flight. At this point in the mission, however, the Service Propulsion System has the capability to abort the spacecraft off the launch vehicle, so the LES is jettisoned.

The Boost Protective Cover is attached to the LES and is jettisoned at the same time. The BPC is a semi-soft fiberglass construction, and its function has been to absorb the CM aerodynamic heating during boost and to provide a protective shield to maintain a satisfactory thermal control surface during the rest of the mission back to entry. The actual jettison operation of the LES, however, produces another thermal problem. The exhaust products of the solid jettison motors will impinge on the clean surfaces of the CM and SM and degrade the thermal performance to some extent. The complete nature of this problem is currently being investigated in ground tests and will be investigated in the early development flights. A later paper in the Symposium will discuss this in detail.

The second stage burn continues for a total duration of about 375 seconds, a little more than 6 minutes. At this point, orbital altitude of 100 nautical miles has essentially been reached but the spacecraft is still short of orbital velocity by about 3000 ft/sec. (Inertial velocity is about 22,650 ft/sec.) The vehicle is almost 900 nautical miles down range from Cape Kennedy. The tracking stations at the Cape are out of range by this time but other stations at Grand Turk or Bermuda have picked up the space vehicle, so that communications have been uninterrupted.

Following shutdown of the five J-2 engines on the S-II stage, S-II/S-IVB separation occurs, and third stage operations being (Figure 18). The separation sequence is similar to the one described for S-IC/S-II stage separation -- the S-II retro-rockets and S-IVB ullage rockets fire and pyrotechnic devices separate the stage. The problem mentioned previously of thermal cotaing degradation from the LES jettison motors is similar to one which occurs during S-II/S-IVB separation in that the S-II retro-rocket gases impinge on the Service Module thereby degrading its thermal coating. As mentioned previously, this problem is currently under investigation.

The S-IVB burn during the boost phase lasts for about 2½ minutes and imparts some 3000 ft/sec to the spacecraft velocity--boosting it up to the orbital velocity of about 25,600 ft/sec. Thrust of the S-IVB stage is about 200,000 pounds--produced by the single J-2 engine. At the conclusion of the S-IVB burn, the spacecraft is at 100 nautical miles altitude and has traveled another 600 nautical miles downrange for a total distance during the boost phase of almost 1500 nautical miles. The total ascent has taken about 11½ minutes (Figure 19).

During this third stage burn, communication with the land-based stations has been lost, but the spacecraft has been acquired by the insertion ship which has been specifically located to fill in this gap and to provide a period of tracking immediately after insertion for the purposes of confirming a safe orbit, and issuing the decision to continue the mission. The insertion ship coverage is shown in Figure 20.

EARTH PARKING ORBIT

Following this confirmation of a good orbit, the spacecraft is on the second mission plateau which was discussed earlier (Figure 21). The capability exists to spend up to three complete revolutions in the earth parking orbit, but normally the plan will be to execute the next commit point, translunar injection. during the second revolution. The limitation of three revolutions is associated with considerations of S-IV consumables limitations. A later paper in the Symposium by M. P. Frank will discuss the geometry of translunar injection opportunities will occur every day over both the Atlantic area and the Pacific area, but one of these will be preferred from a performance standpoint. In addition, it is not planned to have support aircraft covering both areas simultaneously. Therefore, having planned the mission for a specific period during a specific month, and have launched at a specific time of day, then the mission is committed to either a Pacific or Atlantic injection. For reference purposes, the Pacific injection will be considered as the appropriate windown, and for the reference mission, it will occur over the Western Pacific Ocean, near the equator. The only parameter left to choose is which of the three opportunities during the three revolutions translunar injection will occur. The answer to this question is dependent upon how extensive a set of operations is planned during earth parking orbit. These operations will be discussed below. Figure 22 shows the ground track and station coverage for the typical reference missions; it will be useful to refer to this figure during the discussion of earth orbit operations.

Immediately after S-IVB cutoff at earth orbit insertion, the launch vehicle propellant tanks are vented of hydrogen and oxygen gases to relieve the pressure buildup. Venting is performed at this time in order to prevent unpredictable vents from interferring with sensitive attitude operations later on. The venting sequence is preceded by ullage rocket firing to assure propellant settling sc that only vapors are vented. After settling the propellant, the oxygen tanks are vented for about 15 seconds. Hydrogen venting will be done continuously throughout earth orbit, along the spacecraft's thrust axis; however, the propulsive force of the hydrogen venting is extremely low and, therefore, would not interfere with other operations during earth orbit.

Following earth orbit insertion the crew remains in their couches until the Manned Spaceflight Network (MSFN) has verified that the spacecraft is in a safe orbit. This confirmation is provided after about three minutes of tracking by the insertion ship in the Atlantic. At this time a sate vector update is provided by the ground which the crew inserts into the onboard computer.

Following a brief onboard checkout of spacecraft systems, the navigator will leave the center couch and go the lower equipment bay of the Command Module and prepare for the first operation--

the alignment of the CM inertial Platform. This is done both as checkout of the IMU and its associated equipment and to establish a precise onboard inertial reference as a backup to the inertial reference in the S-IVB Instrument Unit.

The spacecraft attitude at this point is such that the longitudinal axis is in the local horizontal and pointed in the direction of flight. This local horizontal mode is maintained automatically throughout the earth orbit phase by a constant pitch rate equal to the orbital rate of .067 deg/sec. The S-IVB control system provides this mode and maintains it within a one-degree deadband. This orientation in the local horizontal mode assures communication coverage of the spacecraft and launch vehicle antennas when passing over a ground station.

The crew takes over manual control of the vehicle attitude through the S-IVB control system during spacecraft operations requiring specific attitudes. This is the case during the IMU alignment mentioned earlier. Depending on the time of day that launch occurred, the sun location may interfere with the optics line of sight during the IMU alignment. A roll maneuver would then need to be performed by the CM crewman before beginning the alignment. Following the alignment, the crew would return the vehicle to the original roll orientation with the -Z axis of the spacecraft pointing down the local vertical.

Because of the relatively high inertias of the vehicle, these maneuvers must be performed at relatively slow rates to conserve S-IVB RCS propellant. Present rates are set at .3°/sec. in pitch and yaw, and .6°/sec. in roll. For the IMU alignment operation, then to roll 60°, say, to avoid the sun will require 2 minutes; allowing 10 minutes for alignment, and another 2 minutes to roll back for a total of 14 minutes. This figure, together with the time required to get set up for the operation, means that some 40 minutes have elapsed since liftoff before this operation is completed. The spacecraft's position at this point would be over the Indian Ocean.

Meanwhile, the other systems checkout are being conducted by the other two crew members. Data transmission and voice communication are being maintained over every ground station. In between the stations, data is being recorded onboard for playback when over a station. Tracking periods by ground-based S-band stations (subsequent to the initial insertion ship tracking) will have been provided by the Canary Island station for the case of northerly launch azimuths and by Ascension for the southerly azimuths. For the range of around 90° azimuth, tracking by a ground-based station will not be available until the pass over Australia; however, a ship in the Indian Ocean will be stationed to provide coverage before this time. These tracking periods by MSFN will provide a precise determination of the spacecraft's orbit and the translunar injection parameters to be inserted into the Command Module computer, which will back up the launch vehicle guidance system for translunar injection.

The actual commit point for translunar injection must occur at least 7 minutes before S-IVB ignition, since this is the period required for the S-IVB restart sequence. This sequence initiation may be inhibited by the crew if it is decided to delay injection until the next orbit, but once the sequence is started on a given orbit it is not possible to delay injection to the next orbit. The sequence can, of course, be terminated at any time, but the limitations on S-IVB consumables (associated with the chill-down and with the ullage propellants) do not allow a second opportunity.

These considerations of tracking, spacecraft checkout, IMU alignment, and data transmission and analysis would, therefore, make it unlikely that translunar injection would be able to occur on the first Pacific opportunity when this injection point lies over the Western Pacific. It may be practical, however, to make the first Pacific opportunity when the injection points are in the eastern part of the Pacific. This is being looked into at the present time and there appears to be no strong reasons why it could not be accomplished. The advantage to planning the mission to inject as early as possible would be to allow maximum time (within the 4½-hour limitation for the S-IVB) to correct a temporary malfunction of either the onboard systems or the gound systems.

For purposes of the reference mission description, however, it will be assumed that injection will occur on the second Pacific opportunity, so the spacecraft continues on in earth orbit passing over the Pacific Ocean ship, over the Hawaii station, and finally coming up on the West Coast of the United States. Across the United States, the tracking is continuous by stations at Goldstone, Guaymas, Corpus Christi, and Cape Kennedy.

Since injection is not taking place during the first orbit, then time will be available to perform a series of landmark sighting in earth orbit to test the ability to perform the same type of navigation to be used in lunar orbit. These are not necessary to earth orbit determination since this has all been done by the ground; however, if time, lighting, and cloud cover permit, then a few sightings may be taken as a further checkout of our onboard system. To perform the sightings, it will be necessary to roll the spacecraft 180 degrees from the standard attitude, so that the optics axis is pointing toward the earth. Following the sightings, the spacecraft is rolled back to its normal attitude.

Voice communications, data transmission, and tracking continue during the second orbit. Another IMU alignment is performed 15 minutes before the planned injection time, the "go" decision is given by the ground, the crew secures the spacecraft, the S-IVB restart sequence begins, ullage rockets are fired, and the vehicle is ready for translunar injection (Figure 3).

TRANSLUNAR INJECTION

During the approximately $5\frac{1}{2}$ minutes of S-IVB burn for translunar injection, the spacecraft velocity will increase by more than 10,000 ft/sec. Altitude will increase to about 160 nautical miles and about 50° to 60° of longitude will be transversed.

Tracking by ground based stations during translunar injection will be available on many missions, but due to the large envelope of translunar injection points, tracking, even with a limited number of ships, will not always be possible. In any case, however, the spacecraft will be acquired and tracked by a MSFN station within no more than 7 or 8 minutes after translunar injection. However, voice communications and data transmission will be maintained during the injection phase by means of relay aircraft.

Having completed translunar injection, the spacecraft is now on the next plateau, translunar coast (Figure 24).

TRANSLUNAR COAST

The translunar injection maneuver was configured such as to place the spacecraft on a circumlunar coast trajectory which circumnavigates the moon and returns to a safe entry condition back at earth with no major intervening maneuvers required. This is called a "free return" trajectory and will be discussed in more detail in a later paper by M. P. Frank.

Immediately after injection the hydrogen and oxygen tanks on the S-IVB will be vented to a low pressure to assure that uncontrolled venting will not occur during the critical attitude operations for the next two hour period.

Following a quick systems status check after the end of injection, the first operation will be for the crew to reorient the vehicle in a direction favorable for docking illumination while at the same time maintaining communications with earth during the next two hour period. One additional constraint is that the maneuver sequences for this reorientation must avoid yawing the vehicle more than ±45° so as not to result in gimbal lock for the S-IVB inertial platform.

For the sun position in the 7° - 20° range at lunar landing, the transposition and docking operations will be in daylight, as shown in Figure 25. The reorientation maneuver will be such that the sun is incident on the LM docking tunnel for best visibility. The maneuver then is such that the vehicle is pitched through about 60 degrees. At this point, both the spacecraft and launch vehicle are communicating over their S-band omni antennas which, in fact, have a 30° to 40° null zone in both fron and back. In a short time, however (about 15 minutes after injection), S-IVB communications will be switched automatically from the omni antennas to directional antennas; this switchover to the directional antenna must be taken into consideration in the selection of the reorientation maneuver. Therefore, following the 60° pitch, there must be approximately a 180° roll maneuver to place the launch vehicle directional antenna in the proper position for transmission when the switchover is made. Actually, the pitch and roll maneuver will be done simultaneously, subject to the gimbal lock considerations. As discussed in the earth orbit phase, these maneuvers are performed at low rates in order not to require an excessive amount of S-IVB RCS propellant.

During this period of reorientation the earth will have acquired the vehicle and tracking will have been continuous except for a brief period during the 60° pitch maneuver when the omni antenna null zone swept through the ground station. A tracking period of about 10 minutes will be required for the ground to accurately determine the vehicle's orbit, and to provide a "go" decision for transposition and docking to proceed.

Having received the decision to proceed with transposition and docking, the Command and Service Modules separate from the S-IVB/LM combination using the SM RCS System. This separation severs the hardline control interface between the crew and the S-IVB; any further maneuvers of the S-IVB will need to come from the ground command. However, the orientation selected before separation was one which will not require any adjustments - at least for the first hour, during which transposition and docking will normally be completed. Present plans are to place the vehicle in an inertial attitude hold mode before separation, oriented so that the launch vehicle directional antenna continues to see the ground station as the spacecraft trajectory sweeps through about 45 degrees of central angle during the next 45 minutes. During this period the S-IVB directional antenna has been switched to its narrow beam, but communications are maintained. During the second hour of the transposition and docking phase, which is provided for contingencies, it may be necessary to reorient the S-IVB (from the ground) to maintain communications with the launch vehicle.

Returning to the transposition and docking sequence, Figure 26 illustrates the separation of the CSM from the S-IVB-LM. The adapter panels are deployed as part of the separation sequence and are held at a 45° angle with respect to the longitudinal axis. This is sufficient to clear the LM and allow a clean extraction maneuver. If the petals are folded back completely, then they shroud the S-IVB antennas located around the periphery of the Instrument Unit.

The Service Module RCS is used to translate some 100 feet away from the S-IVB. At this point, the translation is stopped and the CSM is rolled to the proper indexing for docking and then pitched 180° so that it is pointing back at the LM (Figure 27). The roll-pitch sequence rather than a pitch-roll sequence is used to avoid placing the spacecraft omni antenna null zone at the MSFN station during the 180° pitch maneuver. In the case of the spacecraft, yaw maneuvers must be restricted to less than ±70° to avoid gimbal lock. Unlike maneuvers during the rest of the mission, the turnaround is done at the rapid rate of 5 degrees per second in order to reduce the time required for transposition and docking; and, in particular, to minimuze the time the crew is out of line of sight of the launch vehicle in the separated condition.

Having turned around, the crew will now deploy the spacecraft high gain steerable antenna and orient it to earth before closing on the LM for docking. This is required because the spacecraft omni antennas will be blanketed by the adapter petals once the CSM gets in close to the LM for docking; therefore, communications with the ground will have to be maintained using the high-gain antenna.

The docking operation continues under the manual control of the spacecraft crew as the final translation is made and the CSM slowly closes on the LM/S-IVB. Initial contact is made when the docking proble on the CM engages the drogue mounted on the LM; the docking mechanism pulls the two vehicles firmly together the final few inches, four latches automatically engage, and the initial soft docking is completed (Figure 28). The next step will be to manually hook up the CM-LM umbilicals, and complete the latching operation by manually engaging 8 more latches. The functions of the umbilicals are twofold: first to supply the hardline connection between the CM controls and the explosive ties which attach the LM to the adapter. These ties will be severed by crew command when they get ready to withdraw the LM. The other function of the umbilicals is to supply a small electrical power level to certain LM equipment from the CSM power source during the translunar coast phase of the mission. The chief user of this power are small heaters in the LM IMU which needs to be maintained within narrow temperature limits at all times. This permits use of smaller batteries in the LM than would otherwise be required, so the LM will remain inactive for most of the translunar phase. The operations during the docked period prior to separation will require about 20 minutes and will, of course, necessitate one of the crewmen leaving his couch and moving to the area of the docking tunnel. There is less inherent radiation protection from equipment and storage in the tunnel area for the crew; however, these operations do not require the crewman to be in this area for very long, and, further, this period coincides with the passage between the inner and outer radiation belts where the radiation level is lowest. Hence, there are no radiation constraints as a result of this operation.

A schematic representation of the docking indexing is shown in Figure 29. The CSM-LM axes are offset by 60°. This allows the CM pilot to line up on the docking target located on the LM. Similarly, when docking in lunar oribt, this indexing will allow the LM pilot to see the docking target mounted in the CM right hand window. Also shown in the diagram is the CSM high gain antenna which, unless the pitch attitude is properly selected, will be shrouded by the adapter panels. The S-IVB high gain antenna is located in this same quadrant (along the -Z axis of LM).

After completion of hard docking, the LM attachment ties are severed, and the LM is extracted from the adapter using the SM RCS system to bakc away (Figure 30). Approximately 3 ft/sec. separation velocity is applied by the RCS which is sufficiently high to virtually assure no problem of subsequent recontact with the S-IVB.

It would be well here to point out a general characteristic of the Apollo spacecraft - namely, its large radius of gyration in pitch and yaw, its small value in roll. This being the lunar vehicle, it is not close coupled as Mercury and Gemini, except in roll.

One deg/sec. rate costs 11 lbs. in pitch and yaw and 1 lb. in roll (to start and stop). The 5°/sec rate referred to earlier, then, cost about 50 lbs. This is a large price to pay for that simple maneuver, but experience indicates that long periods remote from the station-keeping target should be avoided. It is illustrative of the cost per maneuver. This cost leads to pre-planned maneuvers to take advantage of the low inertia in roll and to think through each maneuver to minimize propellant consumption. This frugal use of RCS reactant is mandatory until the possible requirement for LM rescue in lunar orbit has passed.

At this point the spacecraft is about one hour past the translunar injection point at an altitude of about 9,000 nautical miles. Inertial velocity has decreased to about 23,000 ft/sec. or about 13,000 ft/sec. less than at injection cutoff. The velocity will, of course, continue to decrease for most of the trip until the spacecraft nears the moon.

The spacecraft is now being tracked by one of the three deep space stations with 85 ft. dishes (Madrid, Canberra, or Goldstone), and two of the unified S-band stations with 35 ft. dishes. Similar tracking coverage will be available throughout the rest of the mission back to entry, except for periods when the spacecraft is behind the moon.

The first midcourse correction will be made in about two hours, after the spacecraft's trajectory has been accurately determined by extensive ground tracking. During this period, the crew will make a series of star-landmark sightings to check out their space mode of navigation, which is a backup to the ground navigation.

The time of the first midcourse correction is not a critical event. Delaying the correction will, of course, allow the initial injection errors to grow, so that a larger delta V will be required for the correction once it is made; however, it is not extremely sensitive in the range of 3 to 5 hours after injection. In some cases, it will even be preferred to wait; if the injection has been particularly good, the delta V required for the early correction may be so low that it could not be performed accurately with the 20,000 lbs. thrust SPS engine (less than about 4 ft/sec.). Such small corrections could be made with the SM RCS engines but it would be preferred to conserve RCS propellant wherever possible, even at the expense of SPS propellant, where the reserves are considerably greater.

The typical midcourse correction, then, will be done with the SPS (Figure 31) about three hours after translunar injection, and will require a delta V of about 25 ft/sec., based on analyses of expected injection accuracies. This corresponds to about 3 seconds burn time by the SPS, and could occur in any direction.

Following the midcourse correction, the spacecraft is set up for the long coast period ahead. Another correction is not expected to be required until the spacecraft nears the moon, about 2½ days later.

The first operation to be performed is to orient the spacecraft for passive thermal control. The object of passive thermal control is, of course, to insure that critical components in the spacecraft do not get too hot or too cold during the long coast period, as a result of either looking directly at the sun or directly away from the sun for long periods of time. For example, the SM-RCS propellant valves are located adjacent to the SM skin, and their temperature is relatively sensitive to spacecraft orientation (even though they are insulated). Heaters are provided for the valves to accommodate the unavoidable thermal cycling experienced in lunar orbit, but the use of passive thermal control during the translunar and transearth coast phases, allows conservation of electrical energy which would otherwise be required for heater operation. More important, however, the use of passive thermal control negates the necessity for an active coolant loop to cool these same sensitive components.

Passive thermal control is usually referred to by the less technical, but more descriptive, term of "barbequing". The orientation maneuver is made such that the spacecraft longitudinal axis is placed perpendicular to the vehicle-sun line (figure 32). The orientation of the longitudinal axis about the sun vector is chosen to minimize the interference of the subsequent roll maneuver with high gain antenna coverage. After stabilizing the spacecraft in this orientation, a slow rotational rate about the X-axis is established to achieve the desired thermal cycling.

As a result of small residual rates about all these axes when the spacecraft was "stabilized", and as a result of other sources of disturbances such as fuel slosh and steam venting, the vehicle will begin slowly to precess about its angular momentum vector. Current analysis indicate that a roll rate of about 2.5 revolutions per hour will be required in order to maintain the spacecraft YZ plans within 20° of the sun line, which is the tolerance required to maintain effective thermal control.

It is emphasized that the thermal cycling operation can be interrupted for periods of up to three hours, provided these attitude hold periods are followed by an appropriate period of barbequing (5 to 7 times the hold period). In addition, the thermal design is such that a three hour period prior to lunar orbit insertion and prior to entry can be accommodated without the necessity of subsequent thermal cycling.

Following the establishment of the barbeque mode, operations onboard the spacecraft will settle down to a routine for the next $2\frac{1}{2}$ days. Periodic systems status checks will be performed by the crew, the spacecraft's position and velocity will continue to be monitored by the ground, and data will be transmitted continuously by the spacecraft. For the sun's position such as to give us a 7° to 20° elevation angle at lunar landing, the passive thermal control maneuver can usually be set up such as to provide continuous coverage of the ground station with the high-gain antenna during the translunar coast phase (except for sites west of about 20° W longitude). This will not be the case for the transearth coast phase, but the situation is not considered to be a problem since the ground station (because of its higher radiative power) will always be able to contact the spacecraft over the omni antennas, and request that the roll maneuver be discontinued for a period, if continuous communications were required for some reason.

A second midcourse correction will be made about one hour after entering the moon's sphere of influence, several hours before reaching lunar orbit. It is executed with the SPS in the same manner described previously.

Following another rest period, the crew begins a period of considerable activity which will continue for the next several hours through lunar landing and the first exploration period. These activities will be discussed in detail in a later paper by Mr. Loftus, but the highlights will be mentioned here.

About three hours before lunar orbit insertion, the LM will be checked out. After pressurizing the LM, which has leaked down during the long coast period, one of the three crew members will transfer to the LM through the tunnel and begin an activation and checkout of the LM systems (figure 33). This is considered desirable for two reasons: First, it provides the ground and the crew with the first knowledge since leaving the launch pad that the LM systems will indeed be able to function properly for a lunar landing. The discovery of some system malfunction which would preclude lunar landing may be sufficient reason not to commit the mission to lunar orbit.

The second reason for LM checkout prior to lunar orbit is to assure that the descent propulsion system is available as a ready means of abort in the event of SPS failure during lunar orbit insertion. Since the abort mode using the descent propulsion system also requires the use of the LM guidance and control system, it will be necessary to activate, checkout and align this system also. Similarly, the Environmental Control and Electrical Power Systems will need to be activated. A second crewman will join the first when freedom from his duties in the CM permits.

Continuous ground tracking confirms that the spacecraft is indeed on the proper course and the decision is made to proceed to lunar orbit insertion. The LM crew has returned to the CM by this time and the next operation is to orient for lunar orbit insertion and prepare the spacecraft for SPS thrusting. A few minutes later, sunset occurs and shortly thereafter the spacecraft passes out of earth line of sight. Three or four minutes later, SPS ignition occurs for lunar orbit insertion (figure 34).

LUNAR ORBIT INSERTION

The spacecraft altitude at this point is typically 150 nautical miles. Burn duration for the lunar orbit insertion is typically somewhat over six minutes, for a delta V of 3200 ft/sec. Corresponding SPS propellant for this maneuver is about 24,000 lbs. During the insertion maneuver the spacecraft has travelled through a central angle of about 20°, and ends up in a circular orbit of 80 nautical mile altitude. During the insertion, a plan change was executed to make the orbit inclination such that the spacecraft orbit passes over the intended landing site on the third revolution. This situation allows an in-plane descent with the LM on this third pass.

LUNAR ORBIT COAST PRE-SEPARATION

The spacecraft is now on another plateau, lunar orbit coast (figure 35). The CSM and LM will remain in the attached condition in lunar orbit for the next $5\frac{1}{2}$ hours (figure 36). During this time, three passes over the front side of the moon will have been made, and the spacecraft will have been tracked by the earth on each pass. This will have been more than sufficient to accurately determine the lunar orbit parameters. In order to reduce the uncertainties in the selenographic position and altitude of the landing site, and to enhance confidence, a series of onboard landmark sightings in the vicinity of the landing site will be made during two of these passes. This point will be discussed in more detail in the paper by Mr. Cheatham.

Communications with earth and data transmission will have been maintained on each pass in front of the moon. During periods when the spacecraft is behind the moon, data is recorded onboard for subsequent playback when line of sight is reacquired.

Additional operations during this period will involve the crew going back into the LM, and activating and checking out those systems not checked out before lunar orbit insertion. After transferring certain equipment to be used on the lunar surface from the CM to the LM, the LM platform is aligned, information in the CM computer is transferred to the LM computer, the hatches are closed, and preparation is made for separation.

CSM/LM SEPARATION

Separation is performed by the LM RCS system (5 seconds burn) about 30 minutes prior to the actual time of LM transfer orbit insertion. The separation delta V is small, about 1 ft/sec., so that the two vehicles drift apart very gradually (figure 37). When the vehicles are about 60 feet apart, the LM will pitch up to an attitude which allows the CM crewman visually to inspect (with the sextant) the external portion of the LM including the landing gear and probes.

A few minutes later the vehicles are far enough apart to perform a checkout of the LM rendezvous radar and CSM transponder. Also, during this separated period, the LM platform is fine aligned, the controls and displays are checked out in the LM-alone configuration, and preparations are made for transfer orbit insertion.

Before the LM passed out of line of sight of earth, data transmission and voice communications had been maintained directly using the LM S-band high gain antenna. After losing line of sight, the LM data will be transmitted to the CSM where it is recorded for subsequent playback. This will be the situation during all phases when the LM is behind the moon, since it carries no onboard data recorder of its own.

The separation maneuver was made in such a direction as to place the LM ahead and below the CSM throughout the 30 minutes coast period to the transfer orbit insertion point. This type of separation maneuver avoids the possibility of jet impingement or collision during the transfer burn by increasing the LM/CSM range (for a small delta V) as compared to the forward or rearward separation methods. Another advantage is that it provides clear VHF communications between CSM and LM during the coast and during the transfer maneuver. The two vehicles are about one-quarter mile apart at the time of transfer orbit insertion.

TRANSFER ORBIT INSERTION

Prior to the engine burn, propellant settling is provided by a five second RCS firing of the four X-axis thrusters.

The insertion maneuver takes place behind the moon about 200° central angle from the landing site (figure 38). Descent engine thrust is maintained at 30% for the first three seconds and is then reduced to 10% for the next 23 seconds and is then increased to 92.5% thrust (9700 lbs.) for the final six seconds of burn. The period at low thrust is required due to the possibility that the vehicle's center of gravity deviates from its nominal location and that the engine thrust vector is not acting through the c.g. Since the engine gimbals are used for trim control only and are relatively slow, the engine thrust is reduced to insure that the moment control of the RCS is not exceeded during the initial travel. The high thrust at the end of the burn allows a check to be made on the engine through its full throttling range.

The fuel expended during the insertion maneuver is about 330 lbs., and the corresponding delta V is slightly less than 100 ft/sec.

COASTING DESCENT

The spacecraft is now on the next plateau, descent coast (figure 39). At this point, the LM is on a slowly descending trajectory which will continue for about the next hour until it reaches its pericynthion altitude of 50,000 ft. During this period, the LM will track the CSM with its Rendezvous Radar and determine its descent trajectory onboard. Similarly, the CM can track the LM flashing light with its sextant and perform an independent determination of the LM orbit. Finally, when the LM comes within line of sight of earth, then the ground station will track the vehicle and provide the LM with the final source of navigation data. As in all operations discussed up to this point, the ground based navigation is the primary source of data.

If the decision is made not to initiate powered descent, the LM is in a safe orbit from which it could subsequently rendezvous with the CSM, or if necessary, the CSM could perform a rescue. Having decided to continue descent, however, another IMU alignment will be made and preparations for powered descent begin. Up to this point in the coasting descent, the LM has been leading the CSM. At pericynthion, this lead angle is about 10 degrees. From this point on, however, once the LM begins powered descent, the CSM will catch up and finally go ahead of the LM during the latter part of the landing maneuver.

POWERED DESCENT

At the 50,000 ft. pericynthion altitude, a propellant settling maneuver is performed with a 5-second RCS firing, followed by descent engine ignition. The thrust profile is the same as before with a 3-second burn at 30% thrust, followed by a 28-second period at 10% thrust, and then increased to 92½% thrust (9700 lbs.). The powered descent phase will be discussed in detail in a later presentation by Mr. Cheatham, so only the gross profile will be described here.

The powered descent is divided into three distinct portions, called the braking phase, the final approach phase, and the landing phase.

Phase I, the braking phase is designed primarily for efficient reduction of the orbital velocity and is therefore performed at maximum thrust with near horizontal flight path angles (figure 40). It is the longest of the three phases - lasting almost 8 minutes while covering almost 250 miles, down to an altitude of about 8600 feet.

Phase II, the final approach phase, begins at the 8600 ft. point, called "high gate", about 8 nautical miles range from the landing site. It begins with a pitch maneuver which brings the horizon and the landing site into the pilot's view for the first time (figure 41). The purpose of this phase is to provide the crew time to assess the trajectory as the LM approaches the surface, to provide the crew the opportunity to assess the landing area, and to allow for pilot takeover of the control tasks if required. The throttle is reduced back to 60% thrust during this phase. Duration of this phase is somewhat less than $1\frac{1}{2}$ minutes. Forward velocity is reduced from about 450 ft/sec. at the beginning to about 50 ft/sec at the end. Altitude is 500 ft. at the end of the final approach and the range to the landing site is about 1200 ft.

Phase III, the landing phase, is designed specifically for pilot control and provides the capability for making a detailed assessment of the landing site. The vehicle is pitched back to a near vertical attitude, the thrust is reduced, vertical and horizontal velocities are reduced, and a vertical descent is made from the last 100 ft. altitude (figures 42 and 43). Duration of this phase is nominally about 75 seconds, but the capability exists to extend it longer if more landing site assessment time or small redesignations are required.

Fuel used during the entire powered descent is around 16,000 lbs. corresponding to a delta V of about 6600 ft/sec.

LUNAR SURFACE STAY

At this point, the spacecraft is on the next plateau, lunar stay (figure 44). Following an assessment of the vehicle situation to determine if there is a necessity for an early abort, the postlanding checkout is begun. The descent engine is disarmed and the descent propellant tanks are vented. The IMU is aligned and placed on standby operation. Systems not required during the lunar surface stay are shut down.

Following a period of coordination with the ground, a thorough check out of the Extravehicular Mobility Unit is performed, an EMU is donned by each crewman and preparations are made for egress. Life support is switched to the Portable Life Support System, the cabin is depressurized, and the forward hatch opened. This occurs about 1-3/4 hours after landing.

The lunar surface stay time on the first mission is planned for approximately 18 hours (figure 45). During this time, two 3-hour exploration periods are provided for each astronaut to be done simultaneously - for a total of 12 manhours of lunar surface activity. The scientific objectives and activities will be the subject of a later paper, but the general activities will be described here.

The initial portion of the first exploration will be occupied with a general external inspection of the vehicle. Measurements will be made of the landing gear stroke, depression made by the pads, any evidence of sliding will be noted, etc., for the kind of engineering measurements that might be helpful for future landings. This will require about $\frac{1}{2}$ hour. During this time, the second astronaut has stationed himself on the forward boarding platform and is making a detailed description and photographic record of the lunar terrain from his vantage point. Both astronauts are in voice contact with the ground during this period.

The next step will involve unloading equipment, including the Lunar Surface Experiments Package from the descent stage storage bay. The lunar surface erectable high gain antenna will be set up so that continued data transmission from the LM can take place at a lower power. Television pictures can be transmitted during this exploration as time permits.

In keeping with the stated scientific priorities, the first scientific task will be to gather lunar samples. One of the two specimen return containers will be filled during this first exploration, and stored in the LM cabin at the end. This insures that at least part of the scientific objectives will be met in the event some malfunction prevented a second exploration period.

If time permits, and if the work load has not been excessive, then one of the astronauts can begin the LSEP deployment, while the other collects lunar samples in the same vicinity. The actual LSEP deployment will be discussed in a later paper by Mr. Vale.

At the end of the first exploration the two crewmen return to the LM, pressurize the cabin and remove the EMU. One of the PLSS units is put on recharge while the crew has a meal, and the other is recharging during the 6 hour sleep period which follows.

Following the 6 hour sleep period, the crew will have another meal, check and don the EMU, and prepare for the second exploration period. During the second exploration, the LSEP will be deployed, and a more selective collection of lunar samples will be made, filling the second specimen return container. The astronauts will be able to range somewhat further from the landing site on this exploration.

During the period that the LM is on the lunar surface, the astronaut in the CM performs periodic systems checks of the CSM systems, and maintains communications directly with earth and indirectly with LM over the earth relay link.

Returning to the LM, following the last exploration period, the prelaunch checkout is conducted wherein all systems are activated, checked out, and prepared for launch. The Rendezvous Radar will have been checked by tracking the CSM on its previous pass over the landing site prior to launch.

LUNAR ASCENT

Ascent engine ignition will occur at the beginning of the 5½ minute launch window when the CSM leads the LM by about 9 degrees (figure 46). The ascent engine propellant tanks are pressurized, propellant valves are opened, and the structural ties and umbilical between the descent and ascent stage are severed by explosive charges.

Following a 12 second vertical rise period, the guidance system commands the LM attitude to an optimum profile designed to boost the vehicle to orbital velocity. The ascent engine has a fixed thrust of 3500 lbs., and is non-gimballed, so moment control must be provided by the RCS engines. The main ascent trajectory is a standard one which ends at 50,000 ft. altitude with a slight overspeed such that the resultant coast trajectory is an ellipse with about a 30 nautical mile apocynthion and a 50,000 ft. pericynthion. Hence, at the end of the main ascent, the LM is on a safe trajectory, or plateau, which will not impact the moon even if the subsequent burns are not performed (figure 47). That is, it is in a relatively stable situation from which a rescue by the CSM could be performed, if necessary.

The main ascent burn is typically of 6½ minutes duration during which about 4800 lbs. of propellant is consumed. Delta V is typically about 6000 ft/sec.

The ascent maneuvers are illustrated in figure 48. Following the engine shutdown, the LM acquires and tracks the CSM with the Rendezvous Radar to determine its orbit. At this point, the CSM is still leading the LM and is about 350 miles away. The ground has been tracking the LM during the entire ascent and continues to do so. Based on this tracking, a determination will be made of the next maneuver to be performed some thirty minutes after beginning of coast.

This next maneuver is called the concentric sequence initiation or CSI. It is a relatively small maneuver made with the RCS jets, and is designed to raise the pericynthion of the LM orbit to an altitude consistent with that required for proper phasing with the SM. At the same time, it is somewhat like a midcourse correction in that it will be calculated to absorb the trajectory dispersions resulting from the main ascent. For an on-time launch (beginning of launch window) the pericynthion altitude will be raised to 65 n. miles with the CSI maneuver. This will require about 60 ft/sec. delta V, consuming about 40 lbs. of RCS propellant; burn time for the two RCS thrusters would be about 50 seconds. If launch had not occurred until the end of the launch window, then the proper phasing altitude would have been 30 n. miles, so no CSI manuever would have been required in this case, except as a small correction to absorb the launch dispersions.

Depending on the landing site longitude, line of sight to the earth has probably been lost by this time. The LM to CSM range has been reduced to slightly less that 200 n. miles at the CSI point, with the CSM leading. The LM continues to track the CSM with the Rendezvous Radar and preparations are made for the next step in the ascent sequence which will occur when the LM reaches the high point in its new orbit about 50 minutes after CSI. At this point, another maneuver is made which circularizes the LM orbit at 65 n. miles (for the on-time launch case), such that the LM and CSM orbits are concentric and separated by an altitude of 15 n. miles.

Actual range to the CSM is about 50 n. miles at this point. The circularization maneuver uses about 45 lbs. of RCS propellant with a corresponding delta V of 65 ft/sec. Burn duration is about one minute.

Following the circularization maneuver the LM continues to track the CSM and shortly emerges from behind the moon, at which time tracking from the ground station is resumed.

Based on the LM onboard tracking and the ground based tracking, the conditions are determined for the next maneuver, called the terminal phase initiation or TPI. This maneuver is made about 20 to 30 minutes from the time circularization occurs and is designed to place the LM on an intercept trajectory with the CSM about 140° away from the initiation.

The range to the CSM is about 30 n. miles at the initiation of the transfer maneuver. The transfer burn is made with the RCS jets and amounts to about 23 ft/sec.; 15 lbs. of propellant are consumed and burn duration is about 20 seconds.

Following the burn, tracking is performed both on-board and by the ground, and based on this information, a small mid-course correction will be made about 10 to 15 minutes after the initial burn.

During the next 35 minutes, the range to the CSM will be reduced from 30 n. miles to about 3 n. miles, and the range rate to the CSM will be reduced from about 130 ft/sec. to about 30 ft/sec.

A series of range - range rate gates are specified such that (1) at the 3 n. mile point an RCS burn reduces the range rate to 20 ft/sec.; (2) at 1 n. mile range a short burn reduces the range rate to 10 ft/sec.; and (3) at 500 ft. the rate is reduced to 5 ft/sec. These burns are all in the range of 4 to 8 sec. and consume a total of about 20 lbs. of propellant.

DOCKING

Having passed this last gate, the vehicles are in close proximity at a low relative velocity, and the manual docking phase begins (figure 49). The capability exists for docking in darkness, (lights are provided on-board and the docking target is luminescent); however, in most cases, it will be preferred to wait a few minutes until the vehicles come into sunlight. The relative rates of the two vehicles will be nulled until this time.

At a range of about 50 ft., the LEM will be pitched back so that the CM becomes visible in the overhead window. The pilot then translates toward the CSM at a low rate and engages the CM probe in the LEM drogue. The docking tunnel is pressurized, the upper hatches are removed, and the docking latches secured (figure 50).

LUNAR ORBIT COAST AFTER DOCKING

The spacecraft is now on another plateau, and the capability exists to remain on this plateau for an extended period if desired. However, the mission will normally proceed on to the next commit point within a few hours.

During the next hour or so, the LM is deactivated, the crew and equipment are transferred to the CSM, and finally the LM is jettisoned by firing a shaped charge which separates the CM and LM, and the CSM translates away with the SM-RCS (figure 52).

The first opportunity for transearth injection will occur in about one hour, but injection will not normally be planned for this first opportunity, since systems readiness checks, exchange of data with the earth, and IMU alignment have yet to be performed. So the spacecraft continues in lunar orbit for one more revolution before transearth injection.

TRANSEARTH INJECTION

Transearth injection occurs on the back side of the moon with respect to the earth, and will normally be in the dark. The SPS burn is preceded by a 14 second ullage burn from the SM-RCS to settle the propellants. This ullage manuever was not required on the previous SPS burns on the way to the moon since the tanks were essentially full, but now they are only about one-third full.

Transearth injection delta V will vary from about 2600 ft/sec. to about 3200 ft/sec. depending on whether the return trajectory is a relatively slow (110 hr.) or a relatively fast (86 hr.) transfer. The 24 hour flexibility is necessary to allow a return to the primary recovery area on earth (in the vicinity of Hawaii) from any mission.

A typical value for propellant consumed during transearth injection is about 8000 lbs. with an SPS burn time of about 2 minutes.

TRANSEARTH COAST

Soon after the end of the burn, the spacecraft comes within line of sight of earth and continuous tracking begins. Data recorded on board the CSM during the injection burn is played back to the ground station, the CSM is powered down, the passive thermal control maneuver is initiated, and the crew goes to sleep. The spacecraft is now on another mission plateau, transearth coast (figures 54 and 55).

Operations during the transearth coast phse are similar to those described during translunar coast, with few notable exceptions:

First, the position of the sun relative to the transearth trajectory will be such as to result in hi-gain antenna communications loss during each revolution of the thermal cycling maneuver. Typical values for a 2.5 revolution per hour roll rate would be loss of earth for 4 minutes out of every 24 minute revolution. As mentioned in the previous discussion, this is not considered a problem, since the ground can contact the spacecraft at any time and request that a communications attitude be held. Also, it is likely that voice and low bit rate telemetry can be maintained over the omni-antennas for much of the transearth phase.

The other major difference from the translunar phase concerns the midcourse corrections. In this case, it is almost certain that the corrections will need to be made with the SM-RCS. The spacecraft is considerably lighter now, but the minimum impulse capability is the same, so that the minimum delta V which can be performed with the SPS is 12 ft/sec.; the addition of 5 ft/sec. to this value for the ullage maneuver results in the fact that the midcourse correction must be at least 17 ft/sec. before it can be performed with the SPS. Error analysis of the MSFN tracking capability and the SPS cut off errors indicate that the corrections will be considerably less than 17 ft/sec.; hence, the plan will be to perform these corrections with the SM-RCS (figure 56). As in the translunar case, two corrections will probably suffice: the first about 10 hours after injection and the second about 2 hours before entry.

About 15 minutes prior to the entry interface (400,000 ft.) the SM is jettisoned when the spacecraft is some 2500 nautical miles from earth (figure 57). The spacecraft is oriented for SM jettison in such a way that the 3.5 ft/sec. separation velocity applied by the SM RCS jets places it on a path which minimizes the probability of subsequent recontact with the CM. The CM is then oriented to the entry attitude using its own RCS jets (figure 58).

ENTRY

Entry will normally begin over the western Pacific at about 400,000 ft. altitude. Range from this point to splashdown will be from 1500 to 2500 n. miles, and it will be controlled by varying the direction and time application of the spacecraft lift (figure 59).

A ground based station at Guam or in Australia, depending on the inclination of the approach path, will track the spacecraft just prior to entry, but the entry phase itself will normally not be in line of sight of a ground based station. Tracking during entry will be provided by two ships positioned along the entry path. This insures that tracking is continuous during this period, except possibly during the blackout period.

EARTH LANDING PHASE

The earth landing sequence begins at 24,000 ft. The forward heat shield is jettisoned, which exposes the CM bay where the parachutes are stored. The two drogue chutes are deployed immediately thereafter and are disreefed a few seconds later (figure 60). The drogue chutes serve to orient the CM properly for main chute deployment, and reduce the velocity from about 400 ft/sec. at deployment to about 200 ft/sec. at 10,000 ft. altitude where the main chutes are deployed.

Three pilot chutes pull out the three main chutes and the drogue chutes are disconnected (figure 61). A few seconds later, the main chutes are disreefed and the descent rate is reduced to about 25 ft/sec. at splashdown.

Recovery is soon effected and the crew and spacecraft are taken aboard ship (figure 62).

DELTA VELOCITY BUDGET AND SPACECRAFT WEIGHTS

In the previous sections concerning the mission description, there were many references to the delta V and propellant requirements for the various propulsive maneuvers during the lunar landing mission. It will be of interest at this point briefly to summarize the overall delta V budget and spacecraft weight data currently being considered for the mission.

DELTA V BUDGET

Figures 63 and 64 show the budgets currently specified for lunar mission planning for the Service Module and Lunar Module, respectively.

A comparison of the minimum possible delta V requirements with the budgeted values indicates that considerable flexibility exists to accommodate the items listed in the figures.

SPACECRAFT WEIGHTS

Figure 65 shows the weight breakdown of the spacecraft configuration as it appears just following translunar injection. The three columns of "current," "predicted," and "maximum" injected weight represent a range of total spacecraft weights to be used for various mission planning purposes. The "current" weight represents the best estimate of what the spacecraft would weigh today, based on present mission requirements. Propellant tanks are not full for this weight but sufficient propellant is included to meet the delta V budget specified in Figures 63 and 64.

The "predicted" weight shown in the center column of Figure 65 is a tentative agreement with the Marshall Space Flight Center to be used for mission planning purposes. Some investigation has to be conducted to understand whether the capability really exists in both the MSFC and the MSC vehicles to handle weights of this magnitude.

The "maximum" weight shown is that weight which MSFC is analyzing presently to determine if the launch vehicle can perform with a payload of 100,000 pounds. This appears to be primarily a structural problem.

Figure 66 is a breakdown of the total Lunar Module weight that was shown included in the spacecraft weight data of Figure 65.

Questions and Answers

GENERAL MISSION SUMMARY AND CONFIGURATION DESCRIPTION

Speaker: Owen E. Maynard

1. General Phillips - Is the Saturn V recycle time quoted in the presentation realistic?

   ACTION - MSC will work with MSFC to verify the recycle requirements.

2. Dr. Haeussermann - Wouldn't direct ascent to translunar injection have advantage over the multiple earth parking orbits? How advantageous is the checkout in earth orbit? What are the trajectory related implications?

   ANSWER - This answer was prepared after the presentation. A significant payload gain can be achieved for the launch vehicle if a direct translunar injection is used; however, to be constrained to using only translunar injections would result in unacceptable limited launch opportunities as was indicated in the presentation. To perform an efficient translunar injection the burn should be in the vicinity of the moon's antipode. This means that for direct injections we could only consider launch dates when the moon's antipode was in the vicinity of the nominal orbital insertion position. Under these conditions we could conduct the S-IVB burns to provide the translunar injection velocity. To estimate the limitations on launch opportunities we can look at where the nominal insertion would occur. Launch opportunities will exist only when the antipode is in this vivinity. This means that the earth launch could occur only when the moon is in the vicinity of the maximum southern declination. It is estimated that there would be from 8 to 10 days each month in which the moon's antipode would be far enough north to be in the proximity of the nominal insertion position. If these 8 to 10 days did not match the days for which the required lunar lighting conditions were met then the mission could not be conducted. The first six months of the years 1968 - 1969, the current sun elevation requirements are met when the moon is at north declination. Since the direct injection is incompatible with northern lunar declination the mission would not be possible during this period. Also, as was indicated in the presentation, only night launches would be possible for these two years.

In addition to the elimination of many of the launch opportunities, the duration of the daily windows would also be greatly reduced. To provide a daily window would require non-optimum steering of the launch vehicle. This would reduce much of the advertised payload gain that is possible with a direct injection. These daily windows would be of approximately 20 minutes duration which is much shorter than the $2\frac{1}{2}$ hours that is available if earth parking orbits are used.

3. Mr. Nix - What happens to the S-IVB after it has been jettisoned?

ANSWER - MSC has recommended that MSFC implement the following CSM-LME/S-IVB post separation maneuvers in the IU. MSC has also requested that these maneuvers be initiated by the S-IVB/IU only upon receipt of a ground command to preclude inadvertent initiative prior to separation.

Upon receipt of this ground command the S-IVB will maneuver to an attitude to optimize communications and separation distance (approximately 170° pitch and 180° roll). This attitude will be maintained inertially until loss of S-IVB attitude control. Once at this attitude, the IU will command a blow down of the LH_2 (non-propulsive) and LOX (propulsive) vents in order to minimize probability of recontact. The S-IVB Auxiliary Propulsion System (APS) will then be used to round off the translational velocity at about 3 ft/sec in order to simplify subsequent recontact calculations.

The above sequence combined with appropriate SPS midcourse precedures in failure mode cases will insure against CSM-LEM/S-IVB recontact. However, use of S-IVB venting and/or S-IVB APS will have only a minor effect on the S-IVB lunar impact probability. Current S-IVB targeting procedures result in about a 50/50 probability of lunar impact in order to optimize spacecraft payload. A payload penalty on the order of 1000 pounds would be required to significantly reduce this impact probability. This would result in targeting for a decreased S-IVB injection energy but maintaining free return capability. The required injection energy would be achieved through use of the CSM SPS during the first midcourse correction burn retargeting to a different free return trajectory.

Due to the significant payload penalty, current lunar mission planning has proceeded without a requirement to minimize S-IVB lunar impact.

NASA-S-66-6085 JUN

MISSION PLATEAUS

1 PRELAUNCH
2 EARTH PARKING ORBIT
3 TRANSLUNAR COAST
4 LUNAR ORBIT PRIOR TO LM DESCENT
5 LM DESCENT
6 LUNAR SURFACE STAY
7 LM ASCENT
8 LUNAR ORBIT SUBSEQUENT TO RENDEZVOUS
9 TRANSEARTH COAST

Fig. 1

NASA-S-66-6074 JUN

ENERGY REQUIREMENTS FOR LUNAR LANDING MISSION

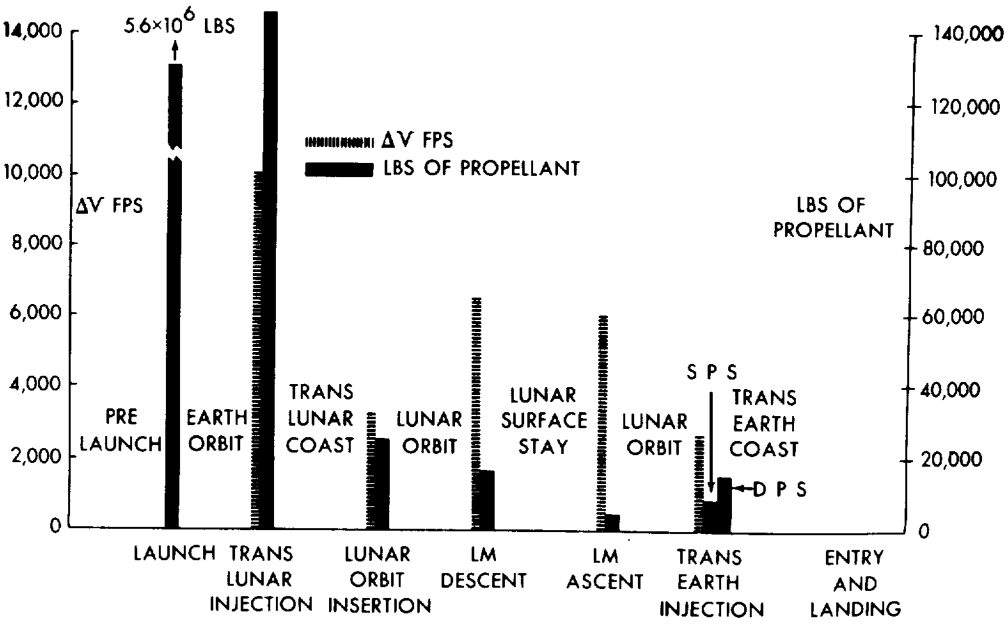


Fig. 2

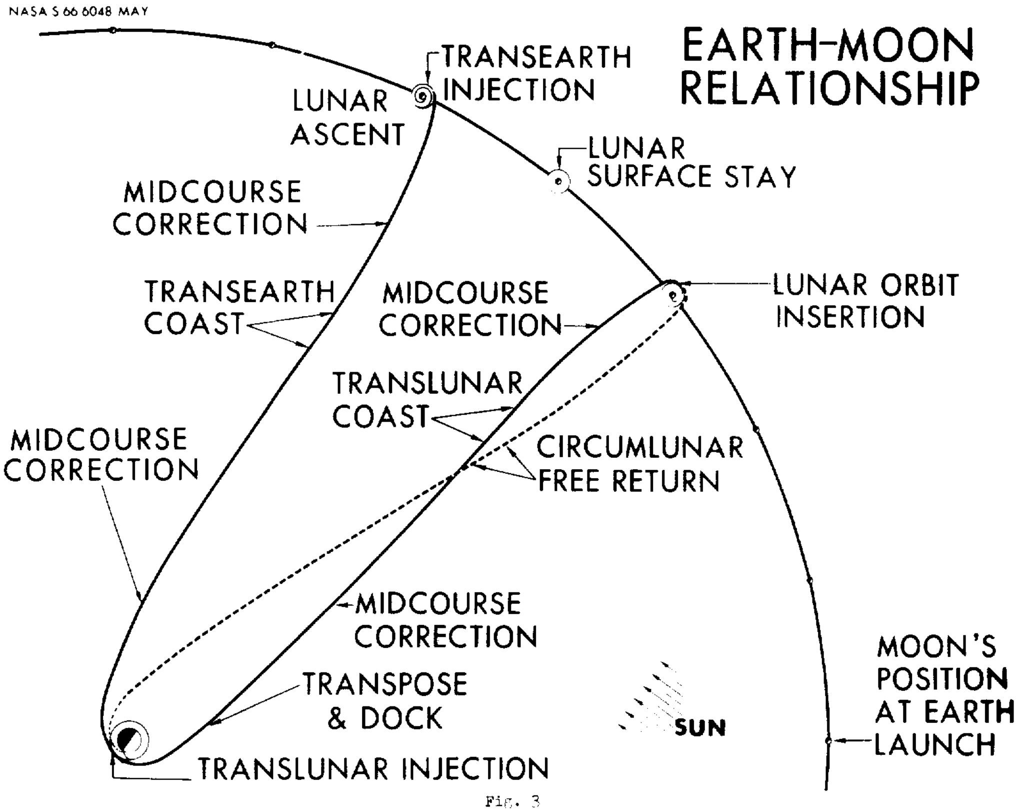

NASA S 66 6048 MAY
EARTH-MOON RELATIONSHIP
TRANSEARTH INJECTION
LUNAR ASCENT
LUNAR SURFACE STAY
MIDCOURSE CORRECTION
LUNAR ORBIT INSERTION
TRANSEARTH COAST
MIDCOURSE CORRECTION
TRANSLUNAR COAST
MIDCOURSE CORRECTION
CIRCUMLUNAR FREE RETURN
MIDCOURSE CORRECTION
TRANSPOSE & DOCK
SUN
MOON'S POSITION AT EARTH LAUNCH
TRANSLUNAR INJECTION

Fig. 3

NASA-S-66-5413 MAY 31

LAUNCH PHILOSOPHY

- PROBABILITY FOR A SUCCESSFUL COUNTDOWN AND LAUNCH IS ENHANCED WHEN WINDOWS ARE SPACED AT LEAST TWO DAYS APART;

  I.E., 1ST, 3RD, 5TH DAY -----

  1ST, 3RD, 6TH DAY -----

Fig. 4

NASA-S-66-6453 JUN

LUNAR LIGHTING AT TOUCH DOWN

- LIGHTING SHOULD BE NEAR OPTIMUM FOR TOUCH DOWN POINT INSPECTION BY CREW
- SUN ELEVATION RANGE IS SMALL = 7° TO 20° FOR HIGH CONTRAST

Fig. 5

NASA-S-66-5947 JUL

AREAS FOR APOLLO LANDING SITE ANALYSIS

▭ ORBITER A AND B SITES
⬚ FAVORED ORBITER A AND B SITES
○ SITES NOW USED IN MISSION PLANNING
X LUNAR LANDING AREA – NATURAL ENVIRONMENT AND PHYSICAL STANDARDS FOR THE APOLLO PROGRAM, APRIL 1965

Fig. 6

1968

NASA-S-66-7322 JUL

EARTH LAUNCH DATES

PACIFIC INJECTION - DAY & NIGHT (N) LAUNCHES

← WEST — EAST →

ORBITER SITE NO.	A-9	B-11	A-8	B-10	B-8	A-5	B-6	A-3	B-3	B-2	B-1	A-1
MIN NO. OF SITES	4	7	8	9	3	1	1	7	1	3	2	2
(JAN)-FEB	7(N)	6(N)	6(N)	6(N)	4	3	3	1	1	1	(31)	(31)
(FEB)-MAR	8(N)	7(N)	7(N)	6(N)	5(N)	4	4	2	2	1	1	(29)
(MAR)-APR	6	5-6	5	5	3-4	2	2	(31)	(31)	(31)	(30)	(30)
(APR)-MAY	6	5	5	5	3	2	2	(30)	(30)	(29)	(29)	–
(MAY)-JUN	5	4	4	3-4	2	1	1	(30)	(29)(30)	(29)	(28)	(28)
(JUN)-JUL	–	3-4	–	3	2	(30)	(30)	(28)	(28)	(27)	(27)	(27)
(JUL)-AUG	–	2	–	1-2	(31)	(30)	(30)	(28)	(28)	(27)	(26)	(26)
AUG	–	31	–	31	30	28	28	26	26	26	25	25
SEP	–	–	–	–	28	–	27	25	25	24	24	24
OCT	–	–	–	–	27-28	–	26	–	24	24	23	23
NOV	–	28	–	27	26	–	25	–	23	22	22	–
DEC	–	27(N)	–	27(N)	25 26(N)	–	24	–	22-23	22	21	–

Fig. 7

1968

NASA-S-66-7323 JUL

EARTH LAUNCH DATES

ATLANTIC INJECTIONS - NIGHT LAUNCHES ONLY

← WEST — EAST →

ORBITER SITE NO.	A-9	B-11	A-8	B-10	B-8	A-5	B-6	A-3	B-3	B-2	B-1	A-1
MIN NO. OF SITES	4	7	8	9	3	1	1	7	1	3	2	2
(JAN)-FEB	7	6	6	5	–	4	–	1	1	(31)	(30)	(31)
(FEB)-MAR	7	–	–	–	–	4	–	2	1	1	(29)	(29)
MAR	–	–	–	–	–	–	–	31	31	30	30	30
APR	–	–	–	–	–	–	–	30	–	29	28-29	28
(MAY)-JUN	5	–	4	–	–	–	–	(29)	–	(28)	(28)	(28)
(JUN)-JUL	4	3 - 4	4	3	2	(30)	(30)	(28)	(28)	(27)	(26)	(26)
(JUL)-AUG	3	2	2	2	(31)	(29)	(30)	(27)	(28)	(26)	(27)	(26)
(AUG)-SEP	–	1	–	(31)	(30)	(29)	(29)	(26)	(27)	(26)	(25)	(24)
SEP	–	30	–	30	28	27	27	25	25	24-25	24	23
OCT	–	30	30	29	28	27	27	25	25	24	23	23
NOV	29	28	27	28	26	24	25	23	23	23	22	21
DEC	27	27	27	27	25	24	24	22	22	22	21	21

Fig. 8

NASA-S-66-7320 JUL

1969

EARTH LAUNCH DATES

PACIFIC INJECTION - DAY & NIGHT (N) LAUNCHES

← WEST | EAST →

ORBITER SITE NO.	A-9	B-11	A-8	B-10	B-8	A-5	B-6	A-3	B-3	B-2	B-1	A-1
MIN NO. OF SITES	4	7	8	9	3	1	1	7	1	3	2	2
JAN	26(N)	26(N)	26(N)	25(N)	24(N)	23(N)	23(N)	21	21	20	20	20
FEB	25(N)	24(N)	24(N)	23(N)	22(N)	21	21	19	—	19	18	18
MAR	25(N)	25(N)	25(N)	24	23	—	22	—	—	19	19	19-18
APR	24	24	24	23	22	—	21	—	—	18	18	17
MAY	24	23	23	22	21	20	20	18	18	17	17	16
JUN	23	23	23-22	22	21	20	20-19	17	17	17	16	16
JUL	—	21	—	21	20-19	18	18	16	16	15	15	—
AUG	—	20	—	19	18	17	17	15	15	14	13	13
SEP	—	—	—	—	17-16	—	15	13	13	13	12	12
OCT	—	—	—	—	16	—	15	13	13	12(N)	12(N)	11(N)
NOV	—	17	—	17	16-15	—	14	—	12	12	11	11
DEC	—	15(N) 16(N)	—	15	14	13	13	11	11	10	10	10

Fig. 9

NASA-S-66-7319 JUL

1969

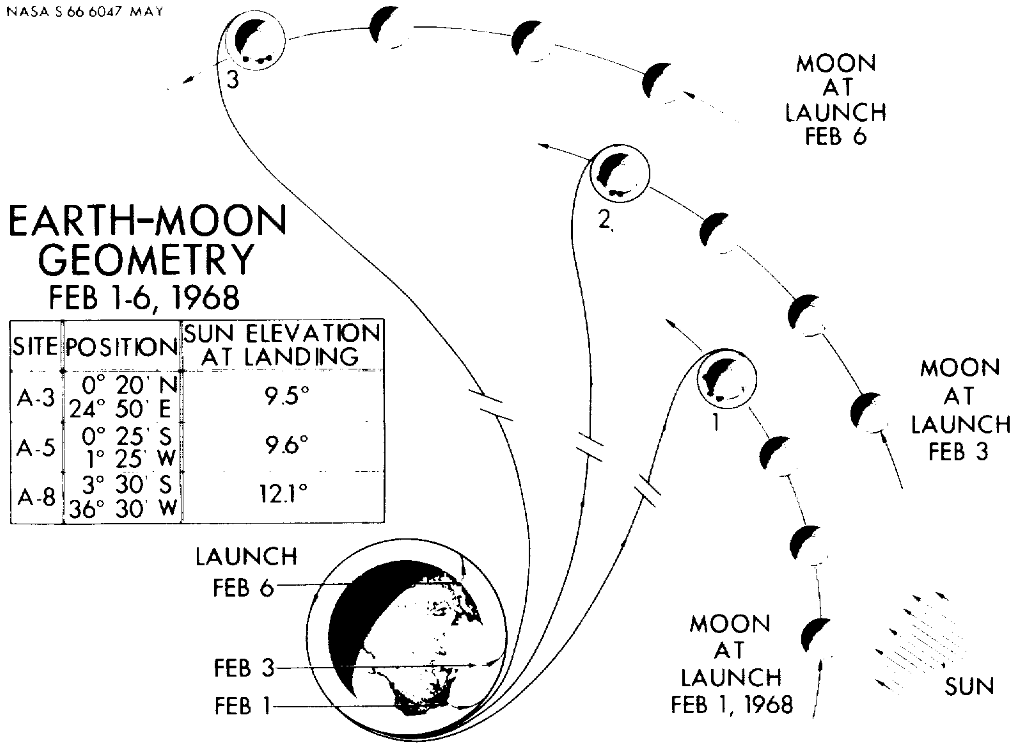
EARTH LAUNCH DATES

ATLANTIC INJECTION - NIGHT LAUNCHES ONLY

← WEST | EAST →

ORBITER SITE NO.	A-9	B-11	A-8	B-10	B-8	A-5	B-6	A-3	B-3	B-2	B-1	A-1
MIN NO. OF SITES	4	7	8	9	3	1	1	7	1	3	2	2
JAN	27	26	26	26	25	24-23	23	22-21	21	21	20	20
FEB	25	—	25	—	—	22	—	19	19	18	18	18
MAR	—	—	—	—	—	23	—	20	—	19	18	18
APR	—	—	—	—	—	—	—	—	—	17	17	17
MAY	24	—	23	—	—	—	—	19-17	—	17	16	16
JUN	24	—	23	—	—	—	—	18	18	16	16	16
JUL	22	22-21	21	21	20	19	18	16	16	16	15	15
AUG	21	20	20	20-19	18	17	17	15	15	14	14	14
SEP	—	19-18	19-18	18	17	16	16-15	14-13	13	13	12	12
OCT	—	18	—	18	16	15	15	13	13	12	12	12
NOV	17	17	17	16	15	14	14	12	12	11	10	10
DEC	17	16	16	16	15-14	13	13	11	11	11	10	10

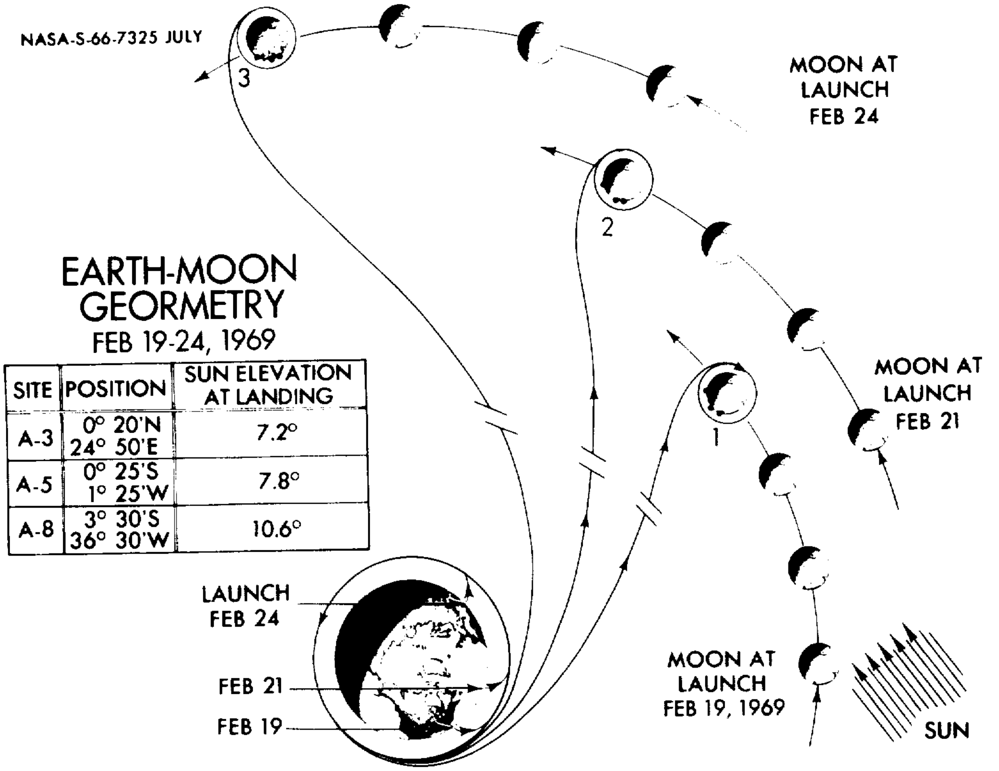
Fig. 10


NASA S 66 6047 MAY
3
MOON
AT
LAUNCH
FEB 6
2
MOON
AT
LAUNCH
FEB 3
1
LAUNCH
FEB 6
FEB 3
FEB 1
MOON
AT
LAUNCH
FEB 1, 1968
SUN

EARTH-MOON GEOMETRY
FEB 1-6, 1968

SITE	POSITION	SUN ELEVATION AT LANDING
A-3	0° 20' N 24° 50' E	9.5°
A-5	0° 25' S 1° 25' W	9.6°
A-8	3° 30' S 36° 30' W	12.1°

Fig. 11


NASA-S-66-7325 JULY
3
MOON AT
LAUNCH
FEB 24
2
MOON AT
LAUNCH
FEB 21
1
LAUNCH
FEB 24
FEB 21
FEB 19
MOON AT
LAUNCH
FEB 19, 1969
SUN

EARTH-MOON GEORMETRY
FEB 19-24, 1969

SITE	POSITION	SUN ELEVATION AT LANDING
A-3	0° 20'N 24° 50'E	7.2°
A-5	0° 25'S 1° 25'W	7.8°
A-8	3° 30'S 36° 30'W	10.6°

Fig.

N ASA-S-66-6833 JUN

SATURN V ATTITUDE LAUNCH CONFIGURATION

THE LAUNCH VEHICLE USES POSITION DESIGNATIONS IN LIEU OF Y AND Z AXES

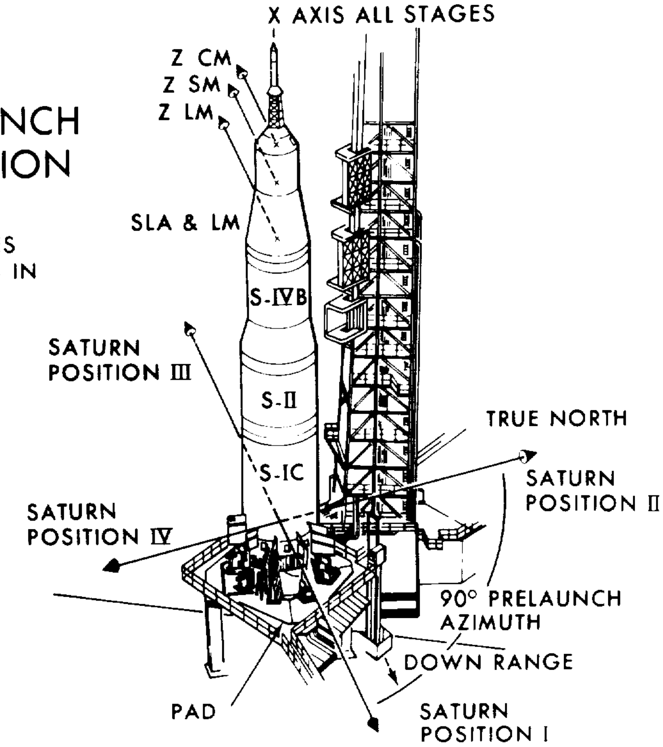

X AXIS ALL STAGES
Z CM
Z SM
Z LM
SLA & LM
S-IVB
SATURN POSITION III
S-II
S-IC
TRUE NORTH
SATURN POSITION II
SATURN POSITION IV
90° PRELAUNCH AZIMUTH
DOWN RANGE
PAD
SATURN POSITION I

Fig. 13

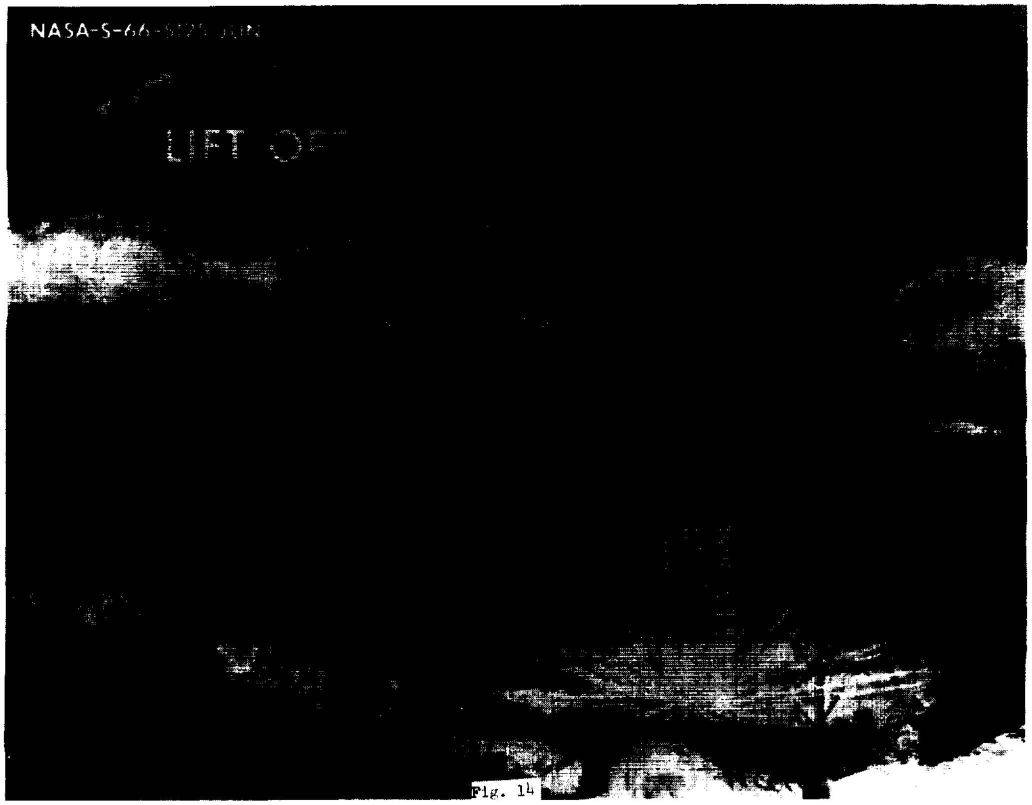

NASA-S-66-[illegible] JUN
LIFT OFF

Fig. 14

NASA-S-66-6024 MAY

ALTITUDE VS RANGE

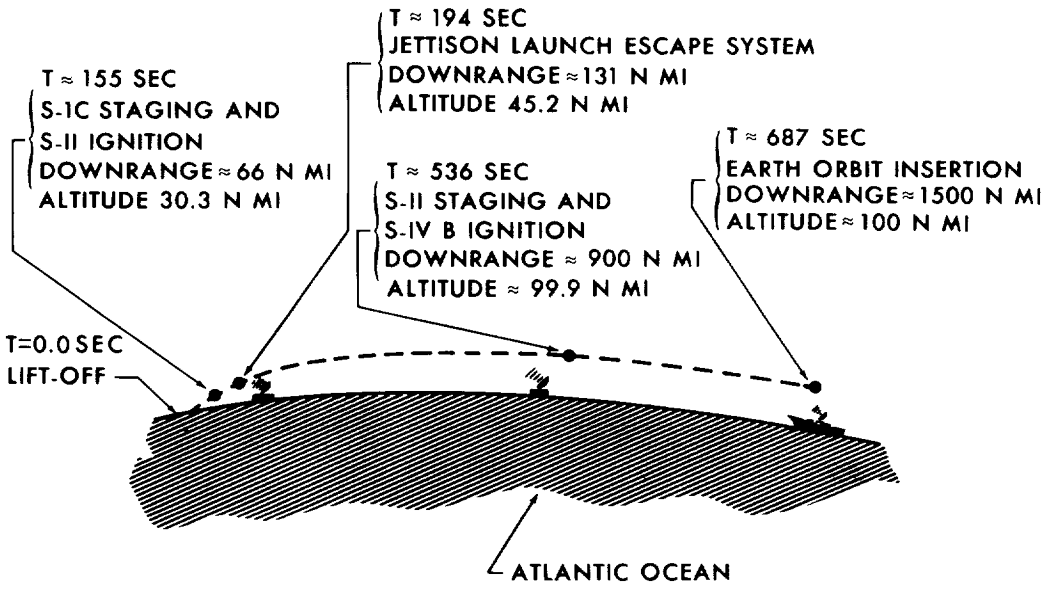

T ≈ 194 SEC
JETTISON LAUNCH ESCAPE SYSTEM
DOWNRANGE ≈ 131 N MI
ALTITUDE 45.2 N MI
T ≈ 155 SEC
S-1C STAGING AND
S-II IGNITION
DOWNRANGE ≈ 66 N MI
ALTITUDE 30.3 N MI
T ≈ 687 SEC
EARTH ORBIT INSERTION
DOWNRANGE ≈ 1500 N MI
ALTITUDE ≈ 100 N MI
T ≈ 536 SEC
S-II STAGING AND
S-IV B IGNITION
DOWNRANGE ≈ 900 N MI
ALTITUDE ≈ 99.9 N MI
T=0.0 SEC
LIFT-OFF
ATLANTIC OCEAN

Fig. 15

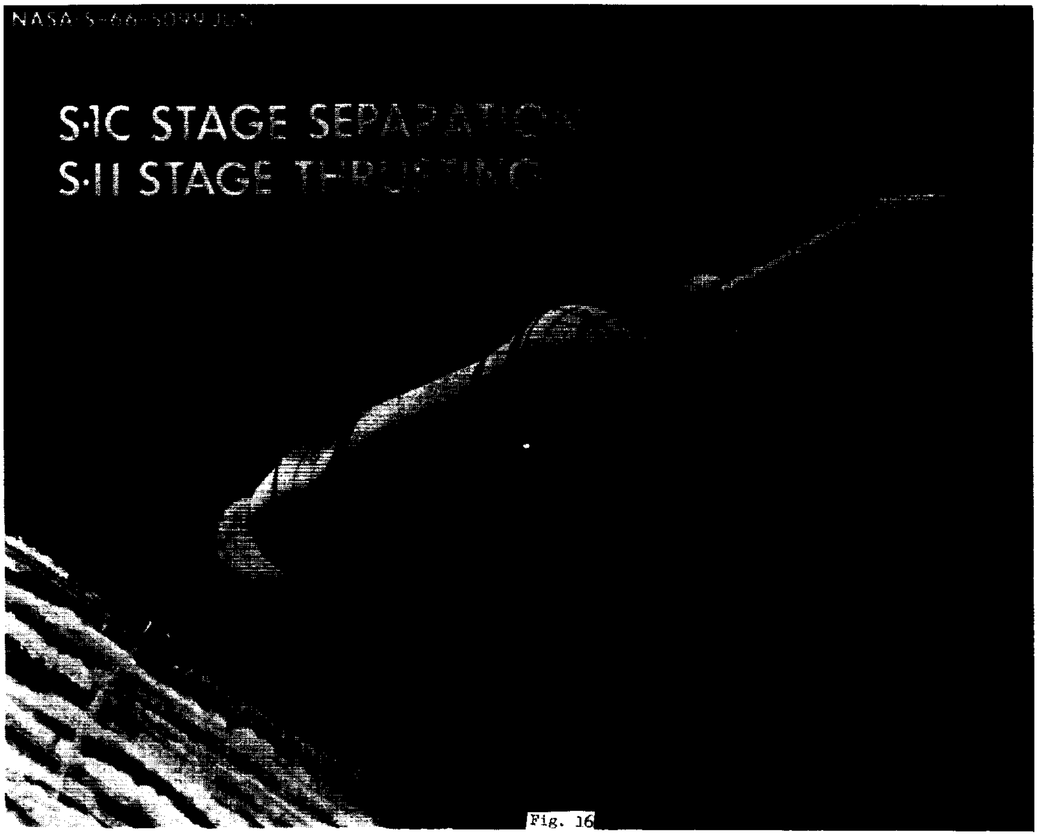
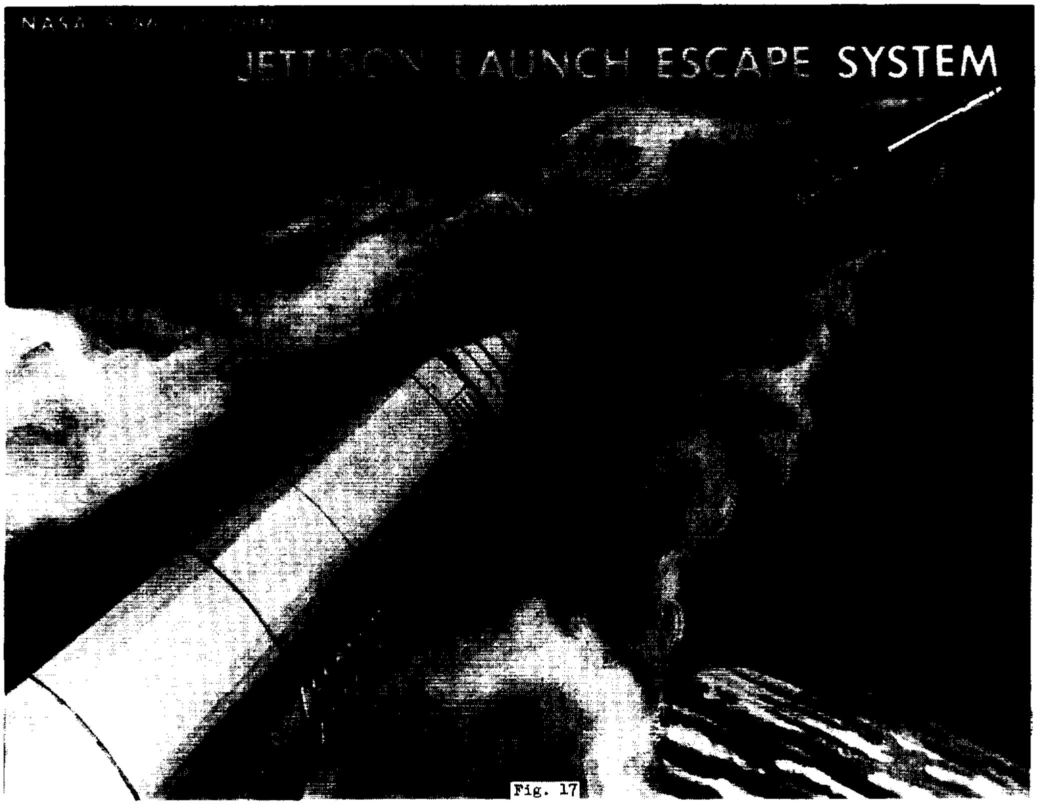

NASA S-66-5099 JUN
S-1C STAGE SEPARATION
S-II STAGE THRUSTING

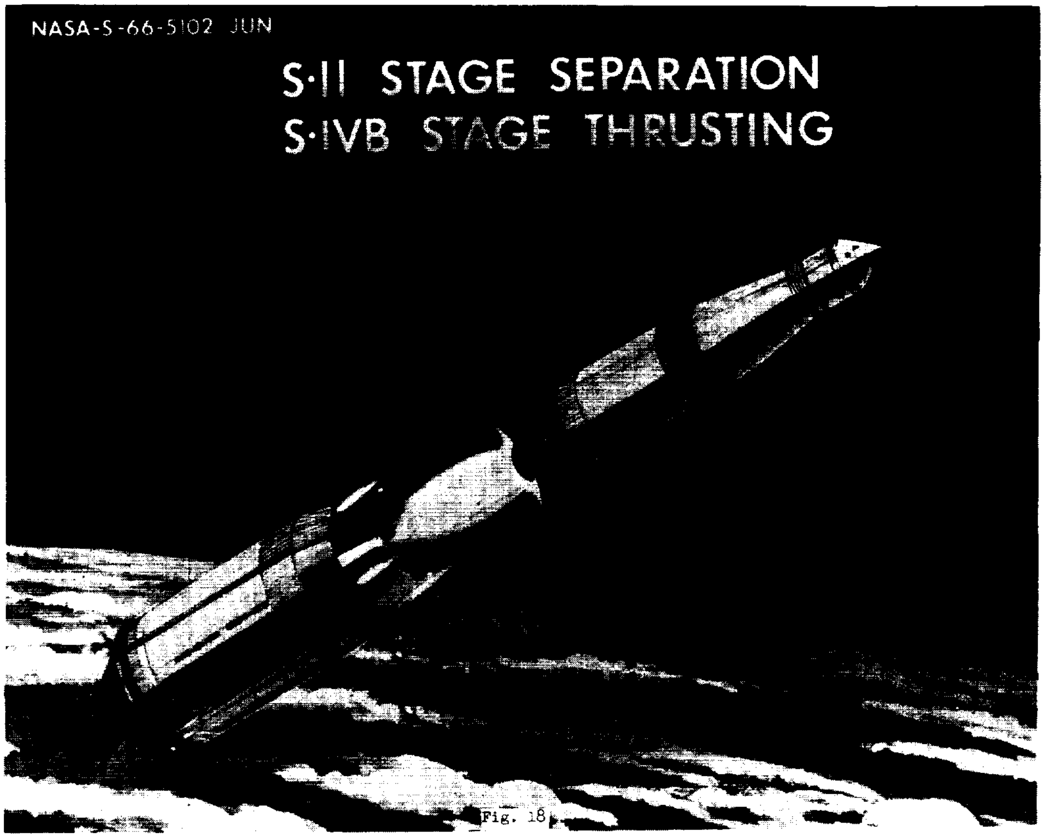
Fig. 16



NASA [illegible] JUN

JETTISON LAUNCH ESCAPE SYSTEM

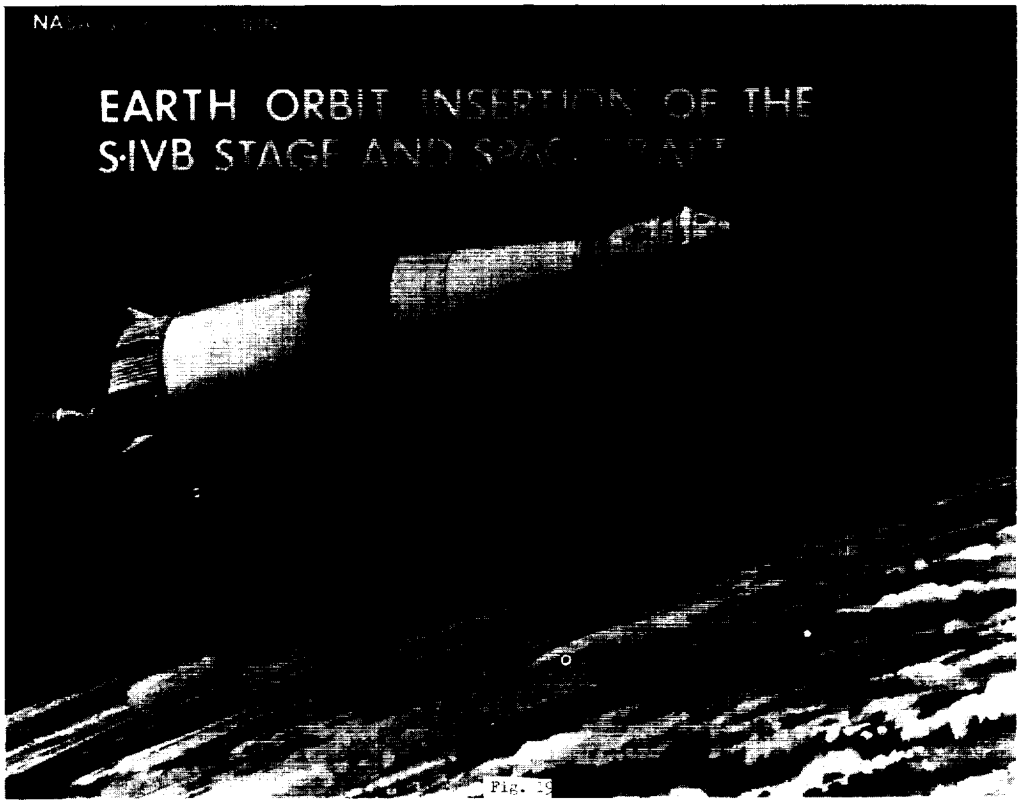
Fig. 17



NASA-S-66-5102 JUN

S·II STAGE SEPARATION
S·IVB STAGE THRUSTING

Fig. 18



EARTH ORBIT INSERTION OF THE S·IVB STAGE AND SPACECRAFT

Fig. 19

ORBIT INSERTION

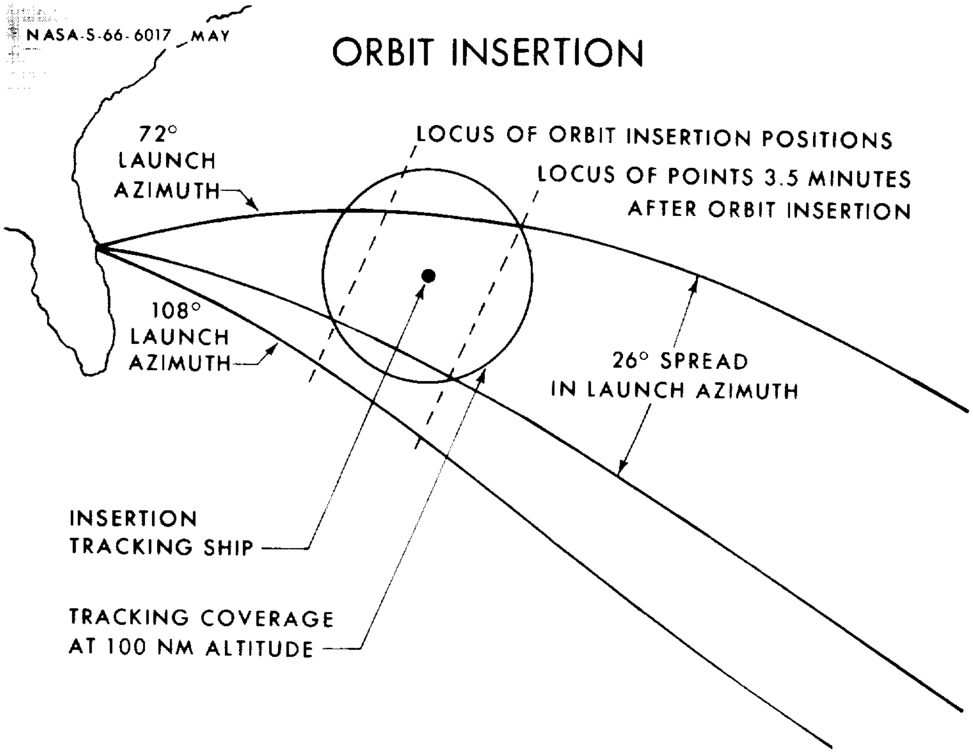

NASA-S-66-6017 MAY
72° LAUNCH AZIMUTH
108° LAUNCH AZIMUTH
LOCUS OF ORBIT INSERTION POSITIONS
LOCUS OF POINTS 3.5 MINUTES AFTER ORBIT INSERTION
26° SPREAD IN LAUNCH AZIMUTH
INSERTION TRACKING SHIP
TRACKING COVERAGE AT 100 NM ALTITUDE

Fig. [illegible]

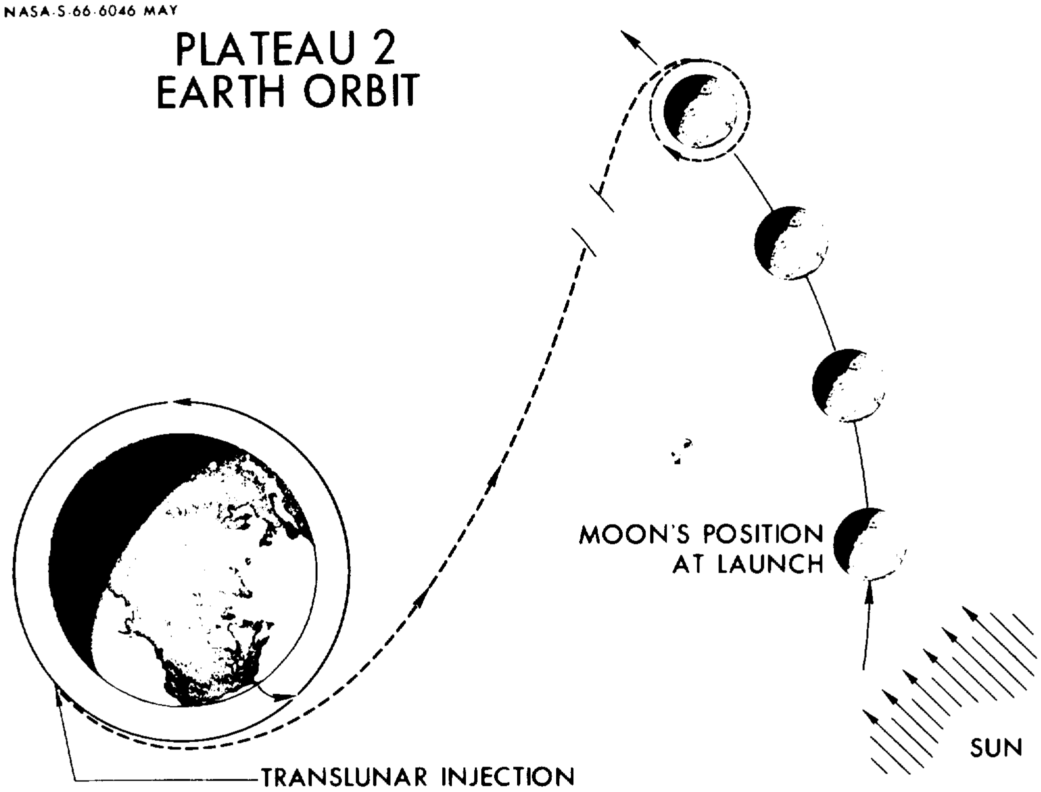

NASA-S-66-6046 MAY
PLATEAU 2
EARTH ORBIT
MOON'S POSITION AT LAUNCH
SUN
TRANSLUNAR INJECTION

Fig. 21

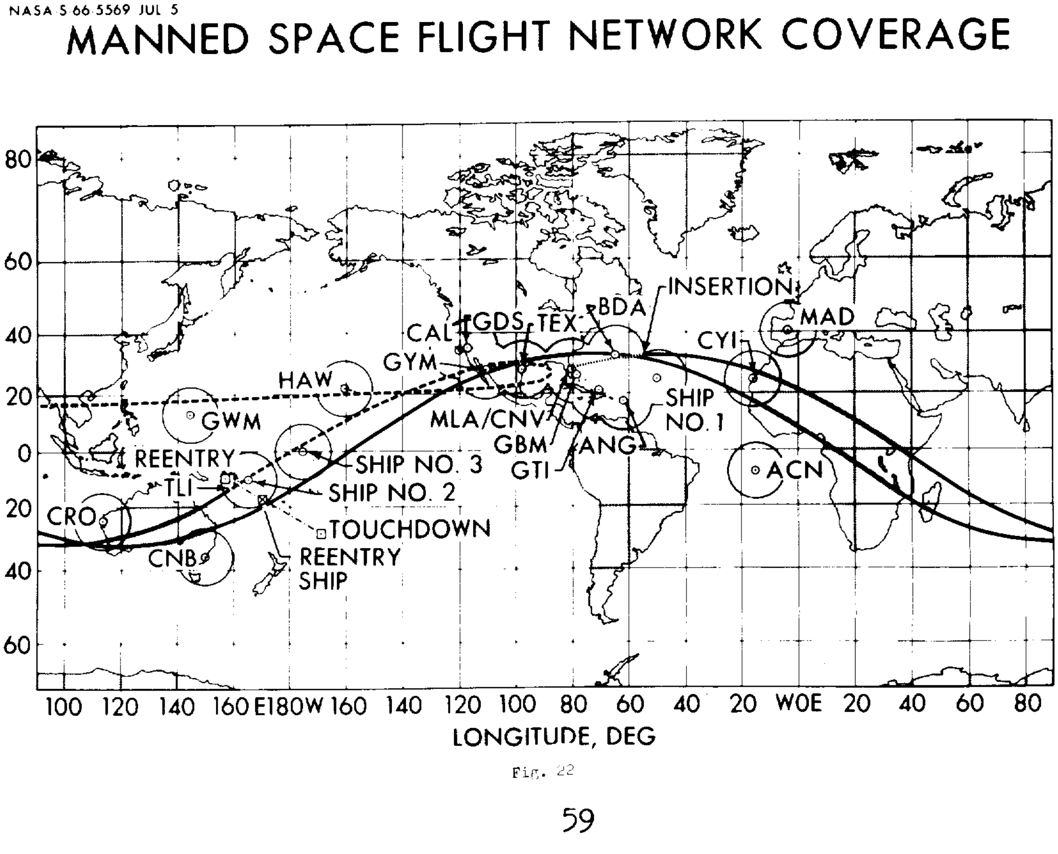

NASA S-66-5569 JUL 5
MANNED SPACE FLIGHT NETWORK COVERAGE
80
60
40
20
0
20
40
60
INSERTION
BDA
CAL
GDS
TEX
MAD
GYM
CYI
HAW
SHIP NO. 1
GWM
MLA/CNV
GBM
ANG
REENTRY
TLI
SHIP NO. 3
SHIP NO. 2
GTI
ACN
CRO
TOUCHDOWN
CNB
REENTRY SHIP
100 120 140 160 E180W 160 140 120 100 80 60 40 20 W0E 20 40 60 80
LONGITUDE, DEG

Fig. 22

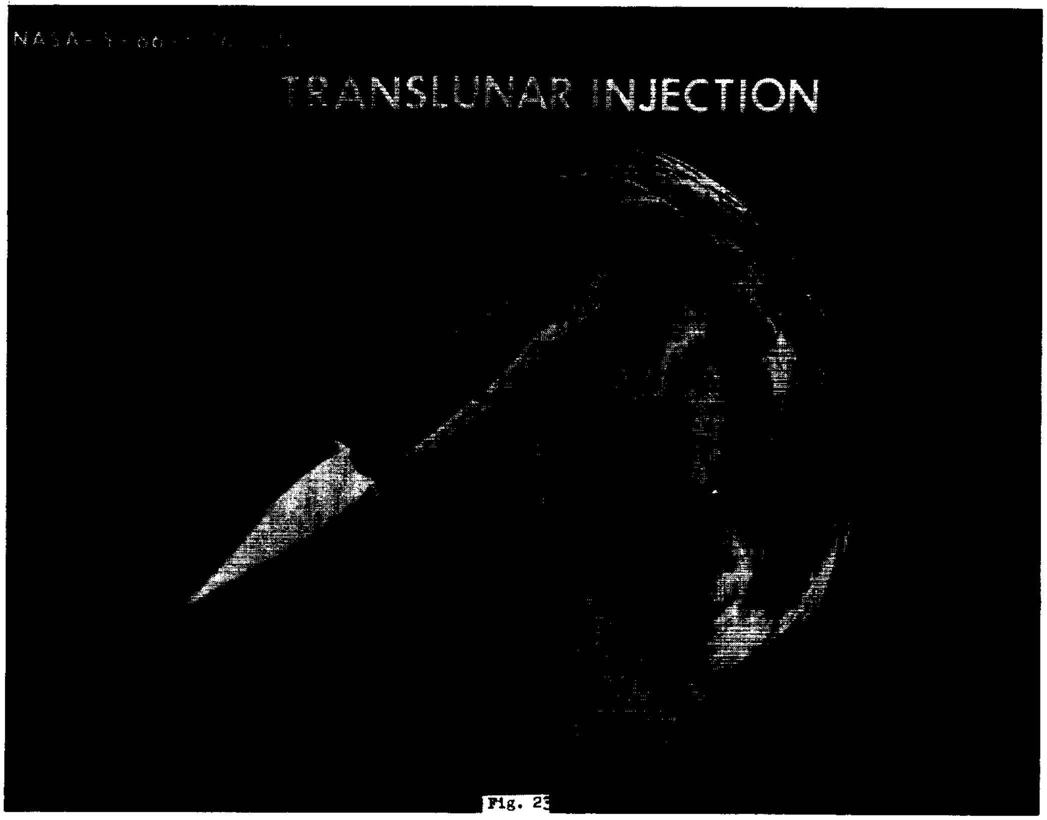

NASA-S-66-
TRANSLUNAR INJECTION

Fig. 23

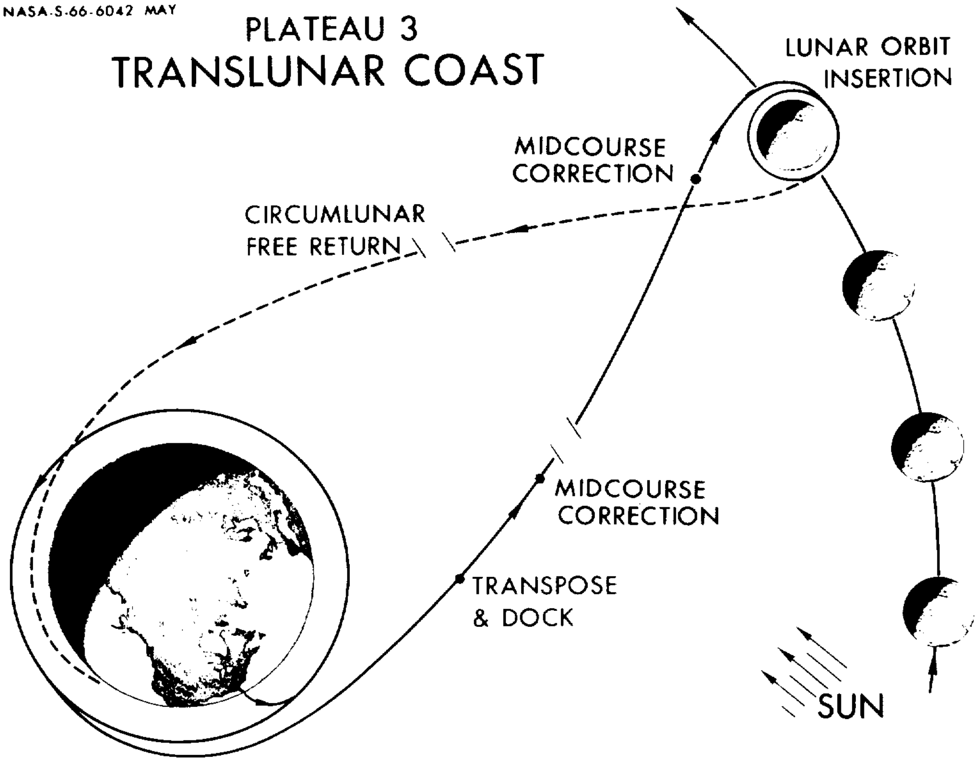

NASA-S-66-6042 MAY
PLATEAU 3
TRANSLUNAR COAST
LUNAR ORBIT INSERTION
MIDCOURSE CORRECTION
CIRCUMLUNAR FREE RETURN
MIDCOURSE CORRECTION
TRANSPOSE & DOCK
SUN

Fig. 24

NASA-S-66-6825 JUN

TRANSLUNAR TRANSPOSITION AND DOCKING

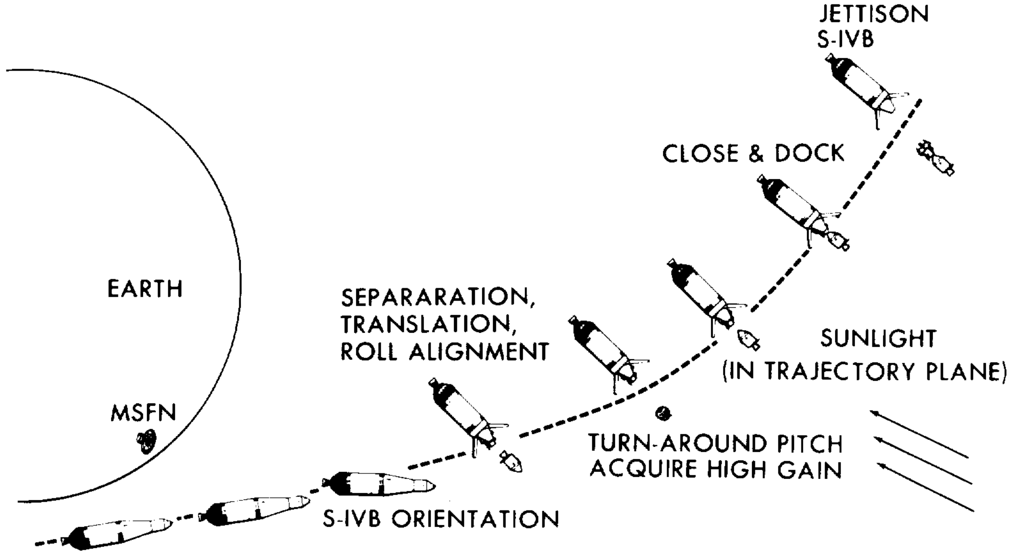

JETTISON
S-IVB
CLOSE & DOCK
EARTH
SEPARARATION,
TRANSLATION,
ROLL ALIGNMENT
SUNLIGHT
(IN TRAJECTORY PLANE)
MSFN
TURN-AROUND PITCH
ACQUIRE HIGH GAIN
S-IVB ORIENTATION

Fig. 25

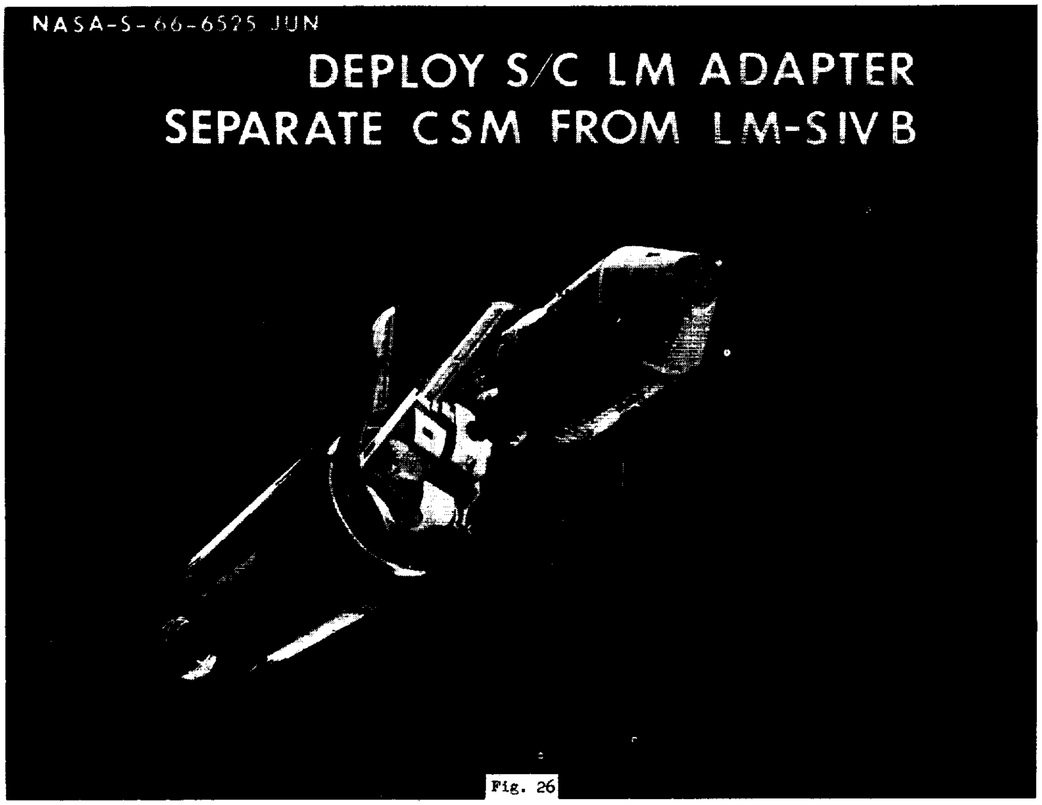
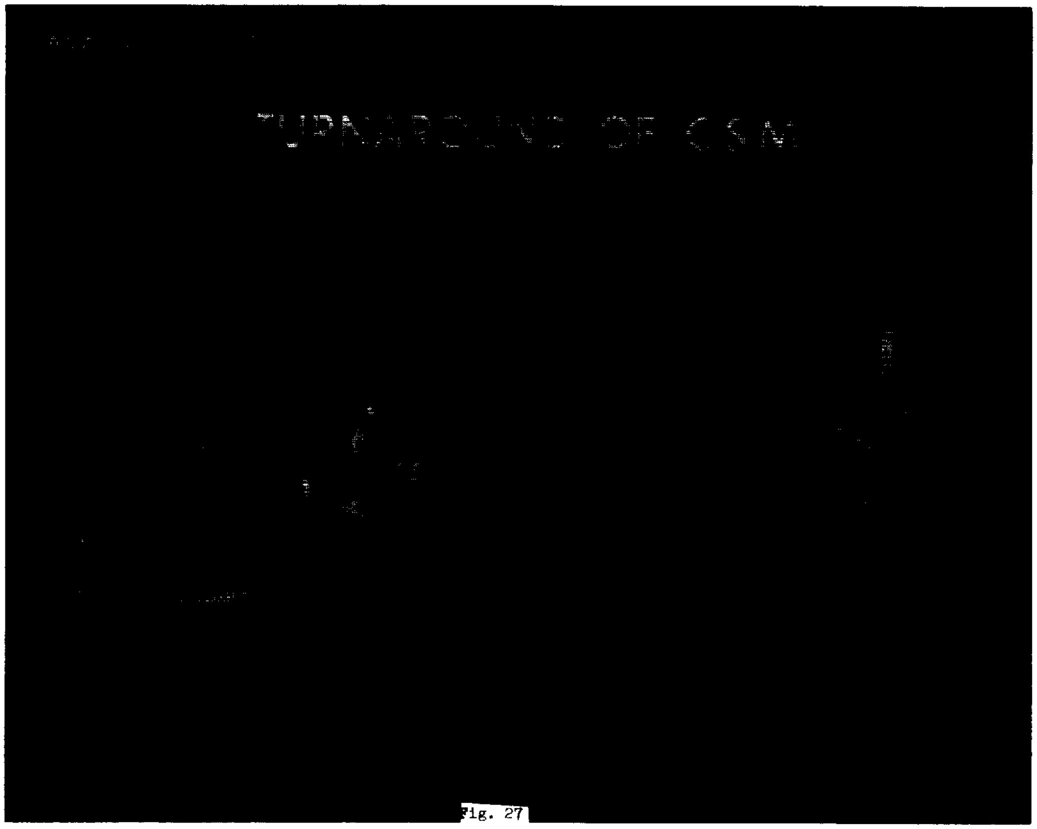
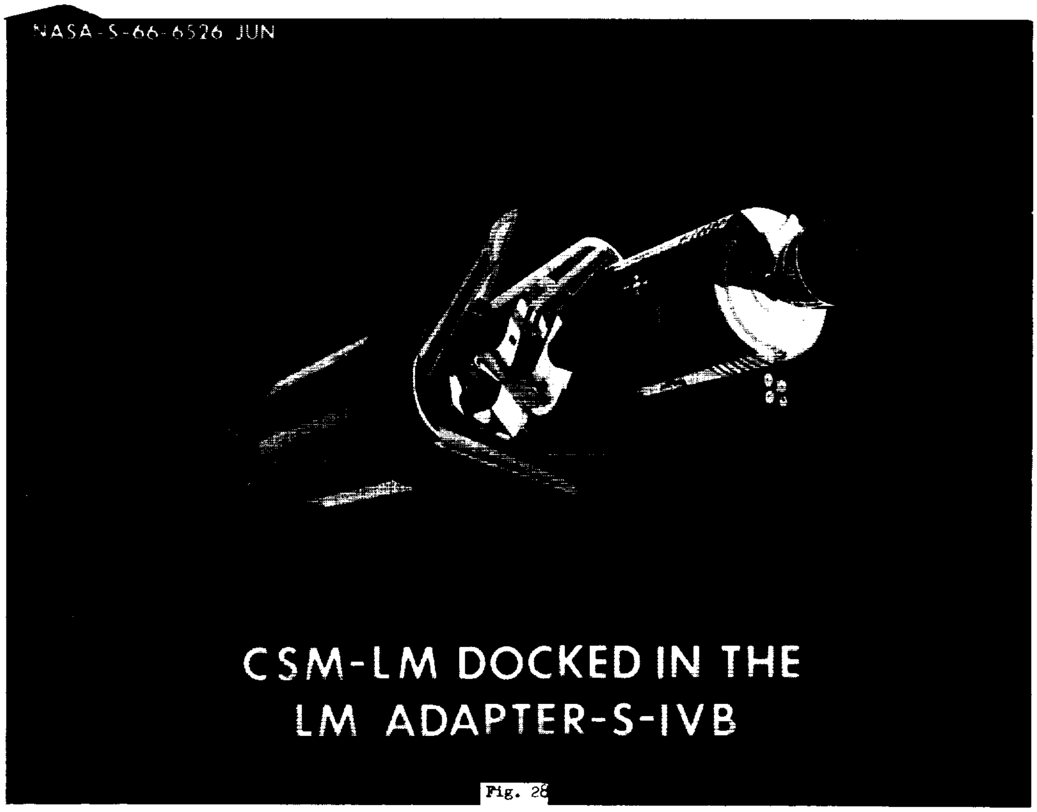

NASA-S-66-6525 JUN
DEPLOY S/C LM ADAPTER
SEPARATE CSM FROM LM-SIVB

Fig. 26



TURNAROUND OF CSM

Fig. 27



NASA-S-66-6526 JUN

CSM-LM DOCKED IN THE
LM ADAPTER-S-IVB

Fig. 28

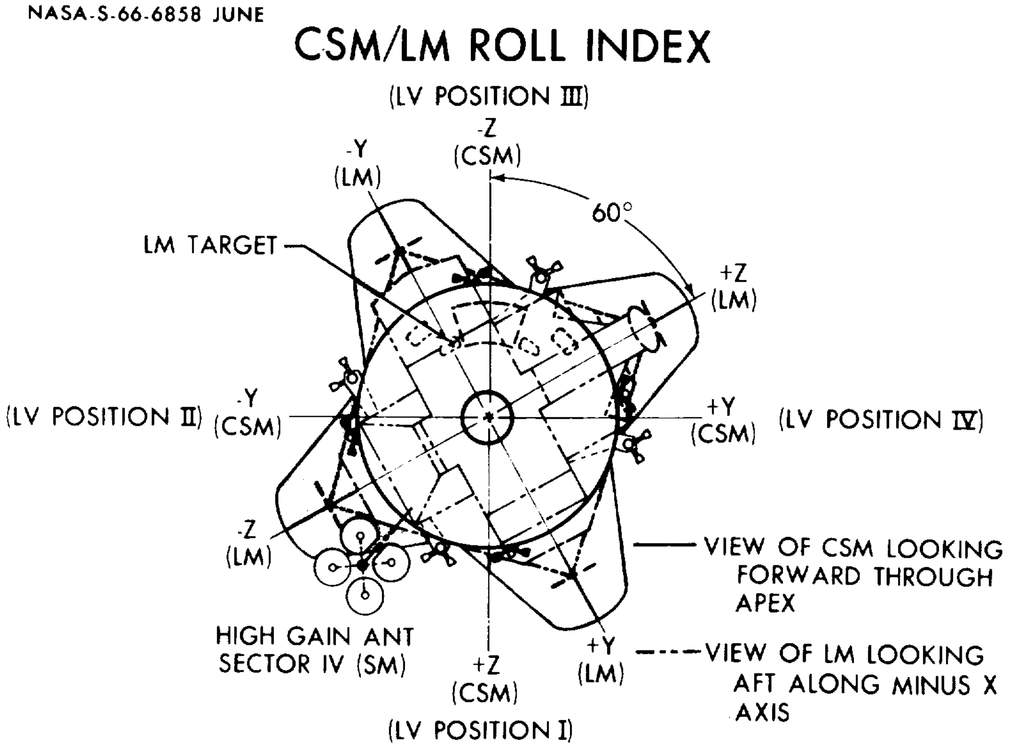

NASA-S-66-6858 JUNE
CSM/LM ROLL INDEX
(LV POSITION III)
-Z
(CSM)
-Y
(LM)
60°
LM TARGET
+Z
(LM)
(LV POSITION II)
-Y
(CSM)
+Y
(CSM)
(LV POSITION IV)
-Z
(LM)
VIEW OF CSM LOOKING FORWARD THROUGH APEX
HIGH GAIN ANT SECTOR IV (SM)
+Z
(CSM)
+Y
(LM)
VIEW OF LM LOOKING AFT ALONG MINUS X AXIS
(LV POSITION I)

Fig. 29

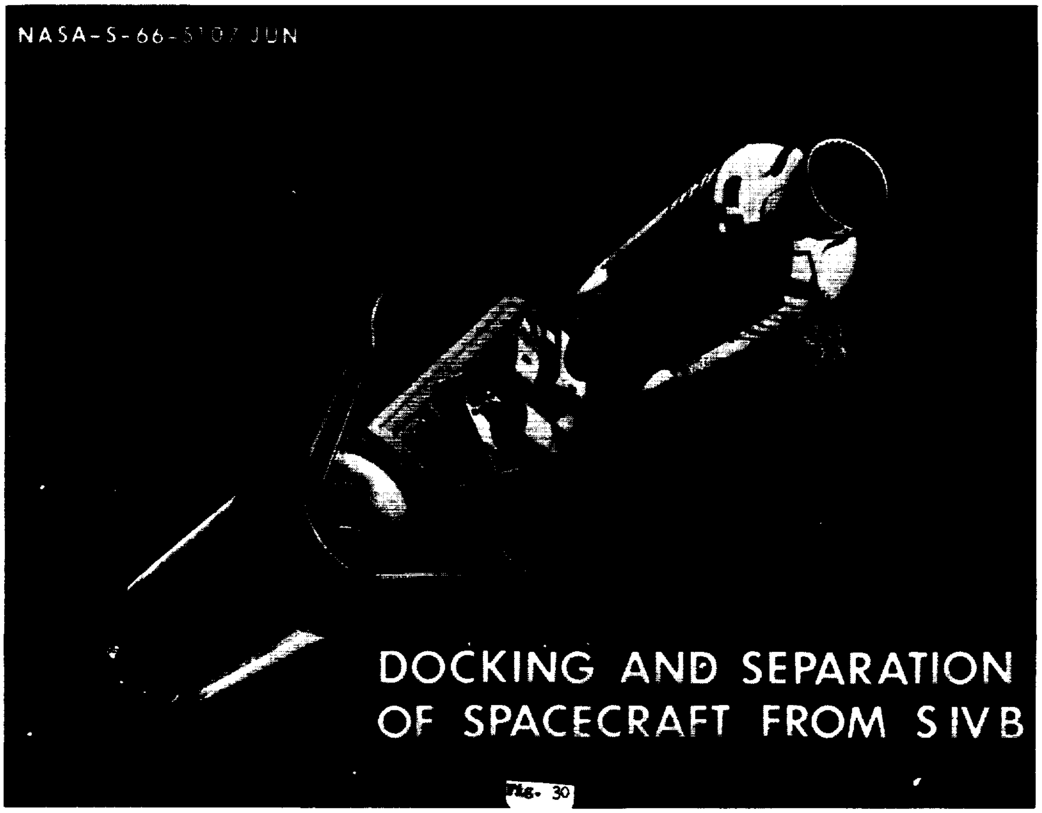

NASA-S-66-5107 JUN
DOCKING AND SEPARATION OF SPACECRAFT FROM SIVB

Fig. 30

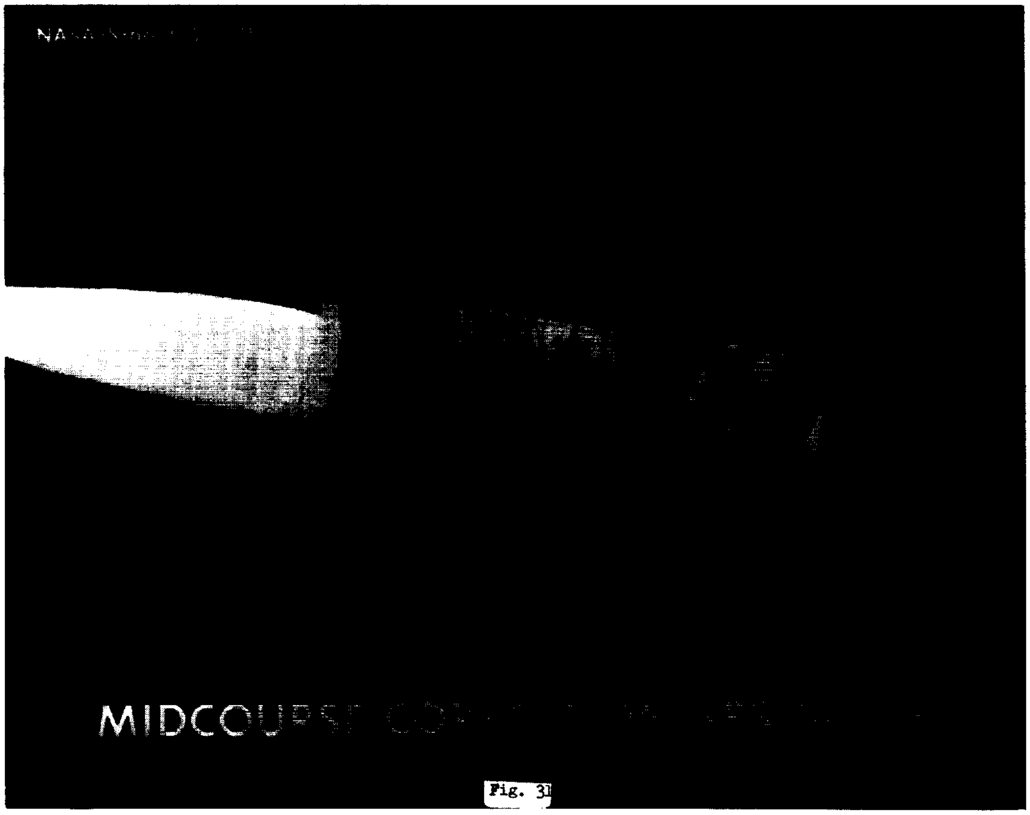


Fig. 31

NASA-S-66-6389 JUN

THERMAL CYCLING

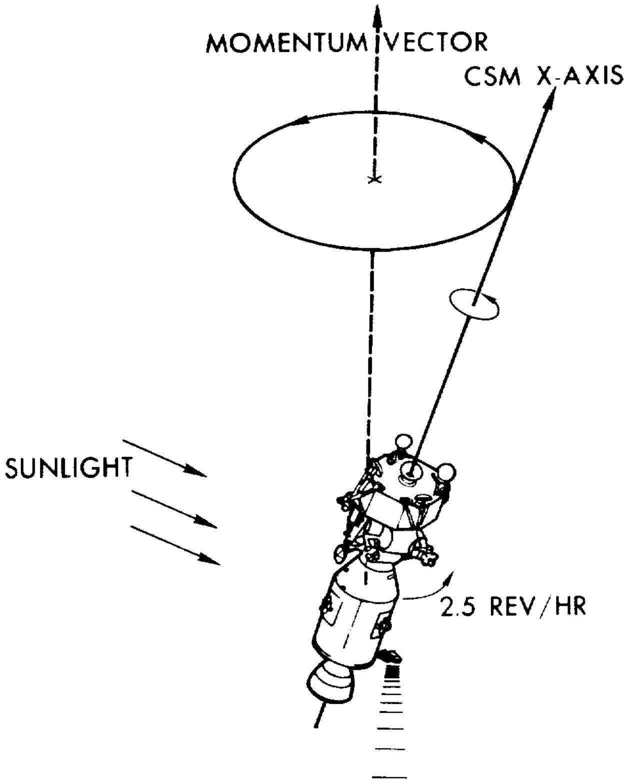
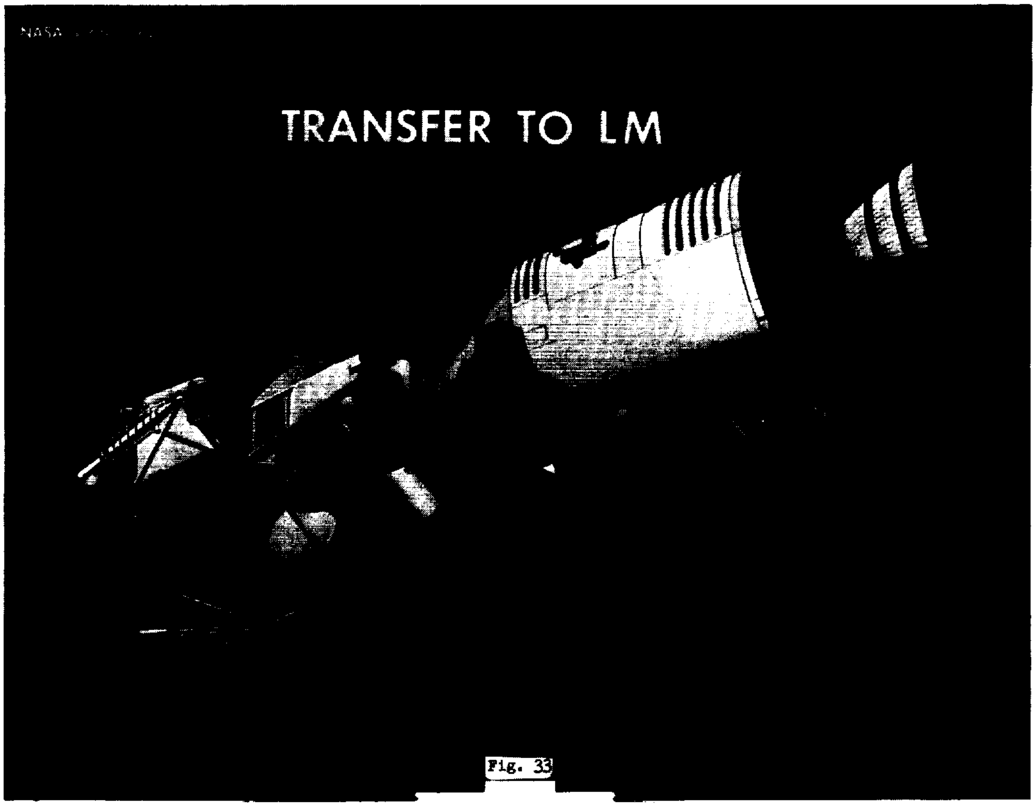
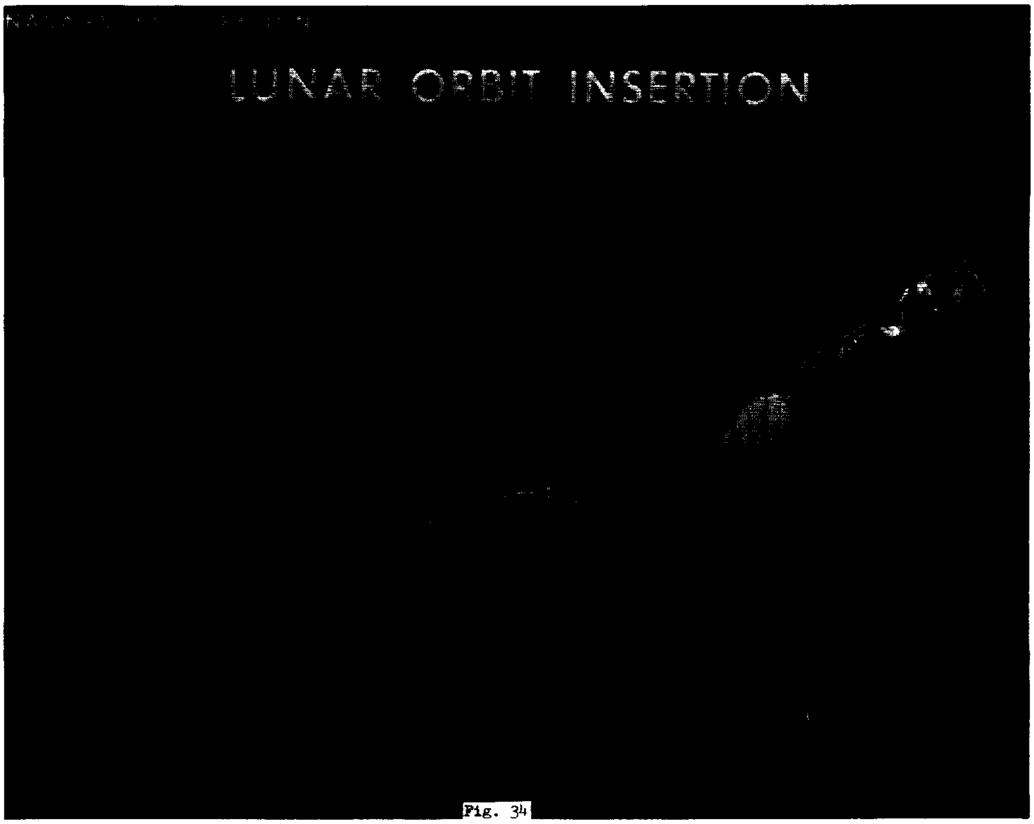

MOMENTUM VECTOR
CSM X-AXIS
SUNLIGHT
2.5 REV/HR

Fig. 3



TRANSFER TO LM

Fig. 33



LUNAR ORBIT INSERTION

Fig. 34

NASA S-66-6030 MAY

PLATEAU 4
LUNAR ORBIT PRIOR TO LM DESCENT

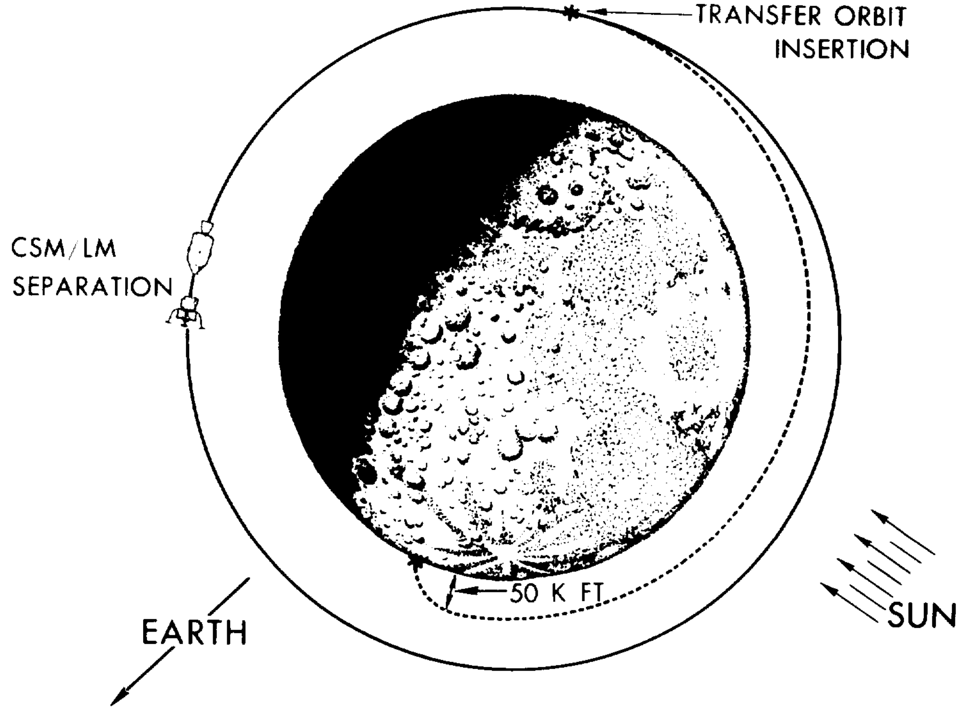

TRANSFER ORBIT INSERTION
CSM/LM SEPARATION
50 K FT
EARTH
SUN

Fig. 35

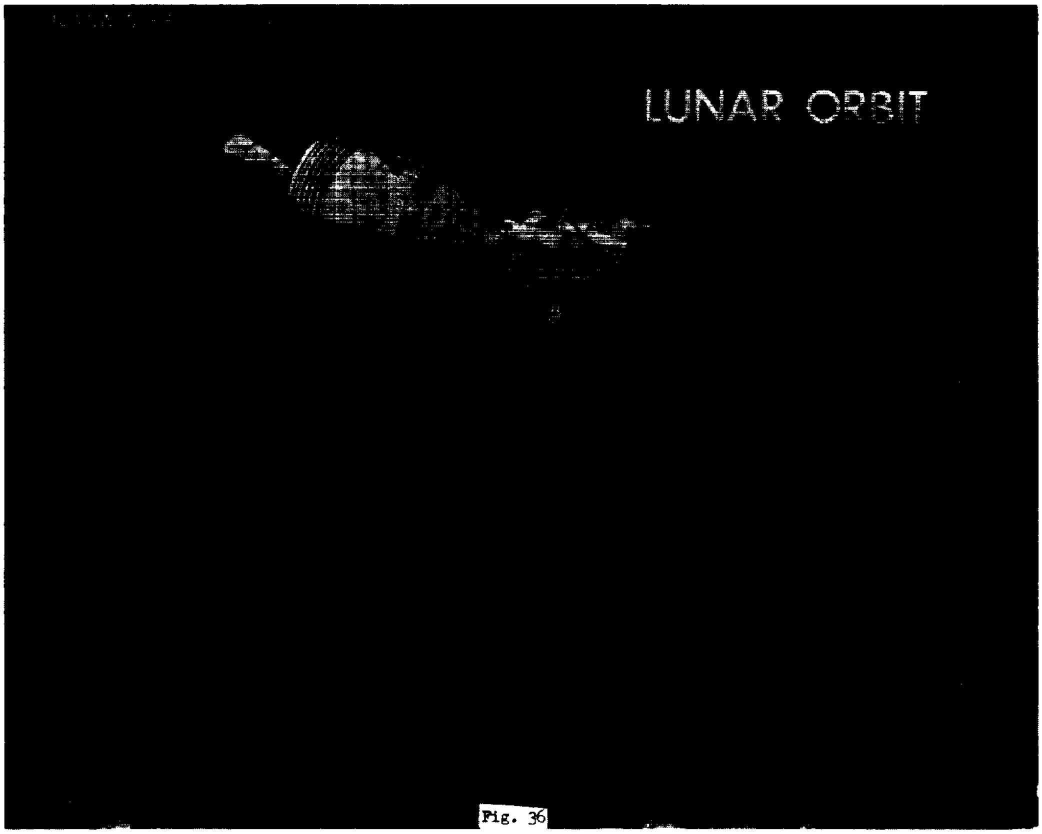

LUNAR ORBIT

Fig. 36

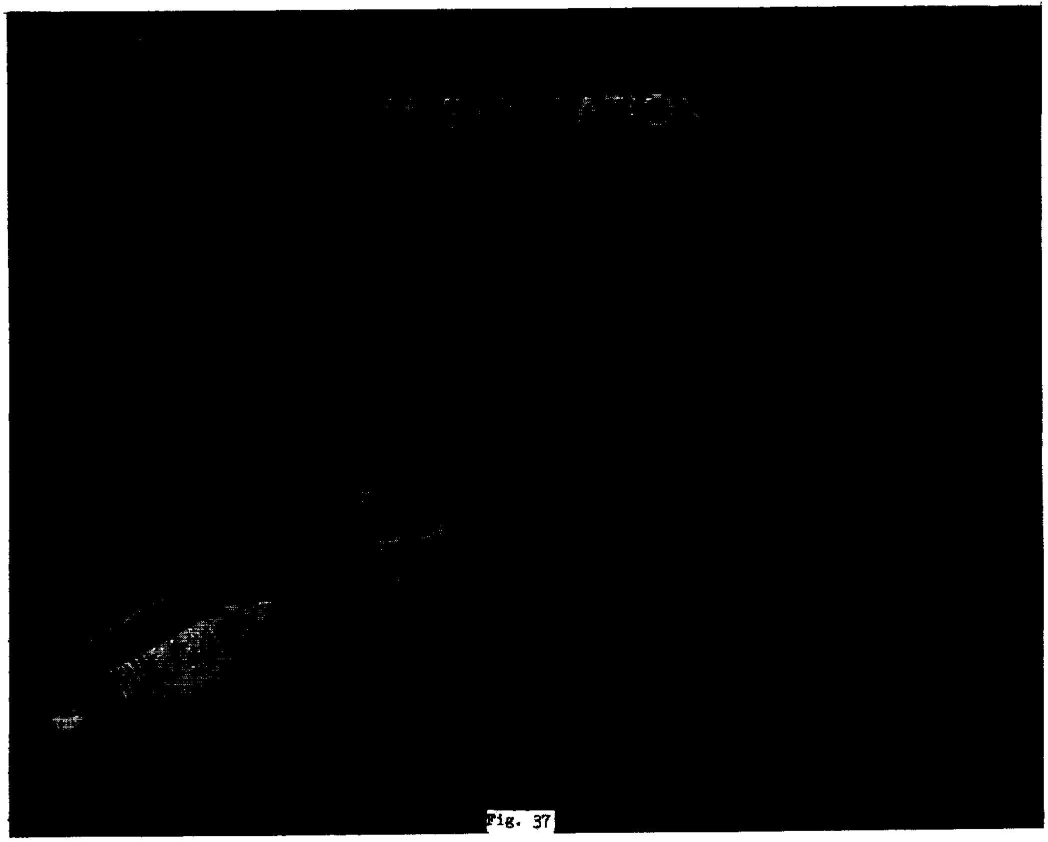


Fig. 37

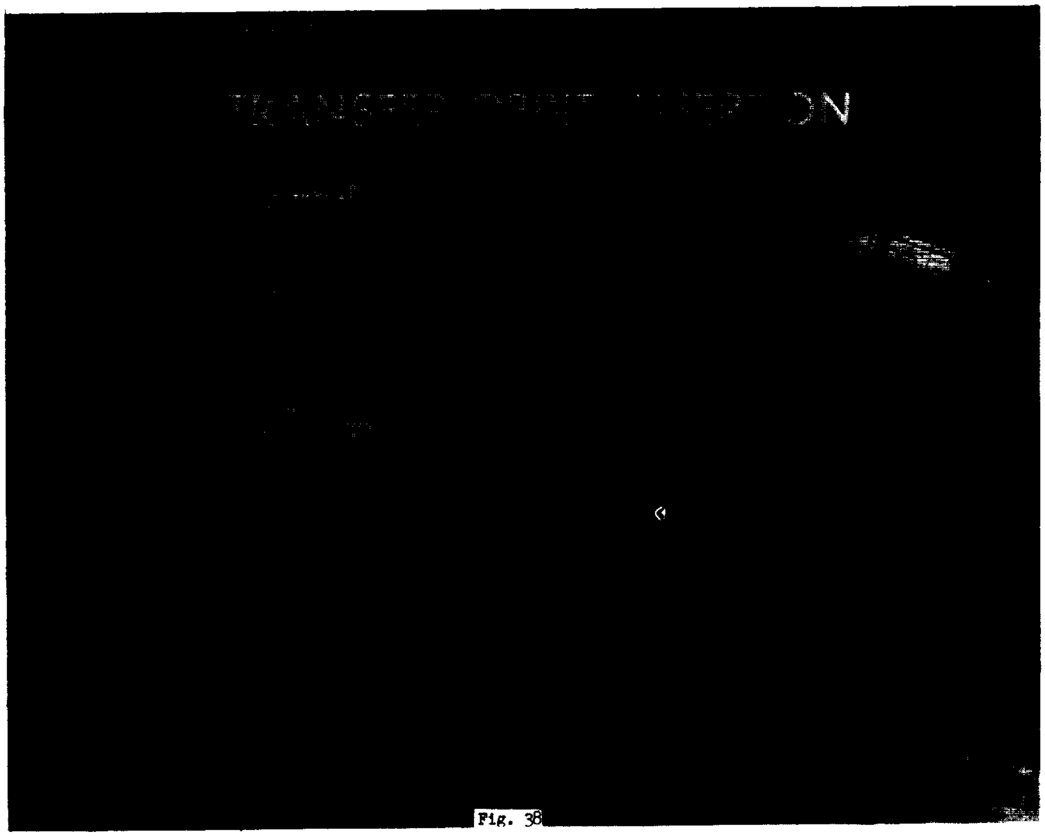

TRANSFER ORBIT INSERTION

Fig. 38

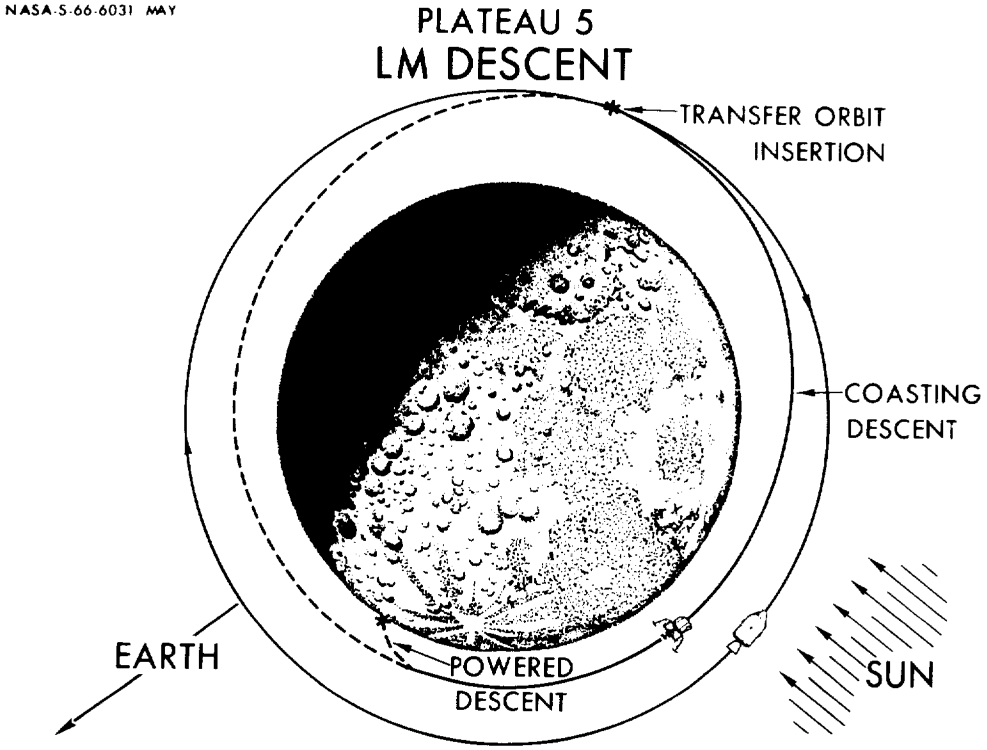

NASA-S-66-6031 MAY
PLATEAU 5
LM DESCENT
TRANSFER ORBIT INSERTION
COASTING DESCENT
EARTH
POWERED DESCENT
SUN

Fig. 39

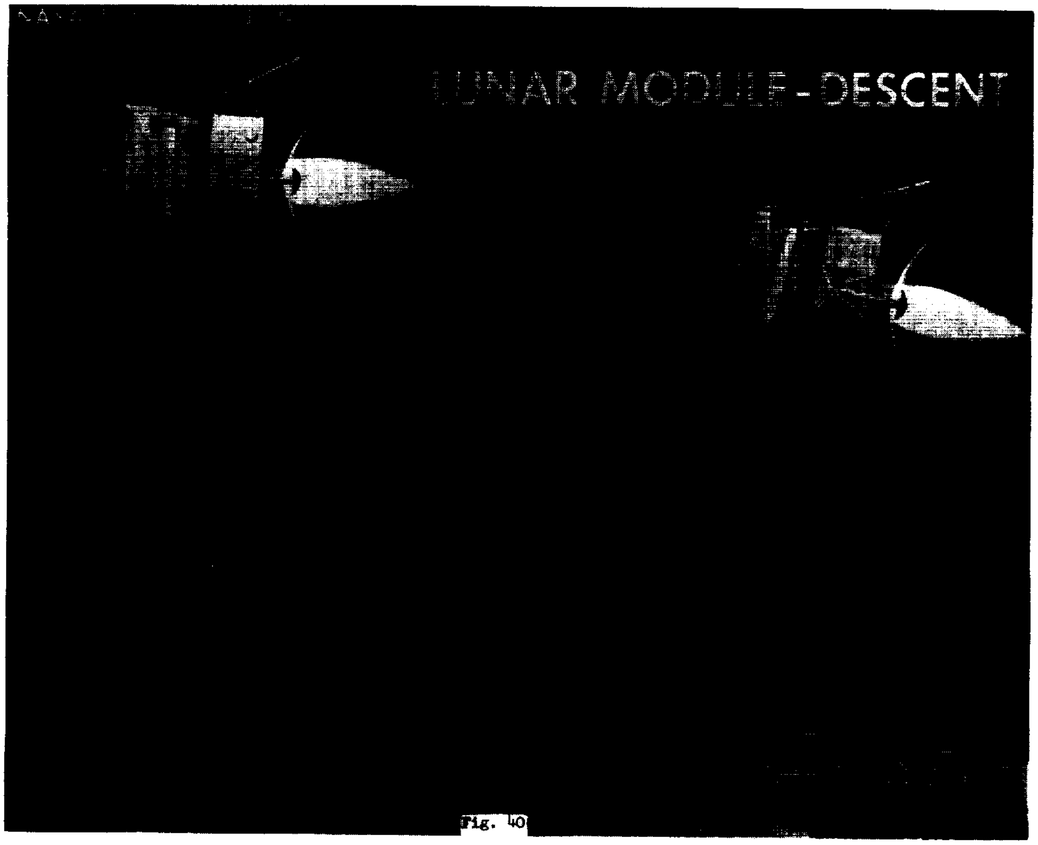

LUNAR MODULE-DESCENT

Fig. 40

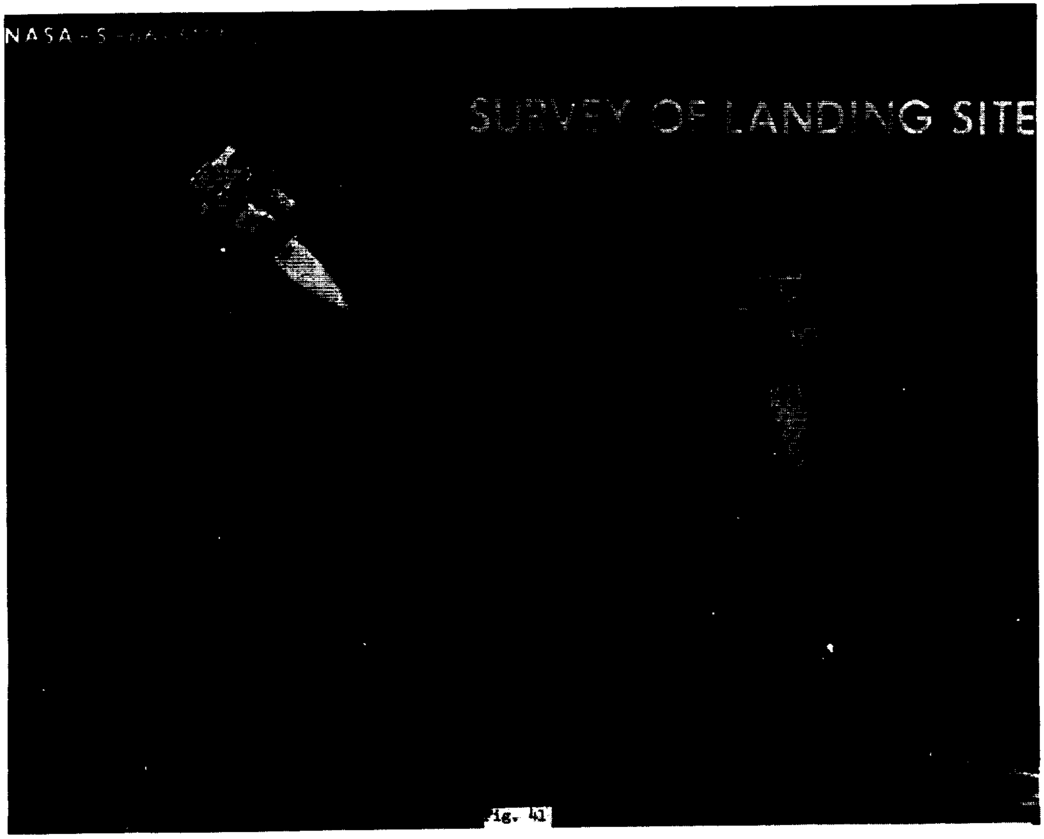

NASA-S-66-
SURVEY OF LANDING SITE

Fig. 41

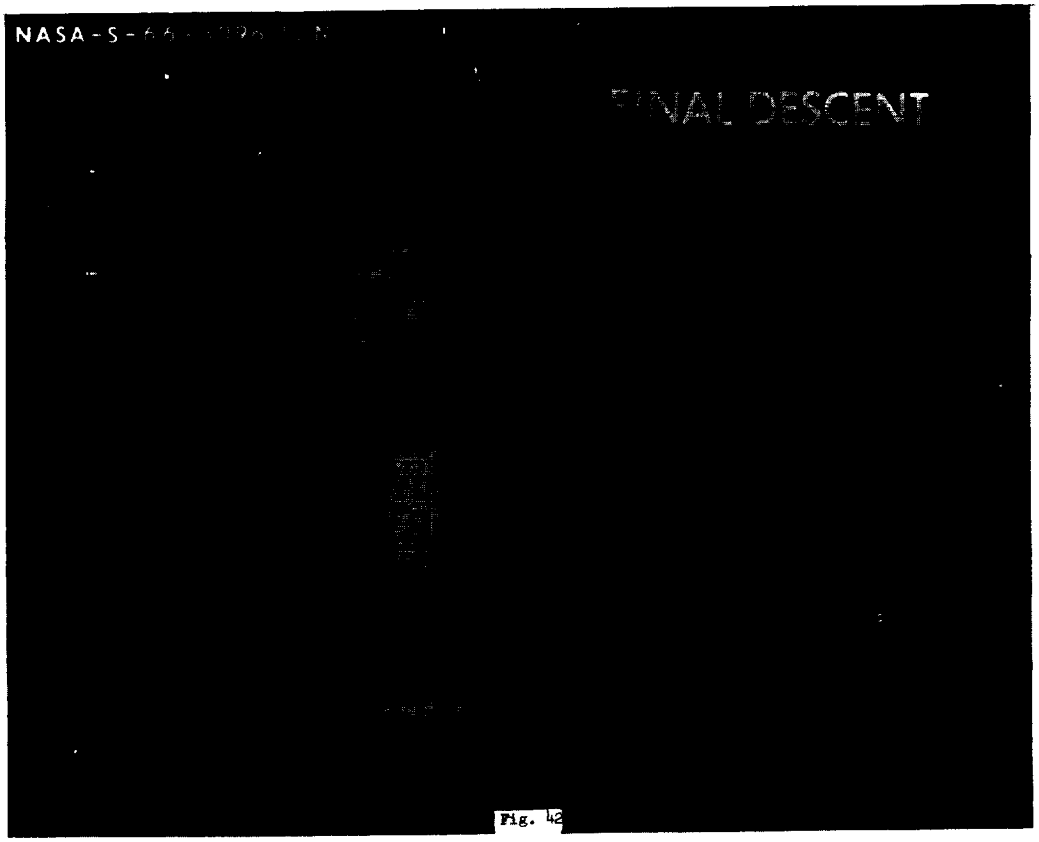
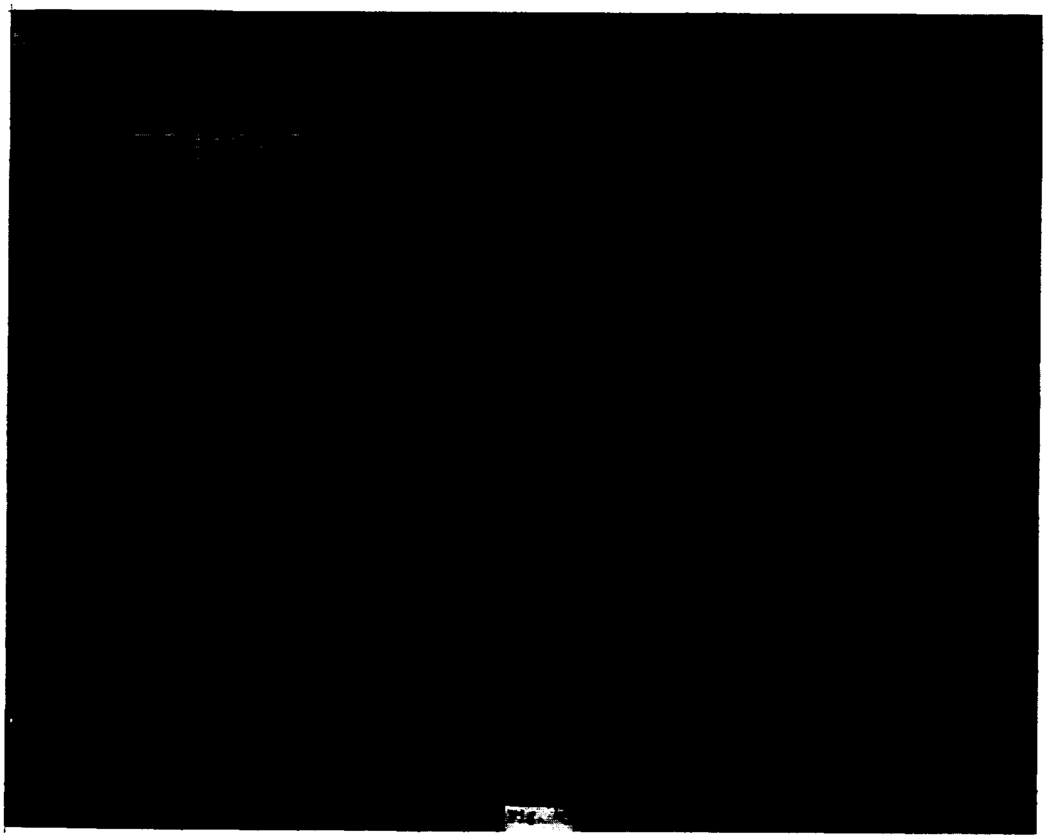

NASA-S-66-
FINAL DESCENT

Fig. 42



NASA-S-66-6029 MAY

PLATEAU 6
LUNAR SURFACE STAY

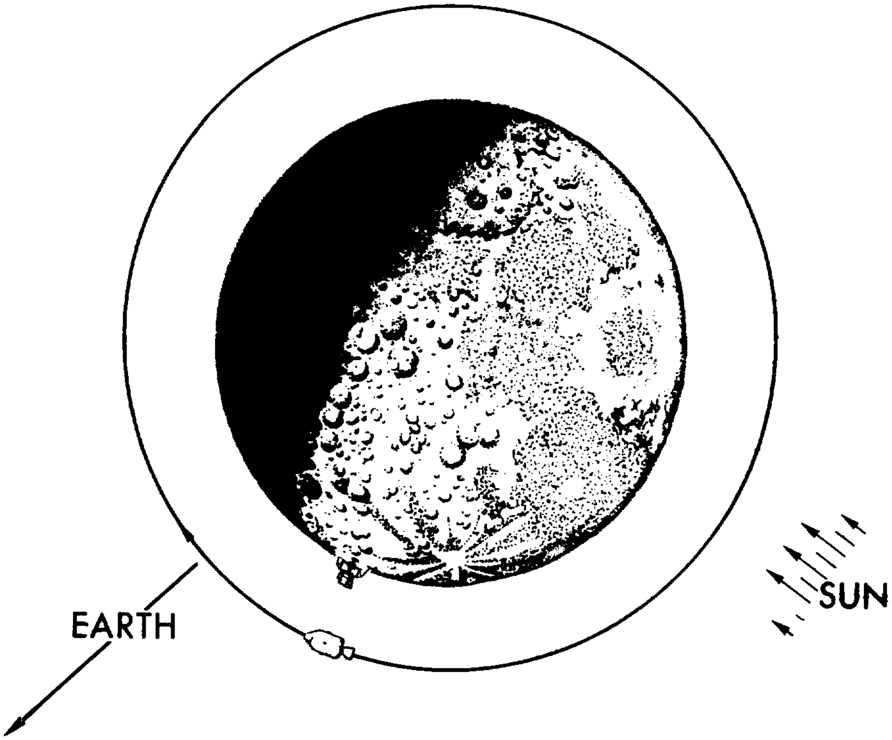


Fig. 44

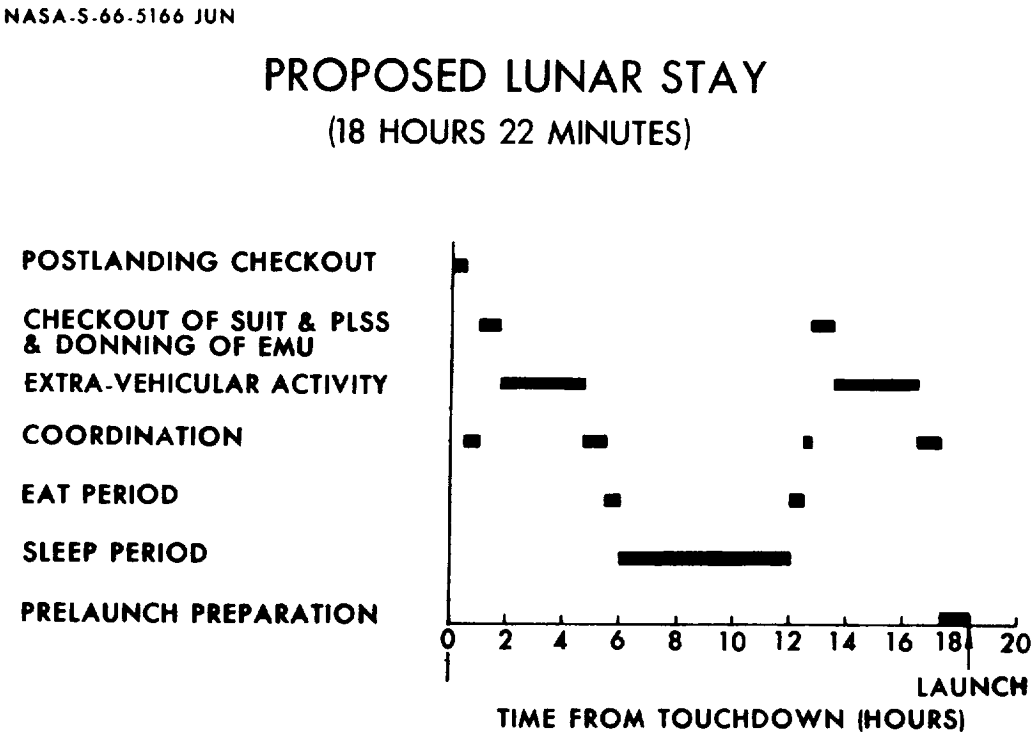

NASA-S-66-5166 JUN
PROPOSED LUNAR STAY
(18 HOURS 22 MINUTES)
POSTLANDING CHECKOUT
CHECKOUT OF SUIT & PLSS & DONNING OF EMU
EXTRA-VEHICULAR ACTIVITY
COORDINATION
EAT PERIOD
SLEEP PERIOD
PRELAUNCH PREPARATION
0
2
4
6
8
10
12
14
16
18
20
LAUNCH
TIME FROM TOUCHDOWN (HOURS)

Fig. 45

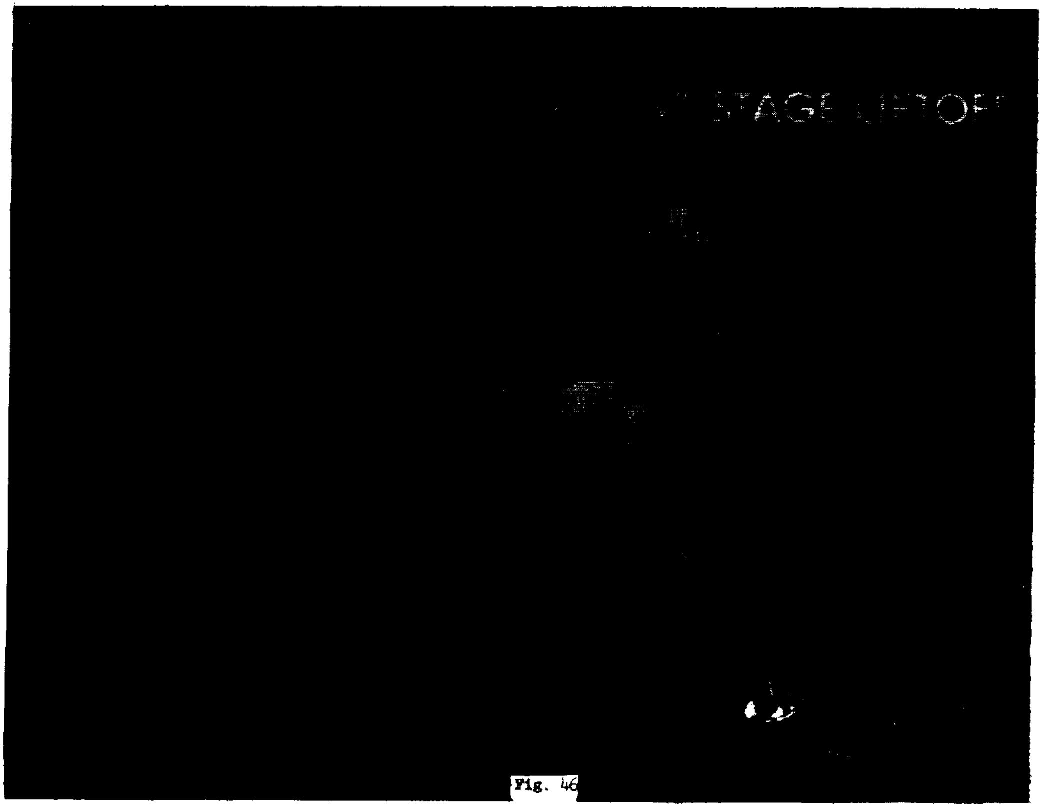

STAGE LIFTOFF

Fig. 46

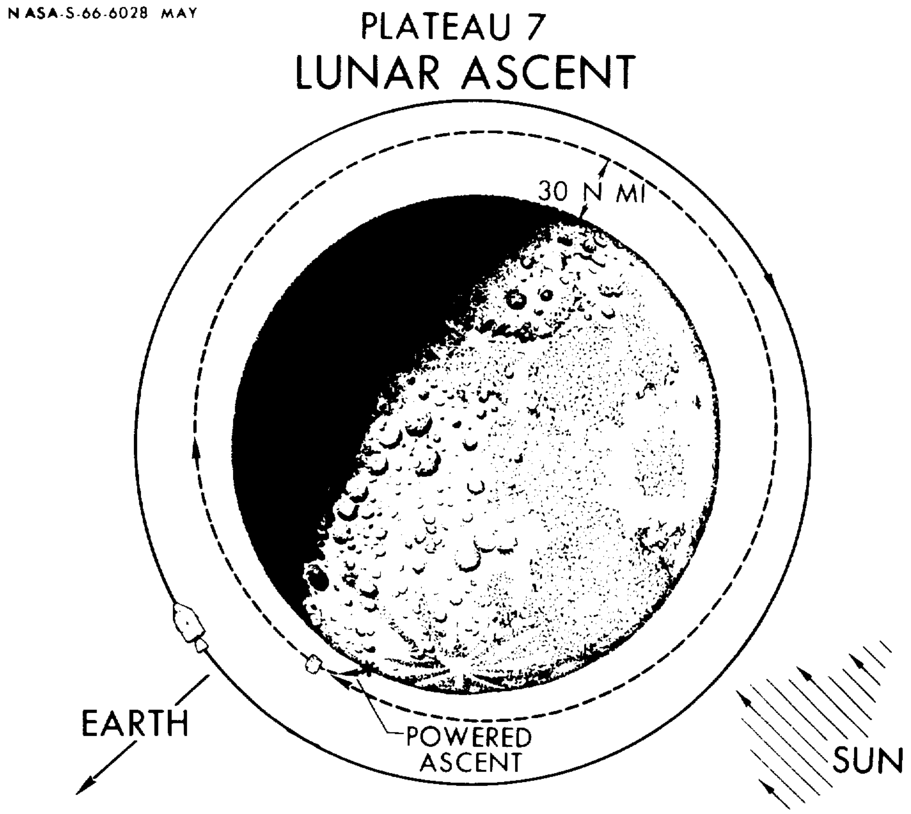

NASA-S-66-6028 MAY
PLATEAU 7
LUNAR ASCENT
30 N MI
EARTH
POWERED
ASCENT
SUN

Fig. 47

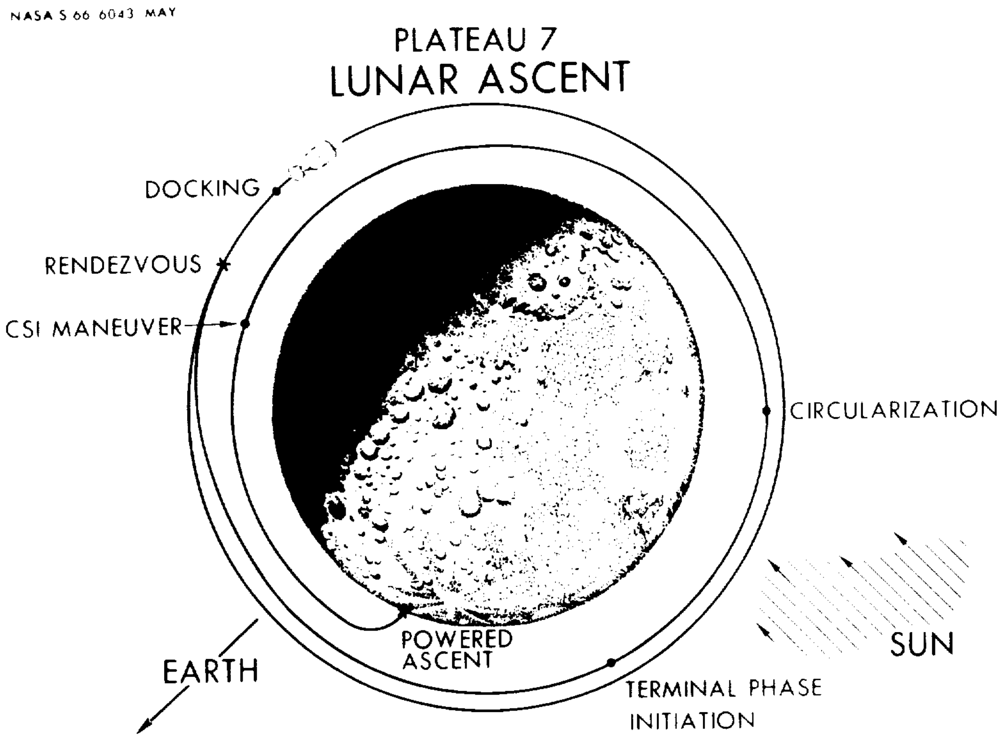

NASA S 66 6043 MAY
PLATEAU 7
LUNAR ASCENT
DOCKING
RENDEZVOUS
CSI MANEUVER
CIRCULARIZATION
POWERED
ASCENT
EARTH
SUN
TERMINAL PHASE
INITIATION

Fig. 48

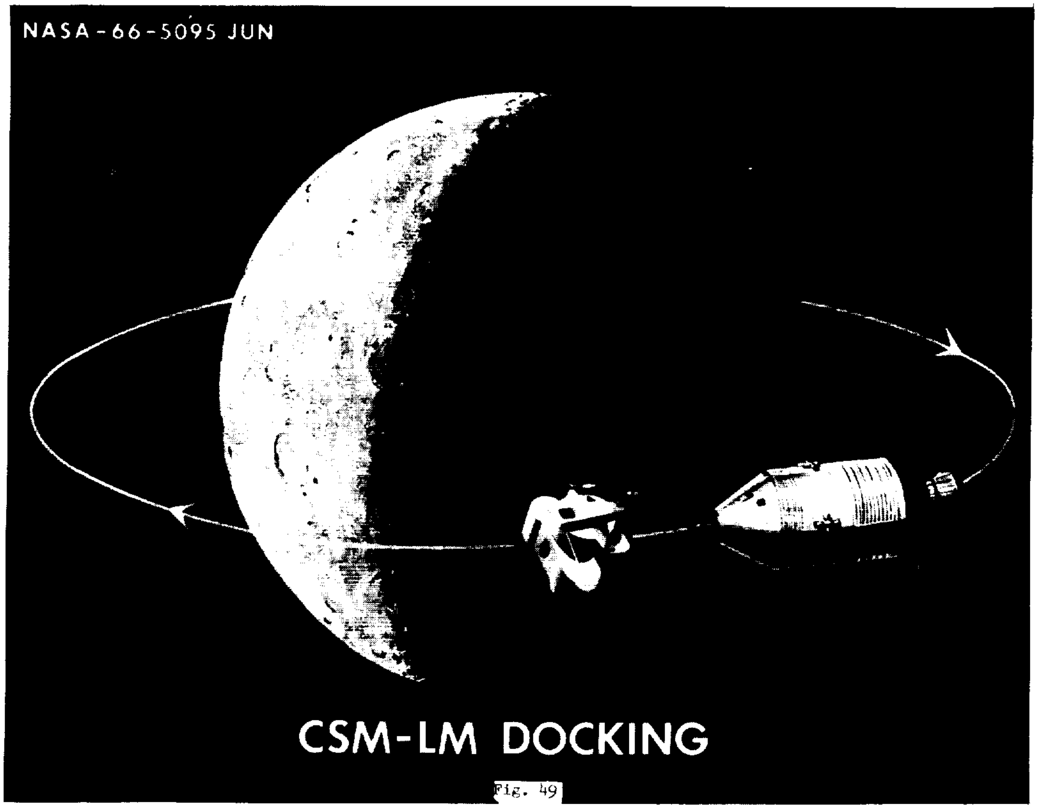

NASA-66-5095 JUN
CSM-LM DOCKING

Fig. 49

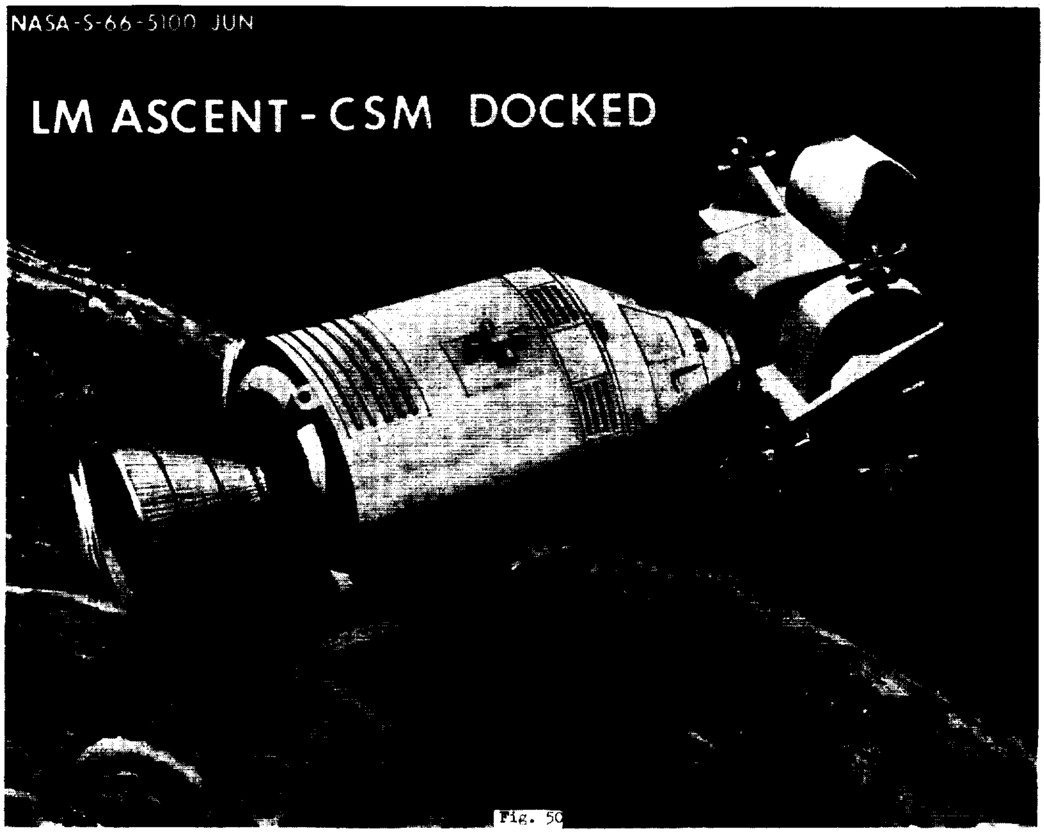

NASA-S-66-5100 JUN
LM ASCENT-CSM DOCKED

Fig. 50

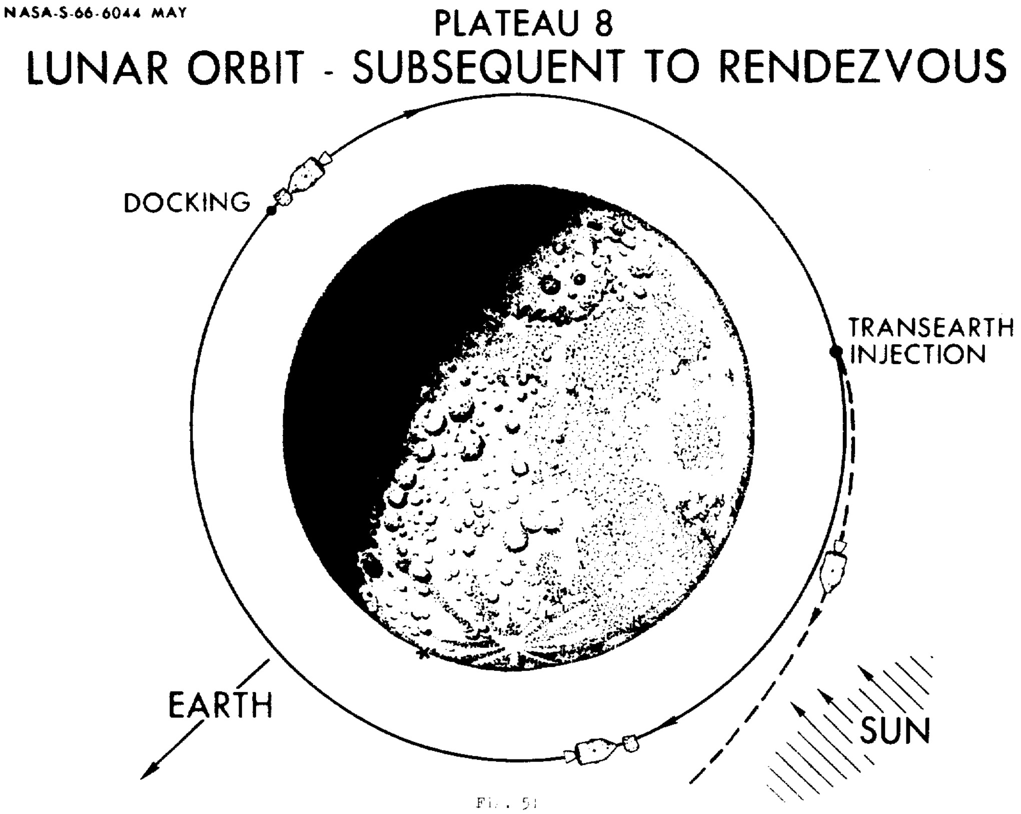

NASA-S-66-6044 MAY
PLATEAU 8
LUNAR ORBIT - SUBSEQUENT TO RENDEZVOUS
DOCKING
TRANSEARTH INJECTION
EARTH
SUN

Fig. 51

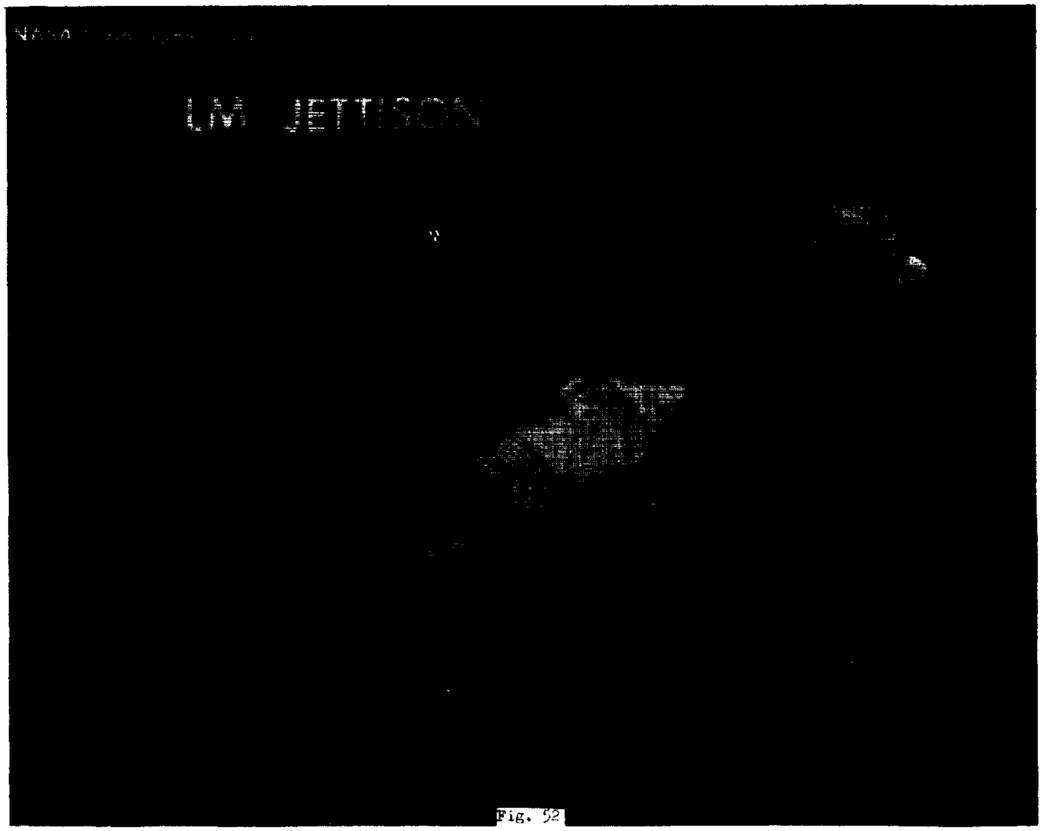

LM JETTISON

Fig. 52

NASA S-66-5090 JUN

TRANSEARTH INJECTION

Fig. 53

NASA-S-66-6045 MAY

PLATEAU 9 - TRANSEARTH COAST

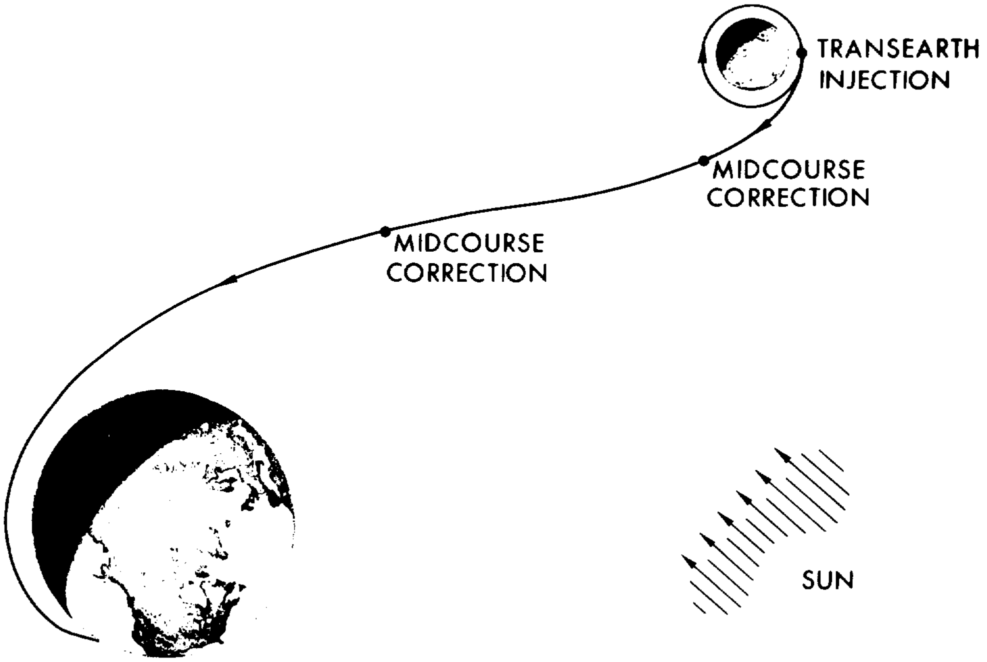

TRANSEARTH
INJECTION
MIDCOURSE
CORRECTION
MIDCOURSE
CORRECTION
SUN

Fig. 54

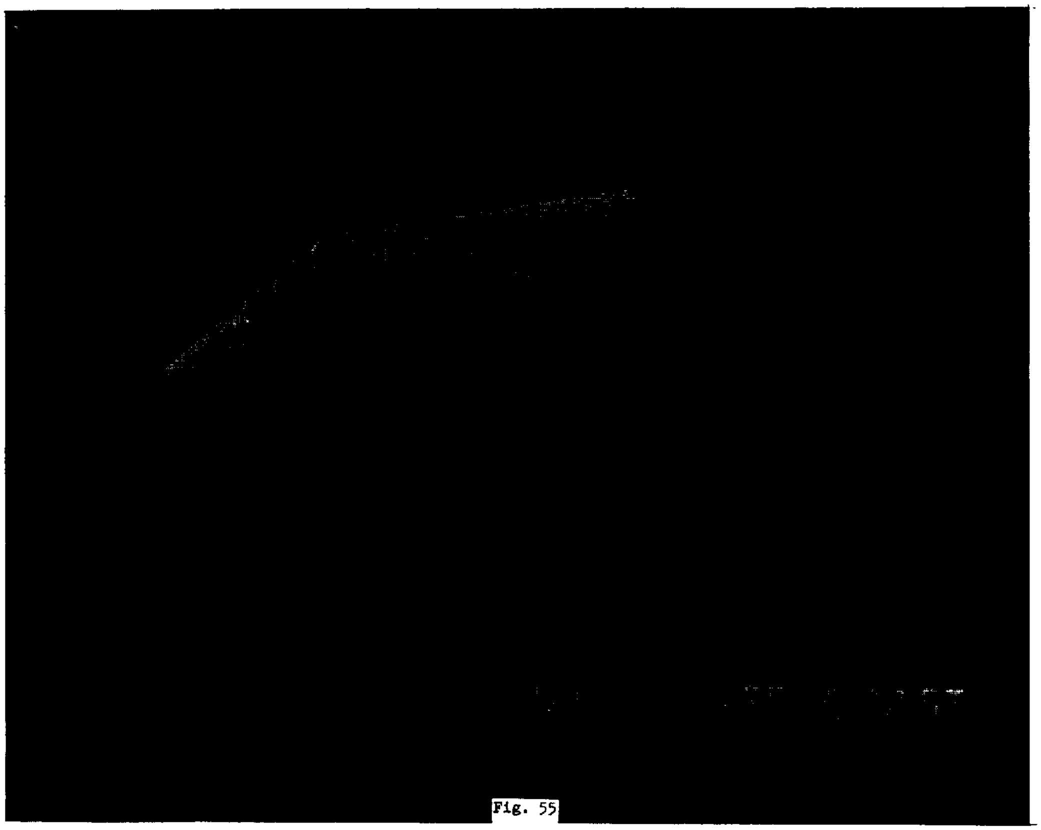


Fig. 55

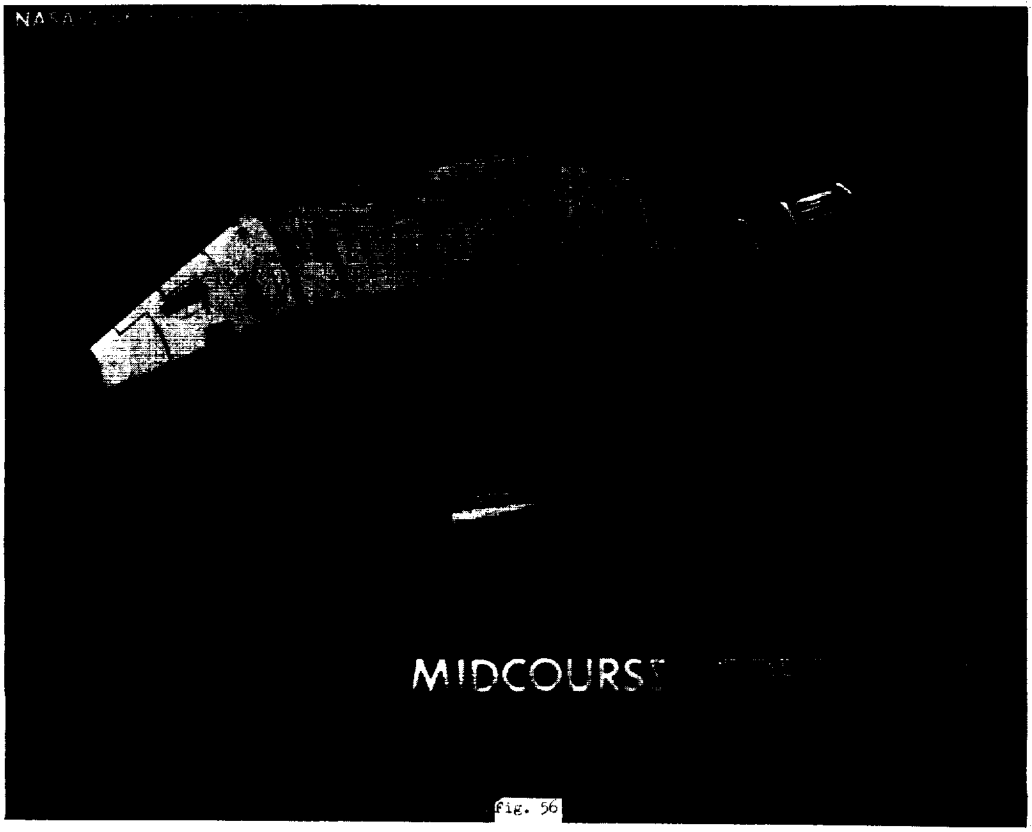

NASA
MIDCOURSE

Fig. 56

NASA-S-66-5107 JUN
CM-SM SEPARATION

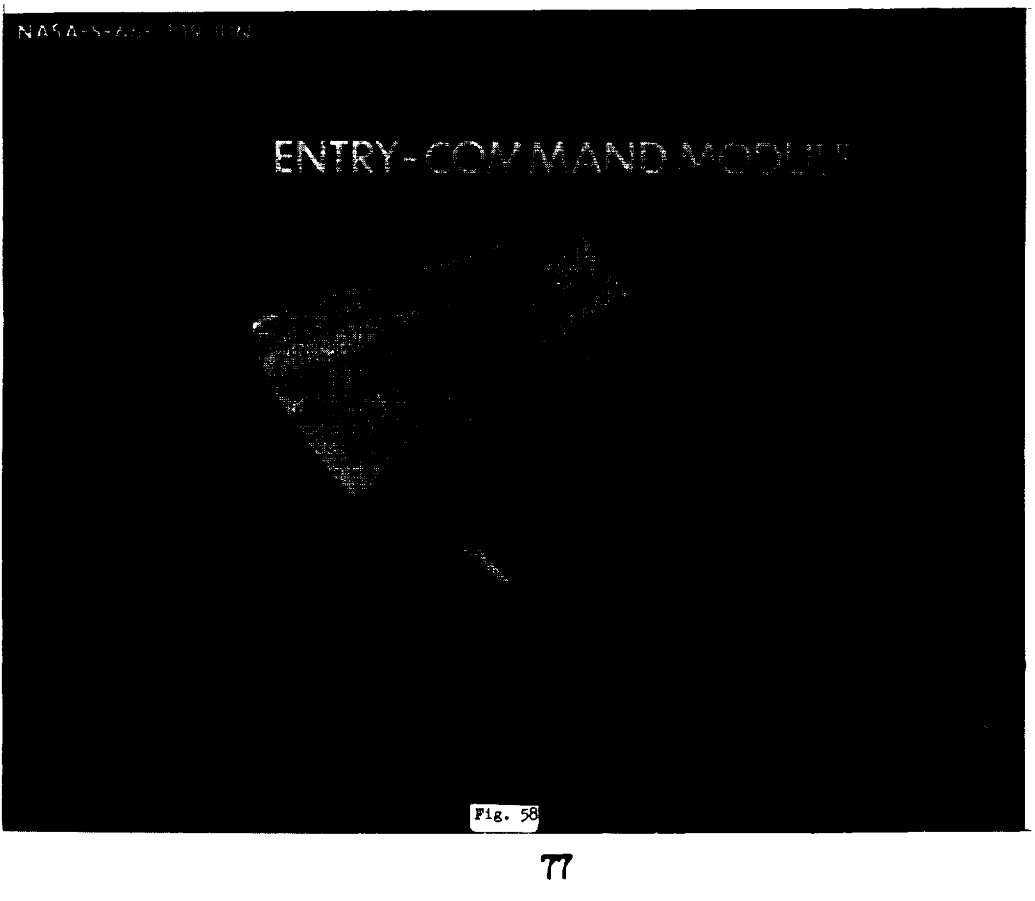

NASA-S-66-[illegible] JUN
ENTRY - COMMAND MODULE

Fig. 58

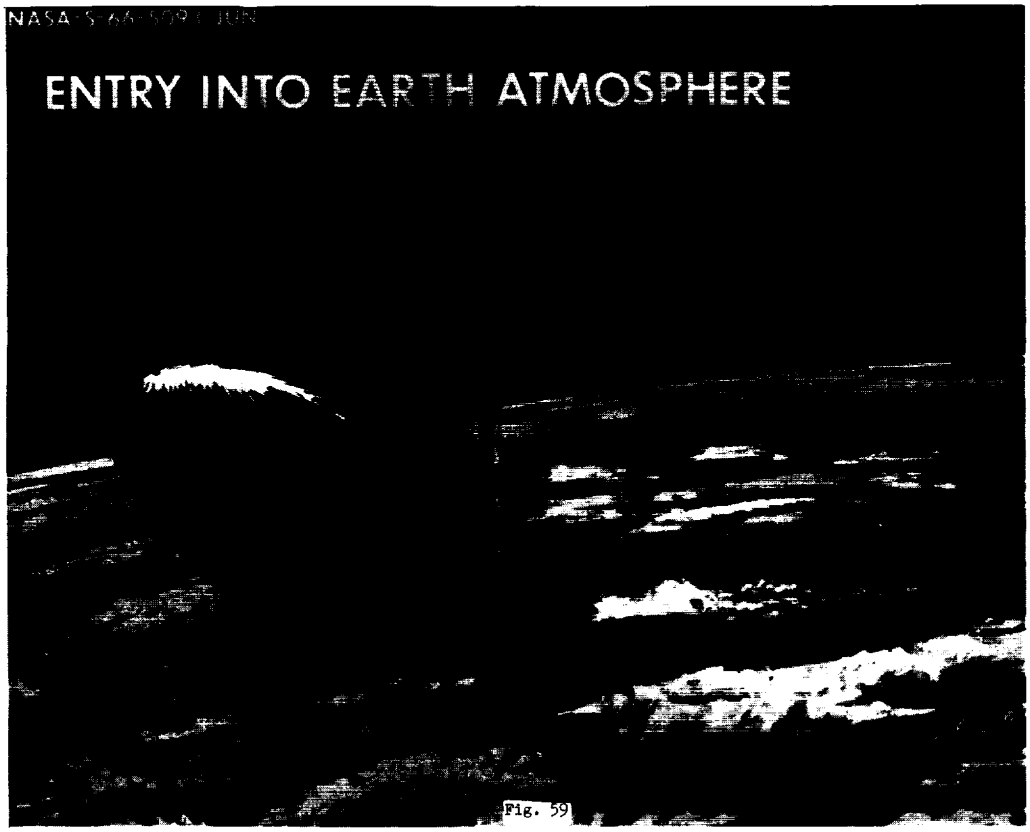

NASA-S-66-5091 JUN
ENTRY INTO EARTH ATMOSPHERE

Fig. 59

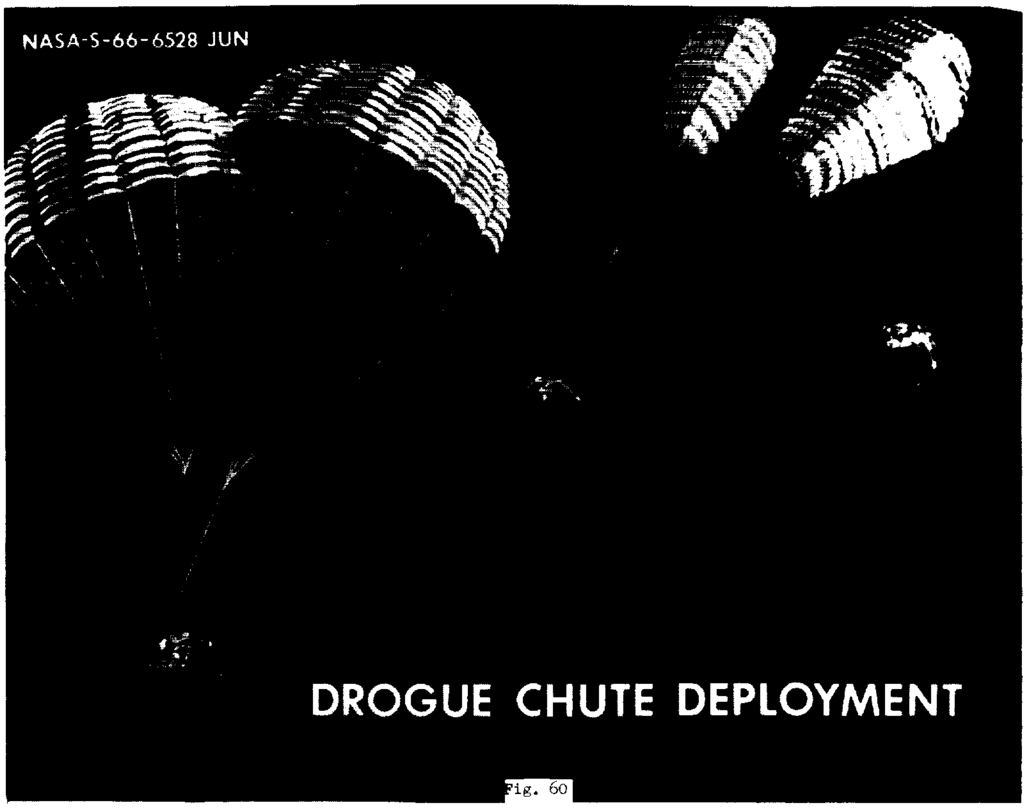
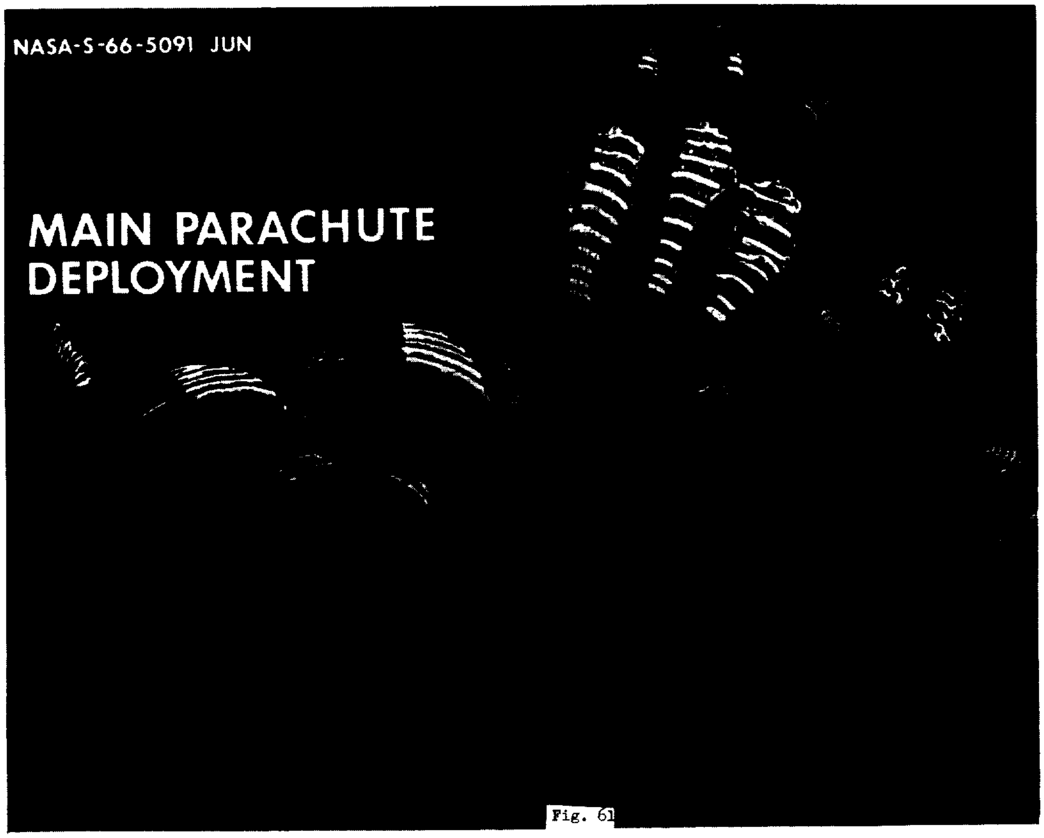
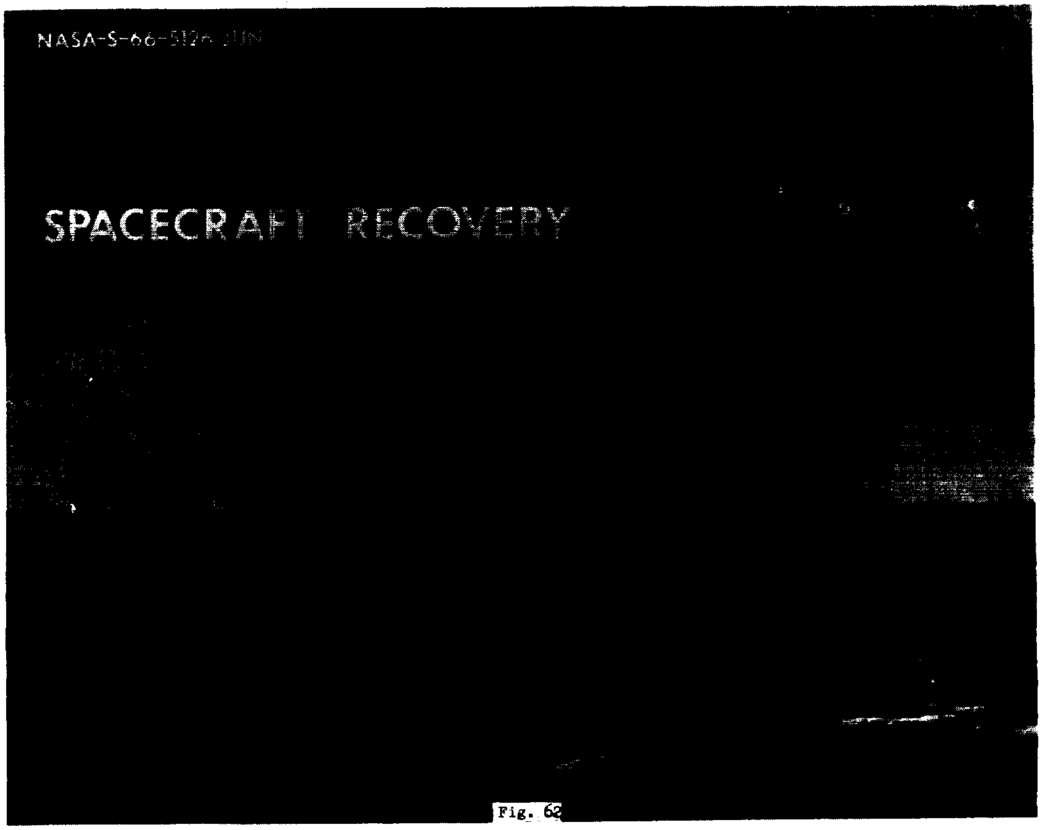

NASA-S-66-6528 JUN
DROGUE CHUTE DEPLOYMENT

Fig. 60


NASA-S-66-5091 JUN

MAIN PARACHUTE
DEPLOYMENT

Fig. 61


NASA-S-66-5126 JUN

SPACECRAFT RECOVERY

Fig. 62

NASA-S-66-6446 JUN

DELTA VELOCITY BUDGET
SERVICE MODULE

	MEAN	3σ	TOTAL
TRANSLUNAR	3513 FPS	94 FPS	3607 FPS

MINIMUM POSSIBLE — 2600 FPS

FLEXIBILITY FOR:
- EVERY DAY LAUNCH OPPORTUNITY
- FREE RETURN TRAJECTORY
- LUNAR LANDING SITE FLEXIBILITY
- TRANSLUNAR AND LUNAR ORBIT TRIM

LM RESCUE

ARBITRARY BUDGET — 680 FPS

REQUIRED FOR TOTAL FLEXIBILITY- — 770 FPS

FLEXIBILITY FOR:
- ANY TIME LM LAUNCH
- LM IN 50,000 FT ORBIT
- CSM COMPLETE RENDEZVOUS AND DOCKING
- RENDEZVOUS AND DOCKING WITHIN 11 HOURS

	MEAN	3σ	TOTAL
TRANSEARTH	3128 FPS	62 FPS	3190 FPS

MINIMUM POSSIBLE- — 2600 FPS

FLEXIBILITY FOR:
- RETURN FROM LUNAR ORBIT CONDITIONS DEFINED ABOVE
- 24 HOUR TRANSFER TIME VARIATION
- 40° MAXIMUM RETURN INCLINATION
- TRANSEARTH TRAJECTORY TRIM

Fig. 63

NASA-S-66-6863 JUN

LUNAR MODULE DELTA VELOCITY BUDGET

	MEAN	3σ	TOTAL
DESCENT	7212 FPS	120 FPS	7332 FPS

MINIMUM POSSIBLE = 6532 FPS

FLEXIBILITY FOR: CSM ALTITUDE VARIATIONS, LANDING AREA OBSERVATION, CHANGE LANDING SITE, LANDING POINT INSPECTION, VARIATIONS DUE TO CREW & EQUIPMENT- RELATED PERFORMANCE CHARACTERISTICS

FUTURE TRAJECTORY CHANGES

	MEAN	3σ	TOTAL
ASCENT	6586 FPS	-	6586 FPS

MINIMUM POSSIBLE = 6176 FPS

FLEXIBILITY FOR: LUNAR LAUNCH WINDOW, CSM ALTITUDE VARIATIONS, ASCENT TRAJECTORY TRIM, 2° ORBIT PLANE CHANGE, DOCKING

Fig. 64

NASA-S-66-6585 JUN

APOLLO SPACECRAFT WEIGHT DATA

	CURRENT (LBS)	PREDICTED (LBS)	MAXIMUM (LBS)
COMMAND MODULE (INCL CREW)	11,755	12,050	12,250
SERVICE MODULE	10,300	11,250	11,300
SM BAY PAYLOAD	0	0	1000
SPS PROPELLANT	37,075	38,800	38,964
LUNAR MODULE (NOT INCL CREW)	30,755	32,000	32,486
SC/LM ADAPTER	3755	3900	4000
TOTAL	93,640	98,000	100,000

Fig. 65

NASA-S-66-6586 JUN

LUNAR MODULE WEIGHT BREAKDOWN

	CURRENT (LBS)	PREDICTED (LBS)	MAXIMUM (LBS)
ASCENT STAGE	4450	4620	4645
RCS PROPELLANT	507	540	540
APS PROPELLANT	4538	4810	4921
DESCENT STAGE	4685	4795	4795
DPS PROPELLANT	16,575	17,235	17,585
TOTAL	30,755	32,000	32,486

Fig. 66

APOLLO NAVIGATION, GUIDANCE AND CONTROL

by

Robert C. Duncan

INTRODUCTION

The purpose of the guidance system is to control the position and velocity of the vehicle. The navigation process involves the determination and indication of position and velocity, and the guidance process involves controlling these quantities in a closed-loop fashion. Fig. 1 shows a generalized functional diagram of the guidance and control system. In order to minimize guidance errors the system must reduce the effect of interferring quantities, and it must respond quickly to command signals. An inertial guidance system is fundamentally mechanized as a specific force measuring system using single axis accelerometers which operate in coordinates that are determined by gyros.

The guidance system operates as a force-vector control system, i.e., the system must change the direction and magnitude of controllable forces (lift, drag, and thrust) in such a way that the vehicle reaches its desired point in space and time. It is usual in the theory of dynamics of rigid bodies in three dimensions to separate the motion of the center of mass from the motion of the body around the center of mass. Guidance is the process of moving the center of mass of the vehicle along some desired path. Stability and control are associated with motions about the center of mass.

The guidance and control systems for all manned spacecraft have involved a mix of spacecraft systems and ground systems. Fig. 2 shows the guidelines used in the Apollo program for this mix of spacecraft and ground systems:

(1) It is mandatory that there be a ground navigation capability provided in earth orbit, cislunar space, lunar orbit, during the lunar landing phases, and during the lunar rendezvous phases.

(2) It is mandatory that the spacecraft contain onboard a completely self-contained navigation, guidance, and control capability to be used in the event that the data link with the ground is lost.

(3) The onboard system is designed in such a way to take maximum advantage of the ground system and to include all necessary interfaces.

Fig. 3 shows the navigation, guidance, and control system which evolved for the command module. The LEM system is very similar and will be discussed later. The primary navigation system in cislunar space is the ground system. This consists of the manned space flight network (MSFN) comprised of a number of tracking stations around the world operating in conjunction with the Houston Mission Control Center (MCC). This system is connected to the onboard system by way of the updata link and voice communications. The updata link provides the navigation state vector to the Apollo guidance computer (AGC). The primary guidance and control system consists of the AGC, the inertial measurement unit (IMU), the scanning telescope (SCT), sextant (SXT), and the display and keyboard assembly (DSKY).

The primary guidance and control system operates the reaction control system (RCS) which is used primarily for attitude control in space and during reentry. The AGC also activates the gimbal servos to drive the service propulsion (SPS) engines. In the event the primary control system has a failure, the backup system (labeled in Fig. 3 the Stabilization System) can also drive the reaction control system and the SPS gimbals. The SCS (stabilization and control system) provides an attitude reference and also has an accelerometer to measure ΔV. The entry monitor system (EMS) is a simplified backup guidance system to be used during the entry phase of the mission in the event of failure of the primary system. An integral part of both the primary system and the backup stabilization system is the astronaut. He obtains information from the computer by the DSKY and from the display panel. He communicates with the computer through the DSKY and is able to control the system through the use of the engine throttle and attitude hand controller.

The stabilization system is shown in block diagram form in Fig. 4. The basic function of this system is:

(1) Drive the jet drivers to turn on and off the small reaction thrusters.

(2) Direct the gimbals of the service module engine to orientate properly the thrust vector of the main engine.

Attitude information comes either from the G&N system (guidance and navigation system) or the AGAP (attitude gyro accelerometer package). Rate information comes from the rate gyro package (RGP) and is displayed on the display panel. Rate and attitude information is used in conjunction with the manual controller to control the attitude jets and the main engine gimbals. The attitude jets can be controlled through two paths, one path via a deadband limiter, pseudo rate logic, and jet select logic to the jet drivers and the other path direct by manual control to the jet drivers. The term pseudo-rate means that the output of the switching amplifier (an on-off device) commands a vehicle acceleration which neglects reaction jet time delays and dynamics. The short period output of this signal through a lag filter is indicative of the vehicle rate. The time constant of the lag network determines the interval over which the output is a valid indication of the vehicle rate. The gains and time constant have been selected for the Apollo SCS to provide the desired signal for an average vehicle inertia during the lunar mission. The configuration of the pseudo-rate feedback for the Apollo SCS has been developed for limit cycle operation. During maneuvers the effect of the feedback should be to pulse the jets prior to the commanded maneuver rate being achieved, thus resulting in an over-damped response. To avoid this, the pseudo-rate feedback is switched out during manual maneuvers.

The guidance and navigation system is located in the lower equipment bay of the spacecraft, Fig. 5.

The G&N equipment is shown in a handling fixture in Fig. 6. The primary components of this system are the DSKY's, the gimbal position indicators, sextant, scanning telescope, displays and controls, power and servo assembly (PSA), and computer. The inertial measurement unit is behind the panel and is mated with the optical system on the navigation base. A precise angular relation must be maintained between the optical system and the inertial measurement unit; this angular relation is provided through the navigation base.

Fig. 7 is a schematic diagram of the gimbals of the IMU. The stable member houses three single-degree-of-freedom 25 IRIG gyros and three 16 PIPA accelerometers. The gyros maintain a coordinate system with respect to inertial space in accordance with reference directions determined by the optical system and gravity. The accelerometers measure specific forces in the three coordinate directions of this inertial reference system. The acceleration measurements are integrated in the computer to give velocity and integrated again to give position. The platform is isolated from the spacecraft by the three-gimbal system shown in Fig. 7. The three-gimballed platform was chosen instead of a four-gimballed platform because it could be built with smaller size and weight. The only disadvantage of a three gimbal platform is that of gimbal lock in certain orientations. This is readily avoided in Apollo by a simple subroutine in the computer program which torques the platform away from potential gimbal locks as the condition approaches.

Fig. 8 shows the Apollo inertial measurement unit (IMU) with the resolvers on one of the outer gimbals removed. This unit is about the size of a basketball and is very similar to a Polaris platform. The corrugations on the outer portion are coolant lines through which the coolant fluid flows to maintain precision temperature control of the IMU.

Fig. 9 shows the IMU with the top removed. Three gimbals, platform electronics, and the accelerometer and gyro package can be seen in this picture.

Fig. 10 is a photograph of the inertial measurement unit and the optical system (scanning telescope and sextant) mounted on the precision navigation base which maintains accurate angular orientation between the two subsystems. The optical system is used to align the inertial system and for navigation in earth oribt, lunar orbit, and in cislunar space. The inertial measurement unit is used as a primary attitude reference and is used for guidance purposes during all maneuvers and during reentry.

Fig. 11 shows the instrument panel in front of the command pilot of the CSM. The switches in the panel to the right control the CM RCS and SM RCS propellant. The switch and dial at the top right indicate the quantity of RCS propellant. The control panel in the center is the display and keyboard assembly (DSKY). This will be discussed in more detail shortly.

The indicator with curved lines and rays at the top left is the entry monitor system. This system is discussed in greater detail near the end of this paper where the entry phase of the mission is discussed. Directly below the entry monitor system is the FDAI (flight director attitude indicator), commonly called the "8-ball" or the "gyro horizon." The needles above, below, and to the right of the 8-ball itself are error needles. To the left of the FDAI are control switches for the SPS (service module propulsion system). Below the SPS switches are the attitude set indicators and controls.

Directly below the FDAI is the "ΔV Remaining" counter and thrust and direct ullage switches. At the bottom are the control mode select switches for the SCS (Stabilization and Control System). It can be seen that the modes available are:

(1) Monitor

(2) G&N attitude control

(3) G&N ΔV

(4) G&N entry

(5) SCS local vertical mode

(6) SCS attitude control

(7) SCS ΔV

(8) SCS reentry

The throttle control is the T-handle in the lower left-hand corner of the photograph.

Fig. 12 shows the faceplate of the display and keyboard (DSKY). The computer in both the command module and LEM are identical. They are microelectronic computers which are designed by MIT and produced by Raytheon. The Apollo computer is a very powerful lightweight computer with the largest memory of any airborne computer in history. It has a memory of 36,000 words (each of 16 bits) and is approximately equal to an IBM 704 in computational capability.

The DSKY provides the communication link between the astronaut and the computer. Through the DSKY the astronaut can monitor system activity, alter parameters, and dictate system modes. In addition, the DSKY has indicator lights which display system and computer status and alarm. The computer display on the DSKY consists of three two-digit displays labeled "Program", Verb", and "Noun" and three five-digit general word readouts. The two-digit displays are coded for various modes in instruction. The program display indicates the major operating mode of the computer such as "lunar landing maneuver." The "verb" and "noun" displays are used together and coded to give numerous possibilities of meaningful phrases or instructions. Examples of typical "verb" and "noun" displays are:

Verb	Noun
Display value	Velocity
Compute	Abort velocity
Read in	Landmark angle

When the computer wishes to communicate a request for data or signal an alarm to the astronaut, the "verb" and "noun" numbers flash until the astronaut takes action. He enters data to the computer through the keyboard which is on the right hand side of the display as seen here.

A schematic representation of the operation of the manned space flight network tracking system (MSFN) is shown in Fig. 13. The vehicle is illuminated by an 85 ft. antenna which provides range, angles, and velocity. This information is transmitted to the Mission Control Center in Houston from which navigation information is determined. The vehicle can also be tracked by 30 ft. antennae which use three-way doppler information to provide position and velocity data.

Distance is determined by modulating the carrier with random digits (0 and 1). The signal is received by transponders in the CSM or the LEM and retransmitted. The measurement of transit time of the signal is a measure of the distance of the spacecraft. Velocity is determined by measuring the doppler shift in the signal returned by the spacecraft.

Fig. 14 shows the location of MSFN unified S-band station sites. There is overlap of coverage among the three 85 ft. stations. These stations are located in the western United States (Goldstone, California); Madrid, Spain; and Canberra, Australia. Additionally, there are ten 30 ft. antennae spaced strategically around the world (Bermuda, Carnarvon, Guaymas, Hawaii, Cape Kennedy, Corpus Christi, Houston, Guam, Ascension, and Antigua).

Fig. 15 summarizes the characteristics of the tracking stations. The one-sigma tracking accuracies are as follows:

Range: Noise 60 ft. Bias 120 ft.

Angles: Noise 0.8 milliradians Bias 1.6 milliradians

Two-way doppler: Noise 0.1 ft/sec Bias .07 ft/sec

Three-way doppler: Noise 0.1 ft/sec Bias 0.2 ft/sec

The frequency reference is a rubidium frequency standard with a short and long term stability of 5×10^{-11} parts per part. The MSFN stations with either the 30 or 85 ft. antenna can track spacecraft at lunar distances using either the high-gain antenna or omnidirectional antenna.

Fig. 16 summarizes the operating modes of the system during a mission. During the earth orbital phase, the system uses C-band and measures range and two angles. During the cislunar phase, the system uses the unified S-band measuring three-way doppler.

Onboard derived navigation data is telemetered to the MSFN. The radar tracking data and the telemetered data are piped into the Mission Control Center at Houston and the guidance and navigation parameters are then computed in the real-time computation center (RTCC). After the guidance and navigation data is determined, it is telemetered to the Apollo guidance computer onboard the spacecraft.

Fig. 17 shows the measurement uncertainties of the MSFN system. Here we assume that the system is operating in the three-way doppler mode with one 85 ft. station transmitting and two stations receiving. The one sigma measurement uncertainty in moise and bias for two-way doppler and three-way doppler were indicated in Fig. 15. Additionally, we assume an uncertainty (one sigma) in location of the station in latitude and longitude to be 1 to 6 arc seconds. The altitude error is assumed to be 100 to 200 ft. The uncertainty of the gravitational constant (one sigma) for the earth is assumed to be 106 x 10^9 ft^3/sec^2. For the Moon, this is assumed to be 6 x $10^9 ft^3/sec^2$. In the orbit determination program, the parameters estimated are three components of velocity, three components of position, and two three-way doppler biases. Using these assumptions, the accuracy in performance of the system was computer and will be discussed shortly.

Let us look now at the techniques for navigating onboard the spacecraft. One technique for navigating is shown in Fig. 18. Here a star is picked up with the scanning telescope and centered, at which time the astronaut shifts to the sectant (a 28-power narrow field-of-vision instrument). The astronaut uses the sextant to position accurately the star over the landmark. When the star and the landmark are superimposed, the astronaut presses a button and the angles between the two, as well as time, are entered into the computer automatically. This information is entered by the astronaut via the DSKY. Fig. 18 shows a star superimposed on the Golden Gate Bridge.

In earth orbit the astronaut can measure his position by tracking known landmarks which are entered into Keplerian equations in the computer. It is also possible, by knowing the error propogation equations, to track unknown landmarks and to compute his position accurately in earth or lunar orbit. At the same time the computer determines the geographic position of the unidentified landmark. When the Apollo program was initiated, it was planned that known landmarks would be tracked and navigation would be performed in this way. During the many Gemini flights, however, it has been observed that it is quite difficult to plan ahead for those landmarks which will not be obscured by clouds. On most Gemini missions much of the earth has been obscured. If one is limited to known and predetermined landmarks, mission planning becomes quite complex. Therefore, it is likely that greater and greater reliance in Apollo will be placed on unknown landmark tracking for both earth orbital and lunar orbital navigation.

Let us look at the geometry of measuring a navigational fix in cislunar space. Fig. 19 shows the various angles involved using the stars Fomalhaut, Deneb, and Antares. In this geometrical sketch, the lunar horizon is used with Antares and navigational landmarks are used with Fomalhaut and Deneb. The angles measured with the three stars form three cones in space. The intersection of two of these cones forms a line and the intersection with the third cone forms a point.

Computation is performed automatically in the computer. Apollo does not use the conventional method of computation performed by mariners for many centuries, i.e., computation based upon two or more star sightings and running the earlier sightings forward to the last sighting and computing a fix. This we call "deterministic" techniques. In Apollo, recursive navigation techniques are used (involving Kalman filter theory). Under this concept the accuracy of position and velocity determination is improved as more and more sightings are taken and the uncertainties are reduced with each sighting. This method involves statistical mathematics techniques. Fixes as they are known in maritime navigation are not performed in Apollo.

Another method of navigation originally planned in Apollo is shown in Fig. 20. Here the elevation angle of a star is measured with respect to the earth horizon. It is common knowledge that a precise definition of the horizon is difficult in space due to cloud cover and the uncertainty in the definition of the terrain horizon. Apollo planned to use a horizon at 100,000 ft. to be measured by a horizon photometer operating at a specific narrow frequency in the optical band. This horizon measurement is made automatically and the angular information is used in much the same way that mariners use elevation angle. The computational procedure involves recursive navigation cited earlier. Due to technical problems in the development of the horizon photometer electronics, this system will not be flown in early Apollo spacecraft.

Fig. 21 shows the accuracy of the navigation and guidance system using the optics in the way cited previously. In the earth orbital phase scanning telescope uncertainties are 4 milliradians and landmark position uncertainties are 1000 ft. Nominally, seven landmark sightings are made. During the translunar and transearth phases, the sextant uncertainty is 10 arc seconds and the uncertainties in the horizon are 1 nautical mile for Earth and 0.5 nautical miles for Moon. Additionally, we assume that forty landmark sightings are made enroute to the Moon and enroute back to the Earth. The velocity corrections made enroute to the Moon and returning to the Earth are accurate in magnitude of 1%. The pointing of the thrust vector is accurate to 10 milliradians.

Fig. 22 continues with the accuracies of the onbaord navigation system. In the lunar orbital phase, the scanning telescope uncertainty is 4 milliradians and the landmark position uncertainties on the Moon are 1000 to 5500 feet in a horizontal direction and 2300 to 3000 feet in a vertical direction. Five landmarks on the earthside are chosen along the lunar orbit track and three sightings per landmarks are made on each of the first two orbits. Periodic updates are made thereafter.

GUIDANCE AND NAVIGATION PERFORMANCE

Let us look now at the performance of the system using the performance of the manned space flight network (MSFN) and the onboard system discussed previously. Fig. 23 summarizes the results of a digital computer simulation program using the accuracies cited previously. The RMS position measured by the onboard system and by the MSFN is given in nautical miles and velocities are measured in feet per second. The accuracies are shown at injection as measured either by land stations or by ship. Also shown are accuracy at the first midcourse, second midcourse, and third midcourse corrections enroute to the Moon and at perilune. In lunar orbit, the accuracies are shown at LEM separation and LEM rendezvous. The return accuracies are shown for injection at the transearth phase and at first, second, and third midcourse corrections and at entry. You will note that at Earth reentry, the MSFN accuracy is 0.5 nautical miles while the onboard system accuracy is about 9 nautical miles. With the MSFN, velocity is accurate to 2.2 ft. per second while the onboard system is inaccurate to the extent of about 45 ft. per second.

Fig. 24 is a comparison of the data as measured onboard and by the MSFN during the translunar phase. The top plot is position and the bottom plot is velocity as a function of time. The MSFN system is more accurate until about 35 hours after the spacecraft is injected toward the Moon. At that time the onboard system is of somewhat greater accuracy. The midcourse corrections are indicated here by the diamonds labeled MCC. These are made at two hours after injection, 49 hours after injection, and 61 hours after injection.

Fig. 25 gives surprising results concerning navigational uncertainties during the lunar orbit phase. The earth-based manned space flight network is more accurate than the onboard system throughout the trajectory. Time is measured from the time of spacecraft insertion into lunar orbit. The solid lines show points of MSFN tracking and the dashed lines show periods during which telescope sightings of lunar landmarks are made. Of course, errors grow during the period between tracking or sightings and they immediately drop to more accurate values as greater data is accumulated.

Fig. 26 shows the midcourse velocity performance during the translunar phase. The bottom plot shows the ft/second uncertainty in velocity correction which would be made as a function of time from injection to the first velocity correction as measured by the MSFN and the onboard system. The upper of the two figures shows the sum of the first two velocity corrections. There is a significant velocity savings by using MSFN data. These figures show that it is worthwhile to delay, from a fuel standpoint, before making velocity corrections. These figures are indicative of the fuel economies which could be made as the uncertainties in velocity corrections are reduced.

Fig. 27 describes in simple form the interface between the onboard system and the manned space flight network. The computations for the MSFN are made by the Houston Mission Control Center. The interface with the computer is made via two routes, the up and down data link directly from the MSFN to the Apollo guidance computer and voice communications to the crew who enter the data through the DSKY to the computer. The computer updata rate is 10 up-link words per second. The total navigation updata information required is a state vector consisting of six components and time. The updata frequency required is once prior to each guidance maneuver. Total transmission time is 60 seconds maximum (for a 99% probability of no errors). The navigational updata information is not time critical.

APOLLO REENTRY

The end result of the lunar mission is a safe reentry into the earth's atmosphere. Fig. 28 represents this problem. The guidance system must hit a corridor approximately 26 miles deep. This accuracy is obtained in three midcourse corrections during the return flight from the Moon. Prior to entering the earth's atmosphere, the service module is jettisoned. If the spacecraft comes in above this entry corridor, too little energy will be transferred from the vehicle to the atmosphere and the spacecraft will enter a highly elliptical earth orbit. The eccentricity of this orbit is a function of how far above the upper entry boundary the spacecraft trajectory carries it. Since there is no significant propulsion onboard the CM other than the 100 pound RCS system, there would be no way for the astronaut to recover from this highly elliptical orbit. It is necessary that this entry corridor be entered. If the astronauts come in too low they will exceed the heat capabilities of the heat shield.

One of the most critical portions of the mission from the guidance and control standpoint is the reentry phase. In Apollo the earth's atmosphere is entered by making one skip. The total range of the vehicle from the point of entry is nominally 2500 miles. No propulsion system is used to remove energy from the vehicle; all of the energy must be removed by the atmosphere. We expect to land within 30 miles of the chosen landing point.

ENTRY TECHNIQUES

There are many different techniques which may be used to guide a spacecraft during the atmospheric entry maneuver. The selection of a particular technique is influenced by a number of considerations, some of which are shown in Fig. 29. The most important consideration which will influence the entry guidance is the requirement for the safety of the crew. The velocity and angle at which the spacecraft enters the atmosphere affects the entry guidance. The entry guidance must consider the physical properties of the spacecraft and the amount of range control which is required. The entry guidance is limited by the physical quantities which can be measured and by the size and speed of the onboard computer. There must be some technique for monitoring the primary entry guidance in order to insure a safe entry. The primary guidance system and the entry monitoring system must be compatible with each other.

The maximum permissible acceleration is normally considered to be 10 g's although the maximum emergency acceleration may be as high as 20 g's. The protection from the aerodynamic heating is provided by the heat shield, but entry guidance must minimize heating problems.

The Mercury and Gemini spacecraft entered at near circular velocities while the Apollo spacecraft enters at near parabolic velocities. The task of the entry guidance system becomes more difficult as the entry velocity increases. The entry angle must be kept within a value which will allow the spacecraft to be captured by the atmosphere and a value which will not result in excessive acceleration.

The aerodynamic properties of the entry vehicle may be divided into the general categories shown in Fig. 30. The first group consists of vehicles such as the Mercury spacecraft which do not develop any lift and which have a constant ballistic number (W/C_DS). In this discussion the aerodynamic properties are considered to be variable only if the values of these quanties can be controlled by the guidance system. Although the aerodynamics properties are termed constant they may vary as functions of Mach number and Reynolds number. The second group listed in Fig. 30 are vehicles which do not develop lift but whose drag properties can be varied. Such a vehicle would have a small amount of range control. The third group includes both the Gemini and Apollo spacecraft. The vehicles of this group develop lift but their lift-to-drag ratio (L/D) and their ballistic number are constant. The trajectory which is followed by a spacecraft of this type can be controlled only by rolling the spacecraft. This changes the direction of any lift which is developed. Finally there is the last group of vehicles for which the ratio of lift-to-drag and the ballistic number can be varied.

ENTRY GUIDANCE SCHEMES

An entry vehicle which is designed to develop lift to change the path of the vehicle overcomes many of the problems which are inherent with a ballistic vehicle. The guidance system of such a vehicle must be capable of utilizing this lift properly. Many of the characteristics of the entry guidance will depend upon the amount of lift which can be generated and the manner in which the lift is generated. Both the Gemini and Apollo spacecraft are trimmed at a constant angle of attack. This results in a constant lift-drag ratio and a constant ballistic number. Therefore the only way that the path of the spacecraft can be controlled is by rolling the spacecraft about its longitudinal axis.

The possible entry guidance methods may be divided into two general classifications:

(1) Guidance using predicted capabilities

(2) Guidance using a nominal trajectory

The choice of which type to use depends upon considerations such as the size and speed of the onboard computer and the range of entry conditions which the guidance system must be capable of handling. The dividing line between the two classifications is somewhat obscure. It is possible that an entry guidance logic will use elements of both techniques.

In the method of guidance about a nominal trajectory, the state variables along the nominal path are precomputed and stored onboard the spacecraft. The variations in the measured variables from the stored values are used in the guidance logic either to control the spacecraft back to the nominal trajectory (path controller) or to establish a new trajectory to reach the destination (terminal controller). For this guidance logic, a desirable nominal trajectory must be selected. The desired nominal trajectory is selected prior to the entry by optimization procedures.

The method of guidance using predicted capabilities is capable of handling a wider variety of entry conditions than the guidance about a nominal trajectory. This guidance technique predicts the path by which the vehicle will reach the desired destination without violating the heating and acceleration limits. The prediction of the future trajectory may be accomplished by a rapid forward integration of the equations of motion for the remainder of the flight, or by using approximate closed-form analytic solutions to the equations of motion. The main advantage of the fast prediction method is that it is able to handle any possible flight condition. The principal disadvantage of this method is the requirement for speed in the computer. The use of closed-form solutions reduces the required computational speed and flexibility of the guidance system.

GEMINI ENTRY GUIDANCE

Projects Gemini and Apollo furnish this country with experience in the entry guidance of lifting manned spacecraft. Two forms of entry guidance logic are used in the Gemini program. The same footprint capability of about 600 nautical miles across range exists for each technique. The two techniques may be termed rolling entry and fixed-bank entry.

In the rolling entry technique the steering logic is based on calculating the difference between the actual range to go and the predicted range based on a continuously rolling entry. During entry, the ratio of the downrange error to the crossrange is used to compute the bank angle required to rotate the axis of symmetry of the footprint the amount required to pass through the desired touchdown point before the downrange goes to zero. When this rotation is completed, the spacecraft flies at zero bank angle until the downrange error is zero. At this point a continuously rolling entry is initiated. Because downward lift is not used, the range calculation is biased slightly to predict a greater range than the time value, thus preventing the target from moving outside the footprint. The continuous rolling will therefore be interrupted occasionally for correction by flying near zero bank angle.

The second technique appears to have certain advantages over rolling entry. The entry trajectory is flown at a fixed series of bank angles. The prediction is based on the range obtained at a fixed bank angle and the bank angle is adjusted to make the predicted range agree with the desired range to the touchdown point. Crossrange error is allowed to increase until it reaches a fixed percentage of the lateral range capable at that particular time. The bank angle is then reversed. Thus the entry is flown in a series of reversals of an essentially constant bank angle. Theoretically, any point in the footprint is available with only one bank angle reversal, but guidance inaccuracies will generally require additional maneuvering.

The fixed-bank-angle method has the advantage of being more compatible with the crew monitoring function than the rolling entry method. By viewing the horizon out the spacecraft windows the astronauts can compare the maneuvers with those expected for the entry condition and take over control in the event of a malfunction. The manually controlled backup technique is similar to the automatic method, but uses a precomputed program of bank angles. The fixed-bank-angle method will require less attitude control fuel than the rolling method. Figure 31 summarizes the results of reentry during the Gemini program to date. The early flights of Project Gemini dramatically demonstrated the effect of the difference between the estimates of the aerodynamics obtained from wind tunnel tests and the aerodynamics of the actual spacecraft. The unmanned flight of GT-2 indicated that the L/D ratio of the spacecraft was higher than the predicted value over most of the Mach number range. This led to the removal of 58 pounds of non-functional ballast for the first manned flight (GT-3) piloted by Grissom and Young.

The rolling entry technique was used for the entry of GT-3. This required the crossrange error to be eliminated first and then the downrange error was to be eliminated. One purpose of this flight was to check out the onboard guidance system. For this reason the spacecraft was controlled manually to the ground-computed commands and the commands generated by the onboard guidance were observed. The crossrange error was eliminated. Although full position lift was used for the remainder of the flight, the GT-3 spacecraft fell 64 nautical miles short of its target point. If the bank angles computed by the onboard guidance have been used, the spacecraft would have come closer to the target but it would not have reached the target.

The GT-3 spacecraft fell short of the target, chiefly, because it developed a lower L/D than had been estimated before the flight of GT-2. The footprint of the Gemini spacecraft was reduced to about one-third of its original estimate size because of the reduced L/D. The estimated footprint is about 200 nautical miles long and 34 nautical miles wide. The length is measured from the zero lift point to the maximum lift point.

It was felt that the reduced footprint was not any cause for concern. It was decided to readjust the retrofire times in order to place the reduced footprint over the target point. Also, the crossrange and downrange errors were to be eliminated simultaneously at the beginning of the entry when the effective lift is the greatest.

The GT-4 spacecraft missed the target point by 47 nautical miles. The guidance computer had failed prior to entry. The spacecraft was flown on the basis of information furnished by the ground-based computers.

As in the case of the first two manned Gemini flights, GT-5 fell far short of its target point. In fact, it was the worst miss of the program. The spacecraft fell 97 nautical miles short of the target. Subsequent study disclosed that most of the miss was due to a ground error in failing to provide the onboard computer with the spacecraft's proper inertial coordinates at the time of retrofire. The information computed by the ground system ignored the fact that the Earth rotates approximately 361 degrees in 24 hours instead of 360 degrees. This error was present during the flights of GT-3 and GT-4 but the duration of these flights was not long enough for the effects of this error to become pronouned. By the end of the eight-day GT-5 mission this error was approximately 8 degrees. As a result of this error the onboard guidance attempted to steer the spacecraft to a target which was approximately 480 nautical miles closer to the actual target point. The miss distance would have been greater except that the Gemini entry guidance is prohibited from rolling the spacecraft to develop negative lift. The entry guidance called for a zero-lift entry which is the minimum lift allowed. By the time the astronauts realized that the guidance was giving erroneous commands, the range capability of the spacecraft was insufficient to reach the target point. During the entry the guidance was providing the correct commands to guide the spacecraft to the false target.

The flights of GT-6 and GT-7 indicate that the problems previously experienced with the entry have been solved. The actual landing point of GT-6 was within 7 nautical miles of the target point. The spacecraft was manually controlled to the bank angles commanded by the onboard computer. The retrograde maneuver was performed with the acceptable tolerance. After the retrofire and the jettisoning of the retro-adapter section, the spacecraft was rolled to the full positive lift position. At 290,000 feet the onboard computer fed bank angle commands to the flight director displays. The nominal bank was first 47 degrees left and then 47 degrees right. These bank angles were held to 80,000 feet. The drogue parachute was deployed at about 50,000 feet and the main chute was deployed at about 10,500 feet. The landing point of GT-7 was also within 7 nautical miles of the target point.

APOLLO ENTRY

The obvious difference between the guidance problems of Gemini and Apollo is the difference between the entry velocities of the two missions. The Gemini spacecraft enters at near circular velocities while the Apollo Spacecraft will enter at near parabolic velocities. The much higher entry velocity of the Apollo spacecraft greatly increases the possibility than the acceleration limits will be exceeded. The range of a spacecraft entering at near parabolic velocities can be increased by allowing the spacecraft to skip out of the atmosphere. The time of flight may exceed the duration of the power supply or the lift support system, if the exit velocity is too high.

The Apollo entry guidance is a combination of the technique of using predicted capabilities and the technique of using a nominal trajectory. The guidance logic during the initial phases uses predicted capabilities while the guidance logic during the final phase uses a nominal trajectory. The logic is divided into portions which reflect the characteristics of the entry trajectory which the Apollo spacecraft will follow.

The typical Apollo entry trajectory can be divided into the general areas which are shown in Figure 32. At some time prior to entry, the entry vehicle is separated from the Service Module and aligned to the entry attitude; the IMU is aligned and the navigation system is updated for the last time. The initial bank angle depends upon whether the vehicle is entering at the top of the corridor or at the bottom. If the spacecraft is entering close to the top of the corridor, the bank angle will be such that the lift is directed downwards in the vertical plane. Otherwise, the lift is directed upwards in the vertical plane.

Upon encountering the sensible atmosphere the remaining phases of the entry are:

(1) An initial phase during which a safe capture is ensured and excessive acceleration is avoided.

(2) A second phase during which the vehicle is steered so that the final phase will be able to guide the spacecraft to the target.

(3) A ballistic lob which may be bypassed if the range to the target is short.

(4) The final phase during which the spacecraft is steered to the target. This final phase usually includes the last six to eight hundred miles of the entry and is similar to the entry from a low orbit about the Earth.

The basic flow of the guidance logic is shown in Figure 33. Certain portions at the beginning and end of the system are entered each time the steering commands are computed. These portions are the NAVIGATION, TARGET DATA, and ROLL COMMAND sections. The LATERAL LOGIC section is also entered each time the steering command is computed except during the initial phase of the entry and during any ballistic lob. If the spacecraft is returning from a low altitude orbit, the PHASE SELECTOR will be set so that only the BALLISTIC PHASE and the FINAL PHASE are used.

The NAVIGATION section calculates the inertial positon and velocity of the vehicle by using simple numerical integration techniques. The measured acceleration is combined with a calculated gravitational acceleration and then is used in the equations of motion.

The TARGET DATA section calculates the distance between the position of the spacecraft obtained by the NAVIGATION section and the inertial position of the target at the estimated time of arrival. The inertial velocity of the spacecraft is used until the velocity becomes less than approximately one-half the circular velocity at an altitude of 300,000 feet altitude.

The PHASE SELECTOR directs the logic to the section which is concerned with the phase of the entry that the spacecraft is in at that time. Initially, the PHASE SELECTOR will direct the logic to the INITIAL ROLL section or to the BALLISTIC PHASE section. The correct path will depend on whether the spacecraft is returning at near parabolic velocity or at near circular velocity.

During the initial phase of entry, the bank angle is held constant until the drag exceeds 2 g's. At this time the vehicle is rolled to a zero-bank angle which is held constant until the second phase is entered. The second phase is entered when the altitude rate is greater than -700 feet per second.

The second phase is the heart of the Apollo entry guidance system. This section inclues the logic which will decide the trajectory which will be flown in order to reach the target. There are three paths which the logic may follow after this section. The three paths are to the CONSTANT DRAG section, to the UP CONTROL section, or to the FINAL PHASE section. If the predicted exit velocity is greater than the circular velocity or the predicted range capability is greater than the actual range plus 25 nautical miles, the CONSTANT DRAG section is used. The second phase will be reentered during the next computation cycle which is two seconds later. The logic will be directed to the FINAL PHASE whenever the predicted exist velocity is less than 18,000 feet per second. The UP CONTROL section is entered only when the difference between the predicted range to the target and the range of the calculated trajectory is less than 25 nautical miles.

The CONSTANT DRAG section attempts to guide the spacecraft along a constant acceleration path. The value of the acceleration is calculated the first time the second section of the guidance logic is entered.

The UP CONTROL section attempts to guide the vehicle to the calculated exit conditions. The guidance system continues to use this section until the drag becomes less than some arbitrary value or the altitude rate becomes negative. If the drag falls below this arbitrary value, the guidance logic enters the BALLISTIC PHASE section. If the altitude rate becomes negative first, the FINAL PHASE section is entered.

During the ballistic lob, the BALLISTIC PHASE checks for the start of the final phase of the trajectory. The attitude of the spacecraft is controlled during the ballistic lob so that the sideslip is zero and the spacecraft is trimmed about its nominal angle of attack.

The FINAL PHASE section uses a stored table of values to attempt to steer the vehicle along a reference trajectory to the target. The steering command of the FINAL PHASE will be modified by the "G" LIMITER whenever the drag is greater than 5 g's. The bank angle will be decreased in an attempt to keep the drag level below 10 g's.

The output command of the CONSTANT DRAG, UP CONTROL, and FINAL PHASE sections is a value of L/D. The commanded roll angle is determined by the ratio of the commanded value of the L/D to the maximum value of the L/D.

The LATERAL LOGIC section is used to decide to which side of the vertical plane the vehicle should be rolled. Basically, the vehicle is rolled toward the target. To avoid a large number of roll reversals, there is a deadband built into the logic. That is, lift may be directed away from the target if the predicted landing point is within limits. This limit has arbitrarily set at on-half the lateral range capability.

The final section of the guidance logic is the ROLL COMMAND. This section selects the direction of roll which will result in the shortest angle to be travelled. The command from this section is then transmitted to the reaction control system.

The entry of a spacecraft at near parabolic velocities presents a number of sources of danger to the crew. The chief dangers are excessive accelerations and exiting along a trajectory which would exceed the lifttime of any of a number of onboard systems such as lift support, power, and attitude control fuel. It is desirable to have an independent and reliable system for monitoring the primary guidance system.

The entry monitoring system (EMS) must be sufficiently accurate to detect impending unacceptable trajectory characteristics such as excessive accelerations or an uncontrolled atmospheric ship in sufficient time to prevent their occurrence. The EMS must not unnecessarily restrict the performance of the primary guidance system and it must be at least an order of magnitude more reliable than the primary guidance system. Obviously something must be relinquished in order to achieve a more reliable system. In this case the capability for precise range control is lost but the system is still capable of gross range control.

The EMS is currently envisioned as consisting of four basic parts:

(1) An entry threshold indicator

(2) A corridor indicator

(3) A bank indicator

(4) A flight monitor

The entry threshold indicator is an on-off signal that is excited when the sensed acceleration is greater than some nominal value. The corridor indicator consists of two signals which are used to indicate whether the entry is at the top or the bottom of the corridor. The signals result from comparing the sensed acceleration to a nominal mid-corridor value at a discrete time interval after the threshold indicator is turned on. The back attitude indicator is a meter which indicates angular rotation about the approximate stability axes.

The flight monitor in the Apollo vehicle is a rectilinear plotter which presents a trace of the variation of total acceleration with the velocity of the vehicle. The astronaut compares this trace with information which is presented on the face of the plotter. Figure 34 shows a simple version of the information which is presented. Two families of curves are presented on the plotter.

One family serves to indicate if the acceleration limit is being exceeded and the second family is used to warn of an uncontrolled skip from the atmosphere. The high g lines are of interest only if the vehicle is not at aero bank angle. Whenever there is a danger of excessive accelerations the vehicle should be rolled to a zero bank angle.

The second family of curves are the most important since these are used to prevent an uncontrollable skip. For every combination of the acceleration and the velocity there exists a limiting rate at which safe atmospheric exits can be made. If the flight trace is compared to a set of rays which emanate from approximately zero g and an exit velocity which is less than the local circular orbital velocity, a safe limiting rate can be defined by tangency of the actual flight trace and the ray. Originally, these rays were straight lines such as shown in Figure 34, but it was found that there are times that the flight monitor would indicate a failure erroneously. This deficiency has been partially overcome by redefining the rays as curved lines and by shaping the entry trajectories so that a violation of the tangency criteria was more readily discernable.

LEM GUIDANCE AND CONTROL

Figure 35 shows the LEM guidance and control system. This is quite similar to the Command Module system, although the nomenclature is different. There are four major elements shown:

(1) Guidance and navigation system

(2) Stabilization and control system

(3) Reaction control system (attitude jets)

(4) Ascent and descent engines

The LEM G&N system has a landing radar and either a rendezvous radar or an optical tracker system for rendezvous purposes. The computer is the central data processor as shown. It receives data from the radar, DKSY, Alignment Optical Telescope (AOT), and the inertial measurement unit. Other inputs to the LEM guidance computer are from the attitude controller (which is the control stick used by the astronaut) and the throttle command. The computer drives either the ascent or the descent engines and the reaction control jets.

If the computer or the G&N system fails, the attitude controller and the throttle operate via the stabilization and control system to drive the ascent and descent engines. Either the stabilization and control system or the G&N system is used to provide attitude information on the FDAI (flight director attitude indicator) which is a gyro horizon. In the event the primary G&N system fails, the system also has an abort guidance system which is used to effect safe recovery into a rendezvous trajectory.

Figure 36 shows a more simplified block diagram of the LEM stabilization and control system showing the flow of attitude data, range data, timing, engine commands, and other information required for control, guidance and navigation.

Figure 37 is the current configuration of the LEM cockpit. The two windows, one for the command pilot and one for the pilot, can be seen to the right and left. The two FDAI's ("8-balls") are in the center of the instrument panel. Immediately above the FDAI's are pointers which indicate translational velocities, the delta V counter, throttle setting indicator, and propellant quantity gages. In the upper portion of the panel are main propulsion system and environmental control system control switches and indicators. Between the FDAI's are various subsystem control switch, vertical velocity indicator and thrust-to-weight ratio indicator.

The lower console in Figure 37 contains control switches and indicators for the stabilization and control system, power generation system, and cryogenic storage. At the bottom center of the lower console is the DSKY which provides a communications link between the astronaut and computer.

Let us look now at some of the gross features of the trajectory from lunar orbit to the lunar surface. This phase of the mission will be discussed in greater detail later in this symposium. Figure 38 shows this trajectory. The Command Module, Service Module, and LEM are initially in an 80-mile circular orbit around the Moon. The velocity of this orbit is approximately 6000 feet per second and the period is about two hours. This means that the spacecraft are hidden from the Earth about one hour out of every two hours. The operation of the onboard guidance and control system is important during these periods.

At such time as the vehicle reaches the proper point for lunar entry, an impulse of approximately 100 feet per second is applied by the descent engine. The descent trajectory is a Hohmann trajectory. After the vehicle has traversed 180° around the Moon, it reaches a perilune of about 50,000 feet. The trajectory is monitored by the astronauts in the LEM tracking the Command Module and by checking out the landing radar. It is also monitored by the astronaut in the Command Module who tracks the LEM through the Command Module optical system. Additionally, the earth-based ground tracking system tracks both vehicles.

The next phase of the landing maneuver is shown in Figure 39. This covers a distance of about 240 miles across the lunar surface. During this phase the vehicle proceeds from an altitude of 50,000 down to an altitude of about 10,000 feet. Here the LEM descent engine has been on for a period of about 450 seconds. After completion of this phase the vehicle enters a point known as "high-gate."

Figure 40 shows the "constant attitude" phase of the mission. This is the phase of the mission from about 10,000 feet down to approximately 500 to 1000 feet. During this phase of the mission the descent engine is throttle back to about 50% and the vehicle is pitched up into an attitude of about 35° to 45° with respect to the horizontal. During this phase of the mission the astronaut is surveying his landing site out the window.

One of the interesting problems associated with the lunar landing is that of visibility. If the sun angle is parallel to the trajectory path, i.e., if the sun is behind the astronaut, it tends to "wash out" the landing area such that he cannot discern characteristics of craters, etc. It is much like the reflection of the sun shining on the ocean. Therefore, the sun will be placed at such an angle as to be somewhat different from the angle of the landing trajectory or else the astronaut will be required to make a dog-leg into a landing to a site which he can survey carefully off to the side. This problem is discussed in greater detail in another paper in this symposium.

After the astronaut proceeds to the "low-gate" point at 700 to 1000 feet, he takes over control of the vehicle manually. He flies to an altitude of about 100 feet, at which time he nulls out all accumulated drifts in the platform with the aid of the landing radar. He is then in a position to make an instrument landing or a visual landing depending upon the degree of obscuration of his vision by a cloud of dust which might be created by his engine plume.

If the astronaut should land with too great a horizontal velocity or if he should begin to tip over into a crater, he can abort the mission as long as he acts before the vehicle exceeds an angle of about 45° measured with respect to the vertical. Similarly, if the bearing strength of the lunar surface is of such a nature as not to support the vehicle and he notices that the vehicle is sinking, he can press a button can abort the mission. Considerable redundancy and safety has been built into the vehicle characteristics and into the landing trajectory to provide for pilot safety.

RELIABILITY

In Apollo, two reliability criteria are applied in system design and in mission planning. One is associated with crew safety and the other is associated with mission success. Figure 41 shows the reliability diagram of the navigation and guidance system and the control system from a crew standpoint. In the navigation and guidance system, the onboard system and the ground system are in parallel paths. The ground system consists of the Mission Control Center, communications systems, and the tracking network. The onboard system consists of the Apollo Guidance Computer, the optics telescope, and the inertial measurement unit.

In the control system there is even greater redundancy from the crew safety standpoint. The computer and inertial measurement unit are connected in parallel with the body-mounted attitude gyros and accelerometers. Also in parallel are the sextant and scanning telescope operating automatically or manually and dual thrust-vector control electronics and dual reaction control jet electronics.

The mission criteria for crew safety is such that if sufficient equipment malfunctions have occurred that one more malfunction of any kind would endanger the crew, then at that point the mission should be aborted. The realiability numbers are such that we can expect potential danger to one flight crew in 200 missions.

Figure 42 shows the mission success block diagram. In this block diagram, there are series connections of the guidance and control system rather than the parallel paths shown in the crew safety diagram. The probability of mission success, of course, is somewhat less than that for crew safety.

QUESTIONS AND ANSWERS

Speaker: Dr. Robert C. Duncan

No questions

NASA-S-64-6589

GENERALIZED GUIDANCE AND CONTROL SYSTEM

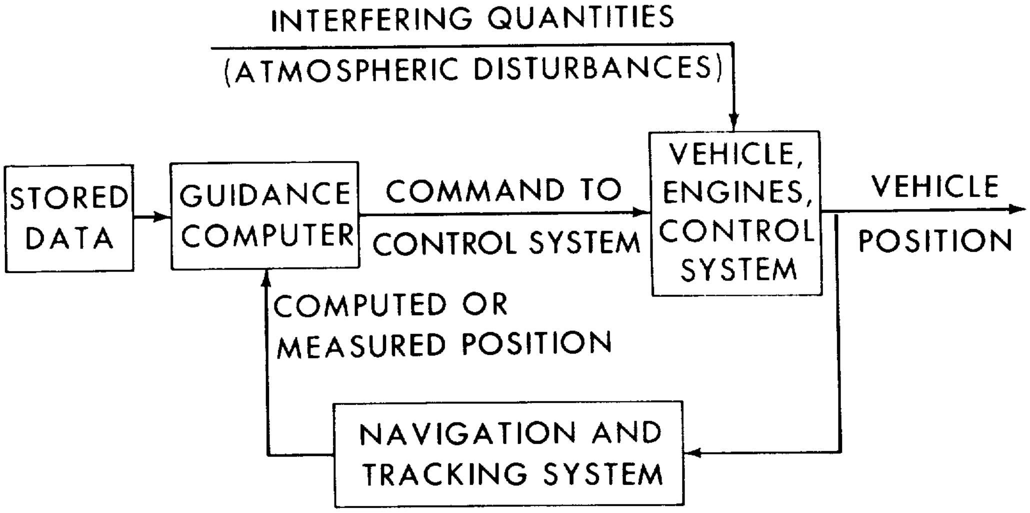

INTERFERING QUANTITIES
(ATMOSPHERIC DISTURBANCES)
STORED DATA
GUIDANCE COMPUTER
COMMAND TO CONTROL SYSTEM
VEHICLE, ENGINES, CONTROL SYSTEM
VEHICLE POSITION
COMPUTED OR MEASURED POSITION
NAVIGATION AND TRACKING SYSTEM

FIGURE 1

NASA-S-65-10001

EARLY PROGRAM GUIDELINES

- GROUND NAVIGATION CAPABILITY MANDATORY REQUIREMENT
- SELF-CONTAINED ON-BOARD NAVIGATION CAPABILITY MANDATORY REQUIREMENT
- ON-BOARD SYSTEM DESIGN CONCEPT TAKE MAXIMUM ADVANTAGE OF GROUND CAPABILITIES AND INCLUDE NECESSARY INTERFACES

FIGURE 2

NASA-S-65-4483

BLOCK II
N, G, AND C CONFIGURATION

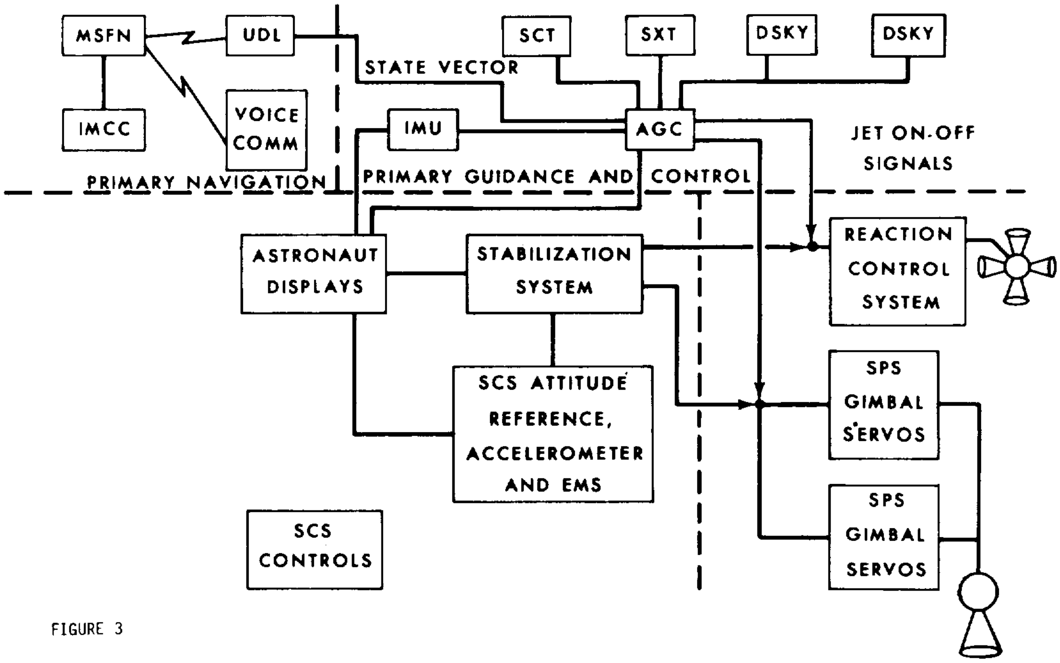

MSFN
UDL
VOICE COMM
IMCC
STATE VECTOR
SCT
SXT
DSKY
DSKY
IMU
AGC
JET ON-OFF SIGNALS
PRIMARY NAVIGATION
PRIMARY GUIDANCE AND CONTROL
ASTRONAUT DISPLAYS
STABILIZATION SYSTEM
REACTION CONTROL SYSTEM
SCS ATTITUDE REFERENCE, ACCELEROMETER AND EMS
SPS GIMBAL SERVOS
SPS GIMBAL SERVOS
SCS CONTROLS

FIGURE 3

NASA S-63-10214

SCS BLOCK DIAGRAM

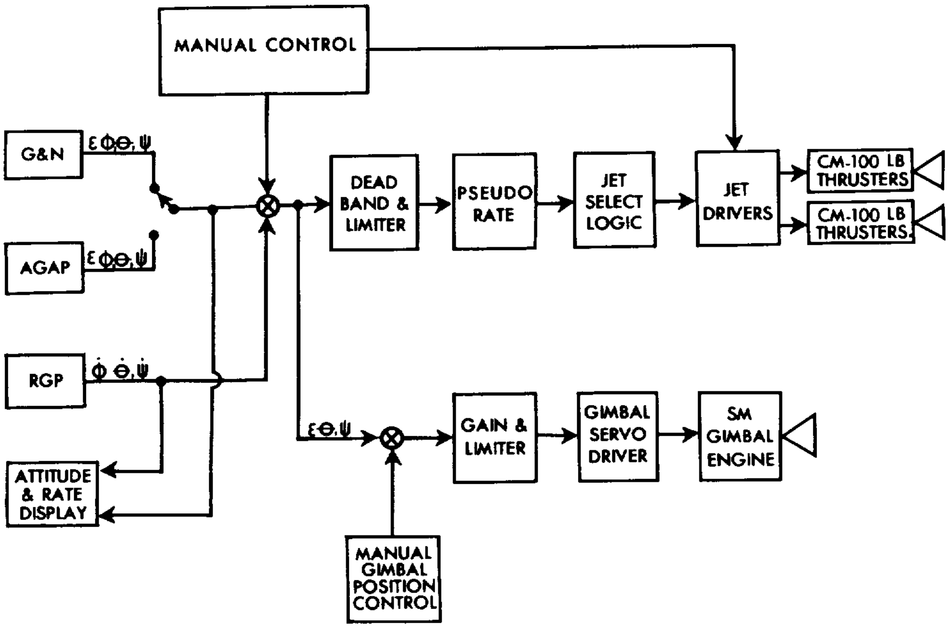

MANUAL CONTROL
G&N
$\varepsilon\phi,\theta,\psi$
AGAP
$\varepsilon\phi,\theta,\psi$
RGP
$\dot{\phi}\ \dot{\theta},\dot{\psi}$
ATTITUDE & RATE DISPLAY
DEAD BAND & LIMITER
PSEUDO RATE
JET SELECT LOGIC
JET DRIVERS
CM-100 LB THRUSTERS
CM-100 LB THRUSTERS
$\varepsilon\theta,\psi$
GAIN & LIMITER
GIMBAL SERVO DRIVER
SM GIMBAL ENGINE
MANUAL GIMBAL POSITION CONTROL

FIGURE 4

NASA-S-64-6653

GUIDANCE & NAVIGATION SPACECRAFT LOCATION

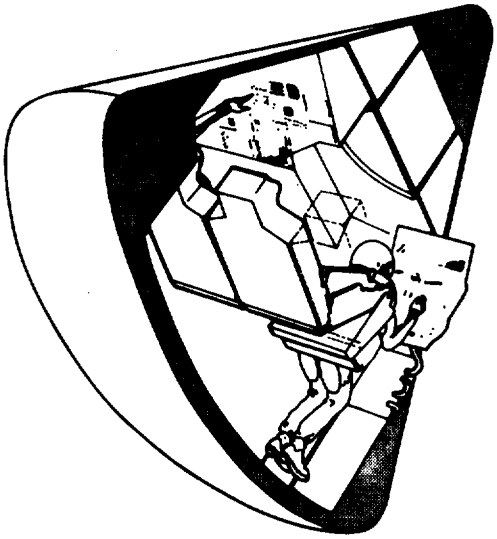
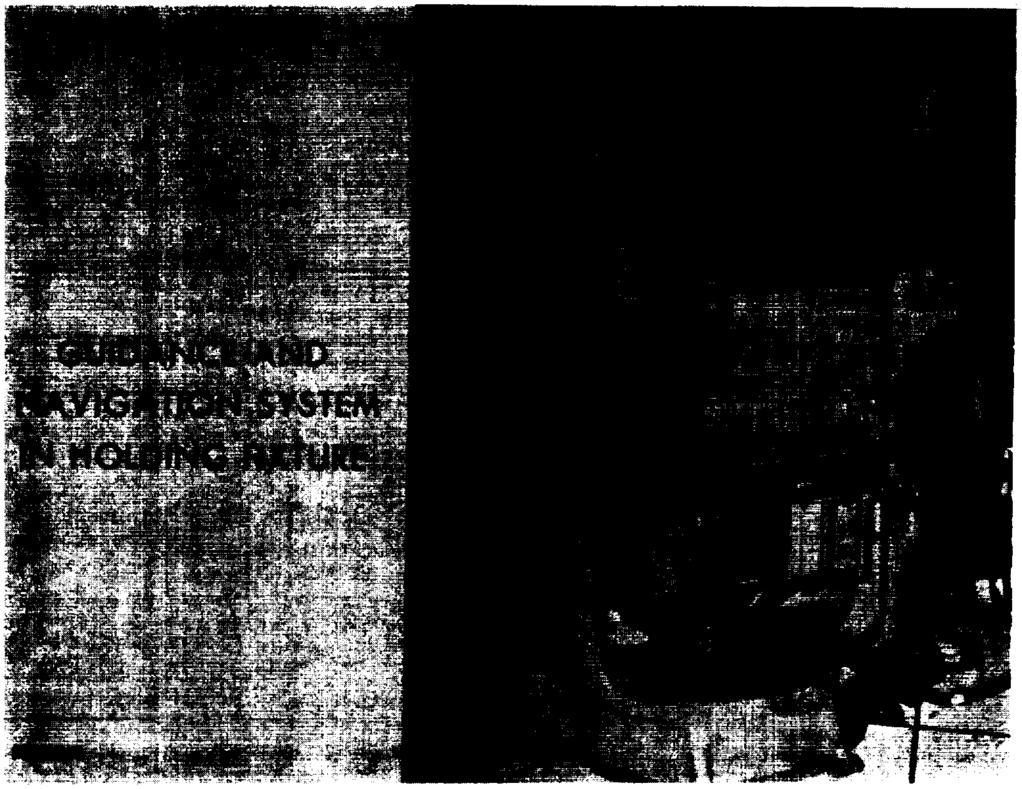


FIGURE 5



GUIDANCE AND
NAVIGATION SYSTEM
IN HOLDING FIXTURE

NASA S-63-10217

THREE DEGREE OF FREEDOM PLATFORM

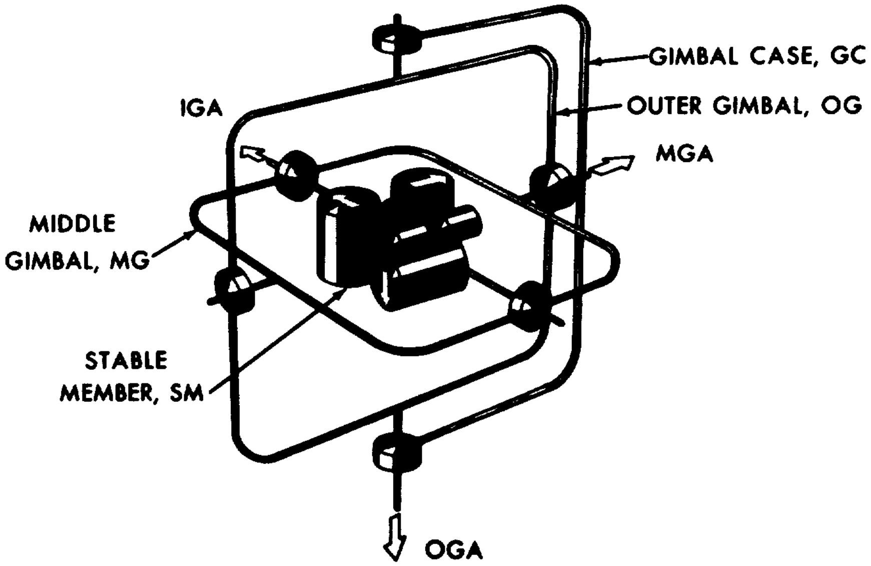

GIMBAL CASE, GC
OUTER GIMBAL, OG
IGA
MGA
MIDDLE GIMBAL, MG
STABLE MEMBER, SM
OGA

FIGURE 7

NASA-S-65-3973

APOLLO
INNER, MIDDLE & OUTER GIMBAL ASSEMBLIES
IMU-5 FOR APOLLO G&N EQUIPMENT

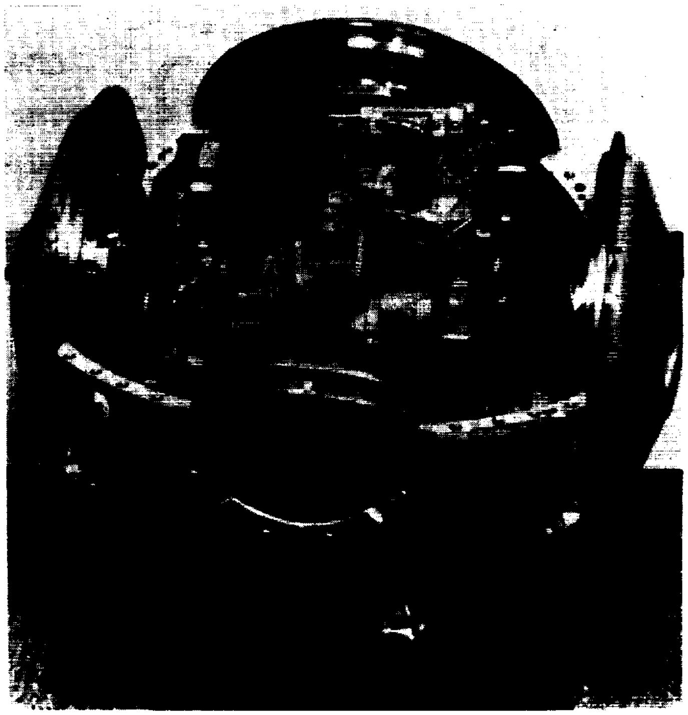


FIGURE 9

NASA-S-65-3978

THE IMU, NAVIGATION BASE AND OPTICAL SUBSYSTEM

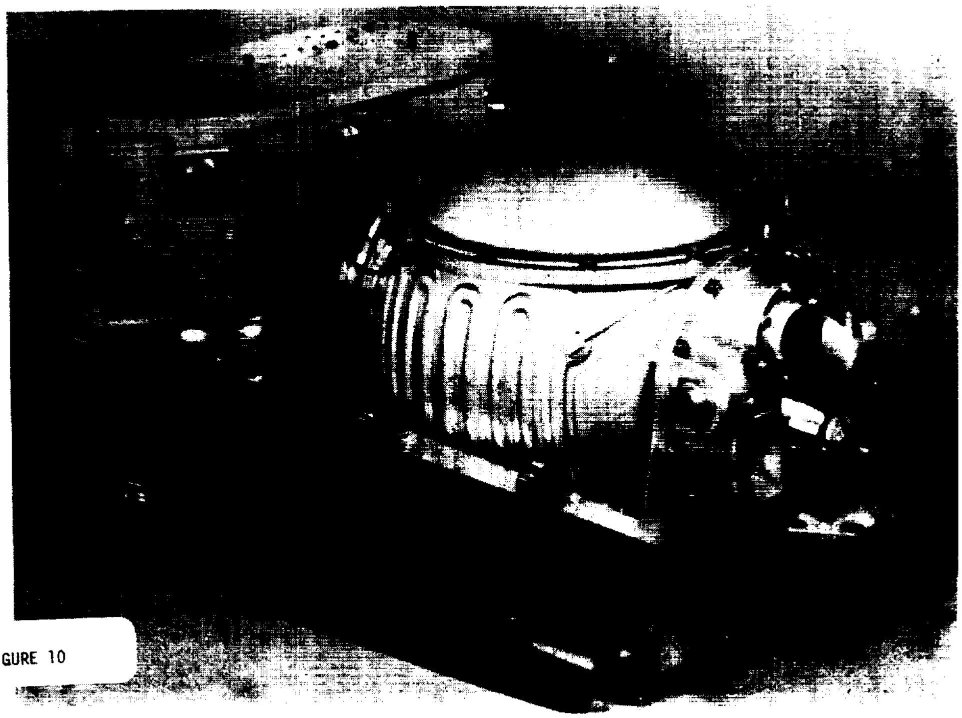


FIGURE 10

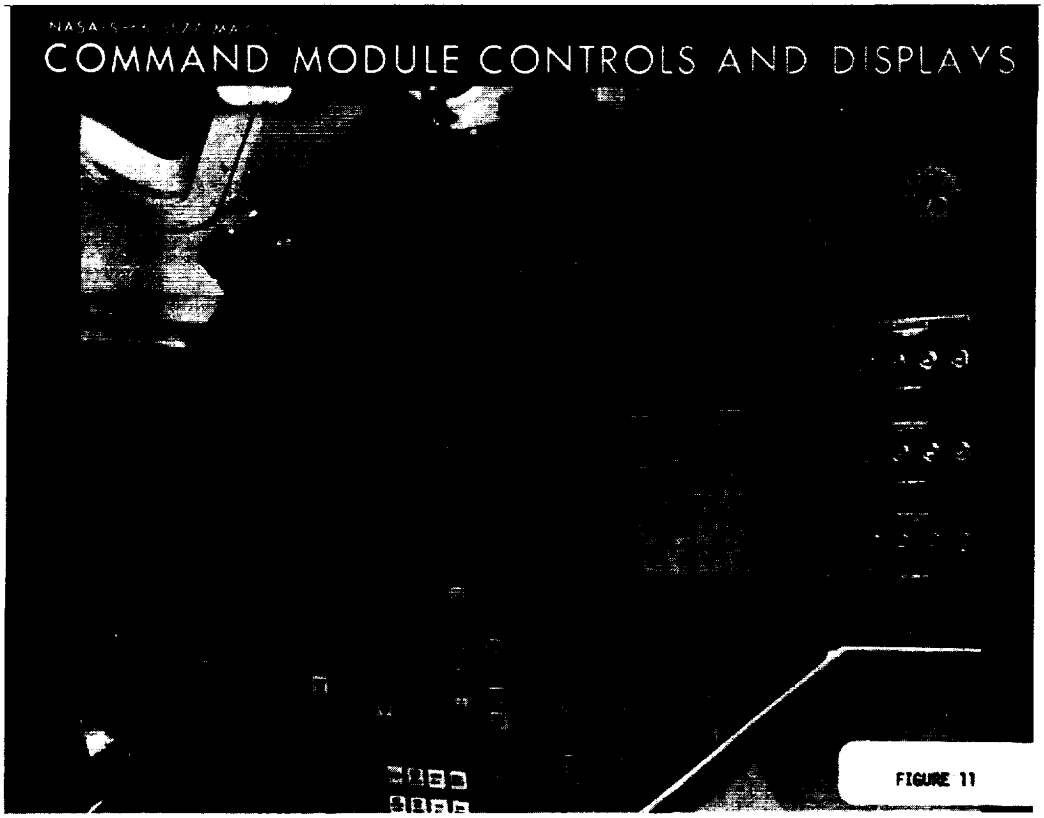

COMMAND MODULE CONTROLS AND DISPLAYS

FIGURE 11

NASA-S-64-6647

AGC DSKY

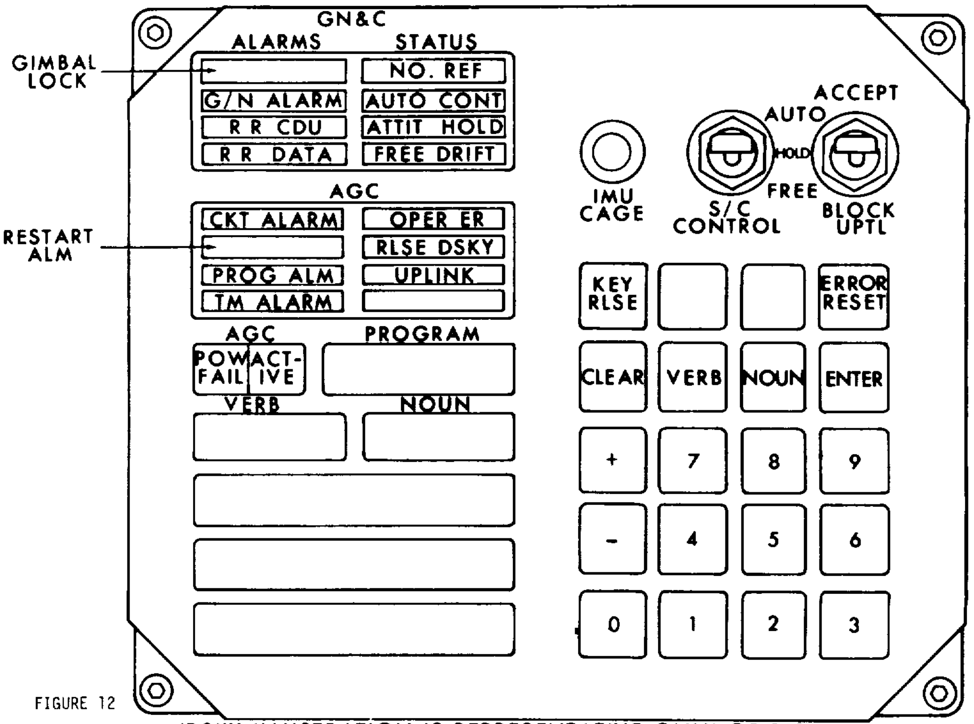

GN&C
ALARMS
STATUS
GIMBAL LOCK
NO. REF
G/N ALARM
AUTO CONT
R R CDU
ATTIT HOLD
R R DATA
FREE DRIFT
AGC
CKT ALARM
OPER ER
RESTART ALM
RLSE DSKY
PROG ALM
UPLINK
TM ALARM
AGC
POW FAIL
ACT-IVE
PROGRAM
VERB
NOUN
ACCEPT
AUTO
HOLD
FREE
IMU CAGE
S/C CONTROL
BLOCK UPTL
KEY RLSE
ERROR RESET
CLEAR
VERB
NOUN
ENTER
+
7
8
9
-
4
5
6
0
1
2
3

FIGURE 12

(DSKY ILLUSTRATION IS REPRESENTATIVE ONLY OF DISPLAY DATA AND CONTROL FUNCTIONS, NOT PHYSICAL CONFIGURATION AND ARRANGEMENT)

NASA-S-65-10004

MSFN TRACKING SYSTEM

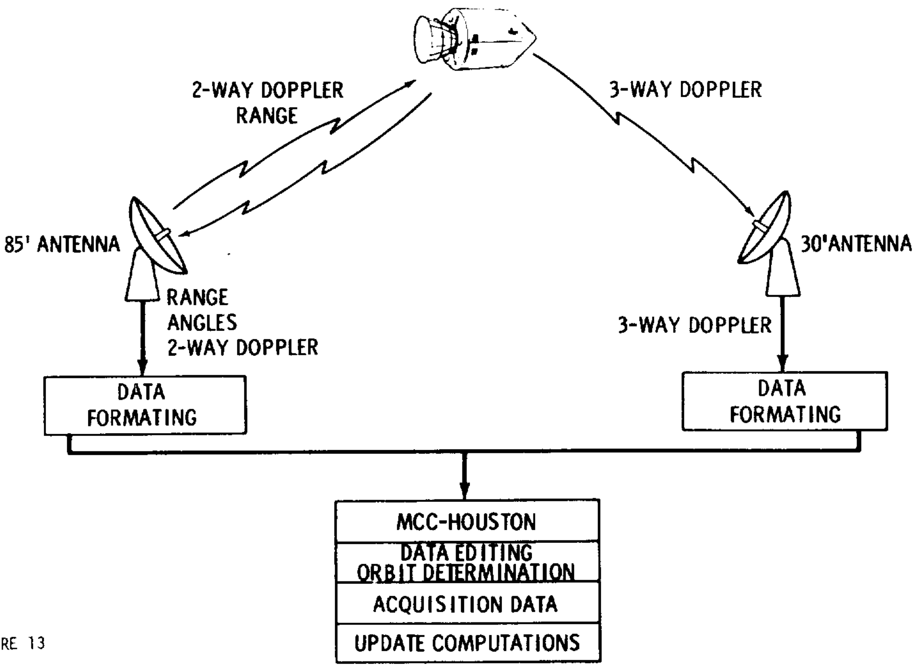
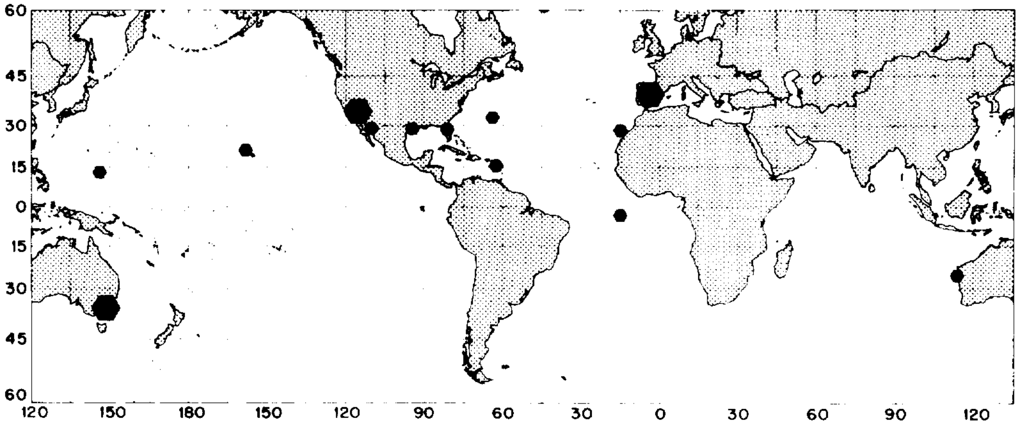

2-WAY DOPPLER
RANGE
3-WAY DOPPLER
85' ANTENNA
30'ANTENNA
RANGE
ANGLES
2-WAY DOPPLER
3-WAY DOPPLER
DATA FORMATING
DATA FORMATING
MCC-HOUSTON
DATA EDITING
ORBIT DETERMINATION
ACQUISITION DATA
UPDATE COMPUTATIONS

FIGURE 13

NASA-S-65-10011

MSFN UNIFIED S-BAND STATION SITES


60
45
30
15
0
15
30
45
60
120
150
180
150
120
90
60
30
0
30
60
90
120

DUAL SITES
MESSAGE SHARING
MULTIPLE SITE COVERAGE
OVERLAP OF COVERAGE BY 85 FT ANTENNAS

⬢ 85 FT ANTENNA
● 30 FT ANTENNA

FIGURE 14

NASA-S-65-9997

STATION CHARACTERISTICS

- TRACKING ACCURACIES (ONE SIGMA)

MODE	NOISE	BIAS
RANGE	60 FT	120 FT
ANGLES	0.8 MR	1.6 MR
TWO-WAY DOPPLER	0.1 FPS	0.07 FPS
THREE-WAY DOPPLER	0.1 FPS	0.2 FPS

- FREQUENCY REFERENCE
  - RUBIDIUM FREQUENCY STANDARD
  - STABILITY (LONG AND SHORT TERM) - $5 \times 10^{-11} \frac{\text{PARTS}}{\text{PARTS}}$
- OPERATING CAPABILITIES
  - MSFN STATIONS WITH 30 OR 85 FOOT ANTENNAE CAN TWO-WAY DOPPLER TRACK WITH SPACECRAFT OMNIDIRECTIONAL OR HI-GAIN ANTENNAE AT LUNAR DISTANCES

FIGURE 15

NASA-S-65-10003

MISSION OPERATIONS

- MSFN TRACKING MODES

PHASE	SYSTEM	MEASURABLE
EARTH ORBIT	C-BAND	RANGE, 2 ANGLES
CISLUNAR	USBS	3-WAY DOPPLER AND RANGE, 2 ANGLES
LUNAR ORBIT	USBS	3-WAY DOPPLER

- DATA FLOW
  - ON-BOARD NAVIGATION DATA TELEMETERED TO MSFN
  - RADAR TRACKING AND TELEMETERED DATA PIPED INTO IMCC
  - GUIDANCE AND NAVIGATION PARAMETERS COMPUTED IN RTCC
  - MSFN GUIDANCE AND NAVIGATION DATA TELEMETERED TO ON-BOARD COMPUTER

FIGURE 16

NASA-S-65-9996

ASSUMPTIONS
MSFN NAVIGATION ACCURACY

- MSFN OPERATING IN 3-WAY DOPPLER MODE WITH ONE STATION TRANSMITTING AND TWO RECEIVING
- MEASUREMENT UNCERTAINTIES (ONE SIGMA)

	NOISE	BIAS
● 2-WAY DOPPLER	0.1 FPS	0.07 FPS
● 3-WAY DOPPLER	0.1 FPS	0.2 FPS

- STATION LOCATION UNCERTAINTIES (ONE SIGMA)

● LATITUDE	1 - 6 SEC
● LONGITUDE	1 - 6 SEC
● ALTITUDE	100 - 200 FT

- GRAVITATIONAL CONSTANT UNCERTAINTIES (ONE SIGMA)

● EARTH	106×10^9 FT^3 / SEC^2
● MOON	6×10^9 FT^3 / SEC^2

- PARAMETERS ESTIMATED IN ORBIT DETERMINATION PROGRAM
  - THREE COMPONENTS OF VELOCITY
  - THREE COMPONENTS OF POSITION
  - TWO 3-WAY DOPPLER BIASES

FIGURE 17

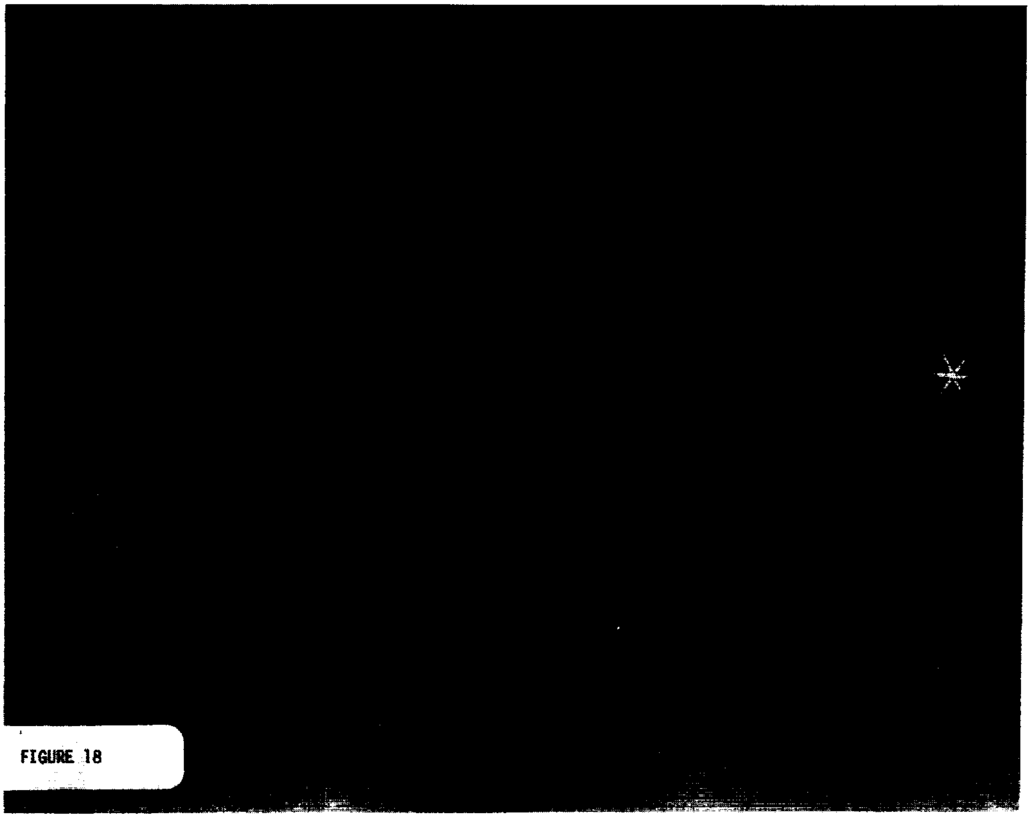


FIGURE 18

NASA-S-66-3569 MAY 12

APOLLO
GEOMETRY OF NAVIGATIONAL FIX IN SPACE

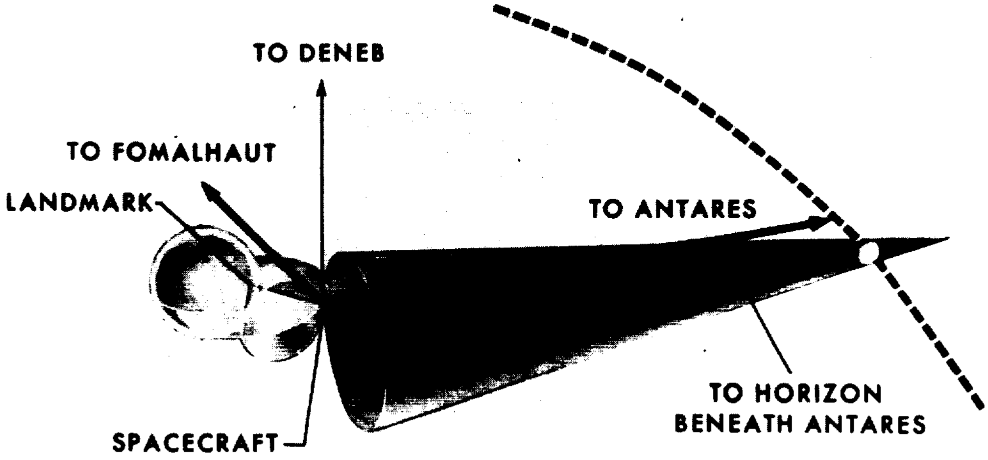

TO DENEB
TO FOMALHAUT
LANDMARK
TO ANTARES
SPACECRAFT
TO HORIZON BENEATH ANTARES

FIGURE 19

NASA-S-66-3570 MAY 12

APOLLO
ILLUMINATED EARTH HORIZON NAVIGATION REFERENCE

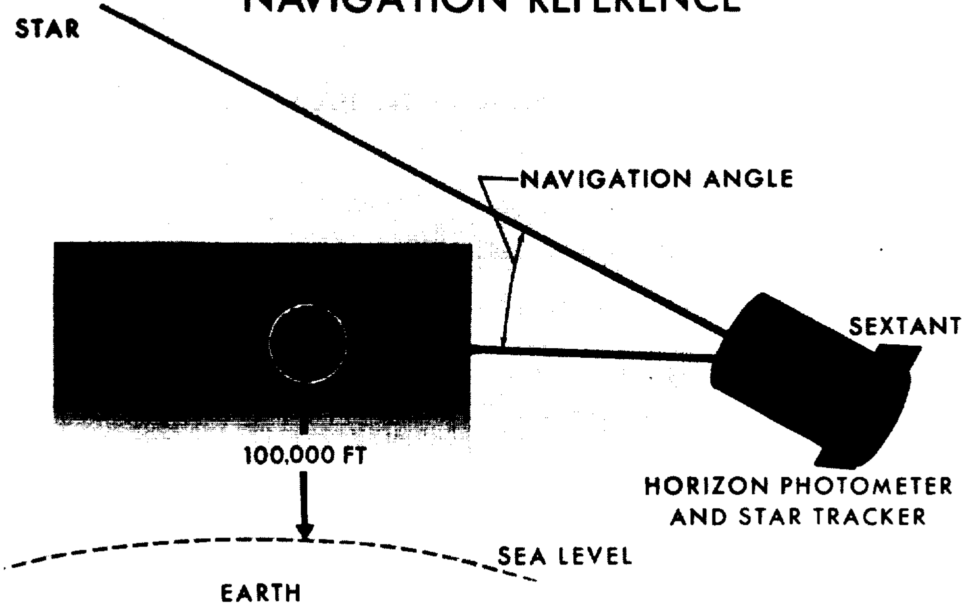

STAR
NAVIGATION ANGLE
SEXTANT
100,000 FT
HORIZON PHOTOMETER AND STAR TRACKER
SEA LEVEL
EARTH

FIGURE 20

NASA-S-65-9998

ASSUMPTIONS
BLOCK II G&C SYSTEM

- EARTH ORBIT PHASE
  - SCANNING TELESCOPE SIGHTING UNCERTAINTY - 4 MR
  - LANDMARK POSITION UNCERTAINTY - 1000 FT
  - SEVEN LANDMARK SIGHTINGS TOTAL
- TRANSLUNAR / TRANSEARTH PHASES
  - SEXTANT UNCERTAINTY - 10 ARC SECONDS
  - HORIZON UNCERTAINTIES - 1 NM (EARTH)
    .5 NM (MOON)
  - 40 LANDMARK SIGHTINGS EACH WAY
  - MIDCOURSE VELOCITY CORRECTION UNCERTAINTIES
    - MAGNITUDE - 1 PER CENT
    - POINTING - 10 MR

FIGURE 21

NASA-S-65-10000

ASSUMPTIONS (CONTINUED)

- LUNAR ORBIT PHASE
  - SCANNING TELESCOPE SIGHTING UNCERTAINTY - 4 MR
- LANDMARK POSITION UNCERTAINTIES 1000 - 5500 FT (HORIZONTAL)
  2300 - 3000 FT (VERTICAL)
- FIVE LANDMARKS ON EARTHSIDE CHOSEN ALONG TRACK
- THREE SIGHTINGS PER LANDMARK ON EACH FIRST TWO ORBITS, PERIODIC UPDATES THEREAFTER

FIGURE 22

NASA-S-65-10006

COMPARISON OF MSFN/ON-BOARD NAVIGATIONAL UNCERTAINTIES

PHASE	EVENT	RMS POSITION, NM		RMS VELOCITY, FPS	
		MSFN	ON-BOARD	MSFN	ON-BOARD
TRANSLUNAR	INJECTION (LAND)	0.1	2.2	1.5	13.5
	(SHIP)	2.7		37.1	
	FIRST MIDCOURSE	1.2	5.4	0.7	5.2
	SECOND MIDCOURSE	10.3	8.4	0.2	0.4
	THIRD MIDCOURSE	3.1	2.5	0.8	1.1
	PERILUNE	1.2	1.2	10.2	5.0
LUNAR ORBIT	LEM SEPARATION	0.4	0.9	3.2	2.8
	LEM RENDEZVOUS	0.2	0.3	0.6	1.2
TRANSEARTH	INJECTION	0.4	0.4	1.0	1.8
	FIRST MIDCOURSE	6.0	4.4	0.3	0.7
	SECOND MIDCOURSE	4.8	7.1	0.2	0.6
	THIRD MIDCOURSE	1.4	8.7	1.1	6.5
	ENTRY	0.5	8.7	2.2	45.3

FIGURE 23

NASA-S-65-10032

NAVIGATION UNCERTAINTIES

TRANSLUNAR PHASE

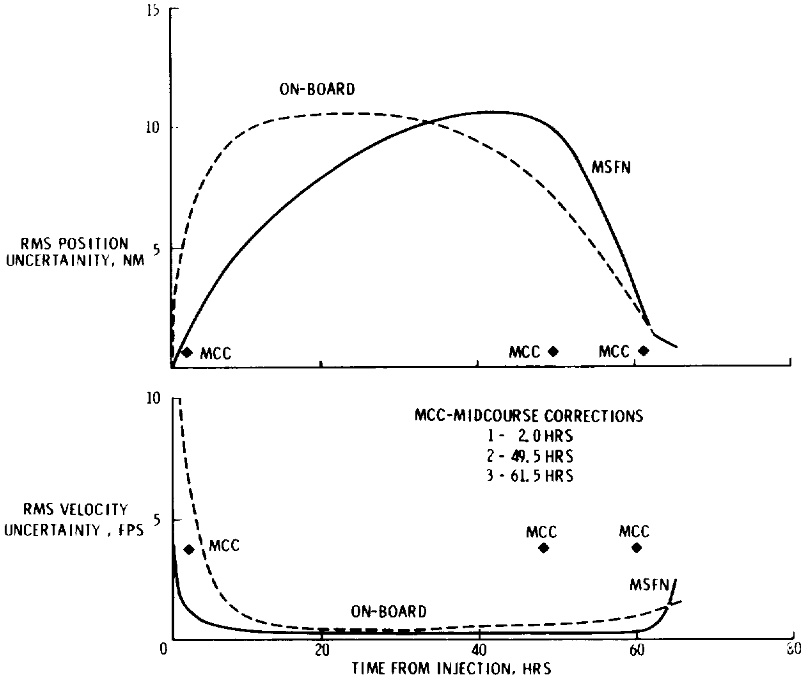


FIGURE 24

NASA-S-66-7324 JUL

NAVIGATION UNCERTAINTIES

LUNAR ORBIT PHASE

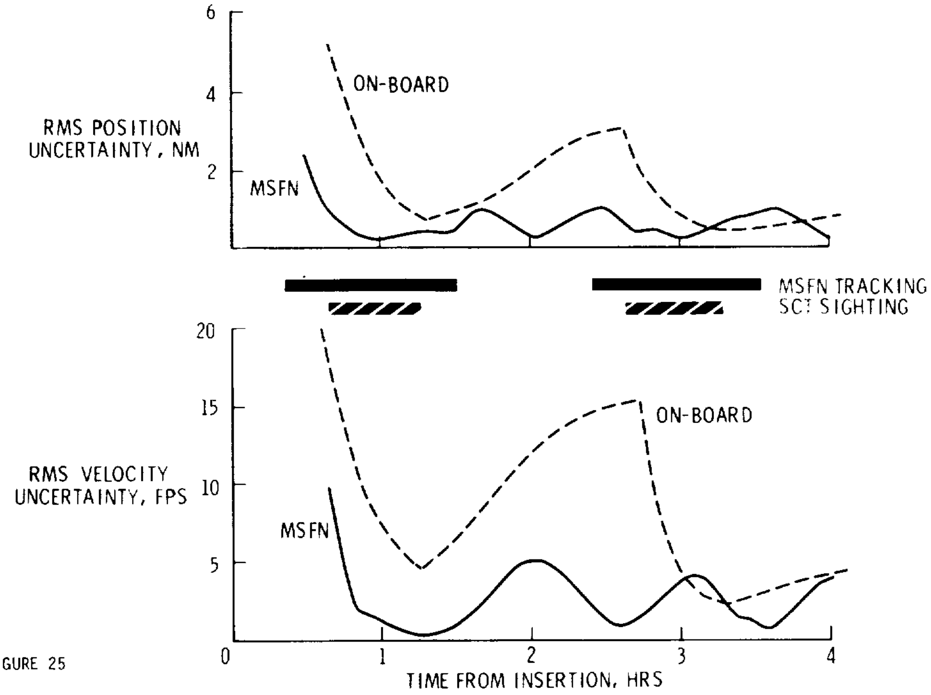


FIGURE 25

NASA-S-65-10034

MIDCOURSE VELOCITY CORRECTION PERFORMANCE

TRANSLUNAR PHASE

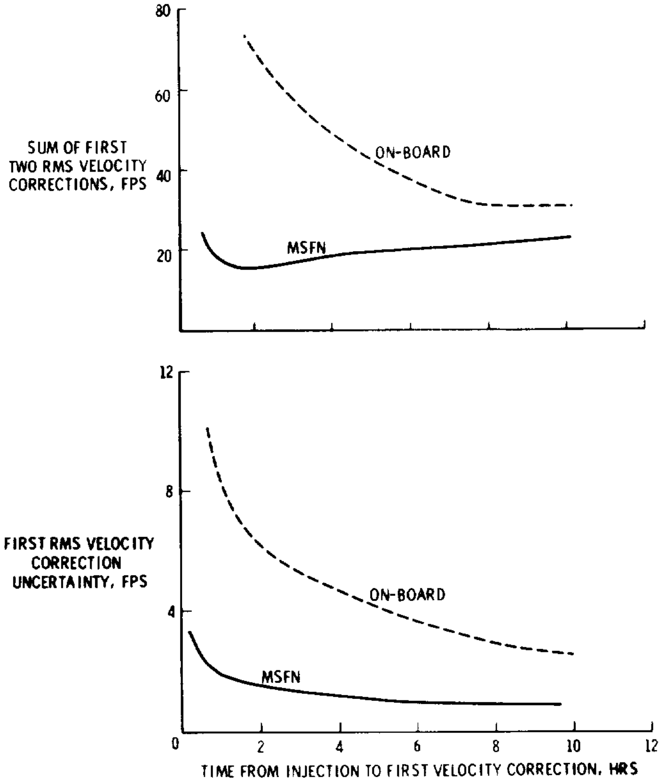
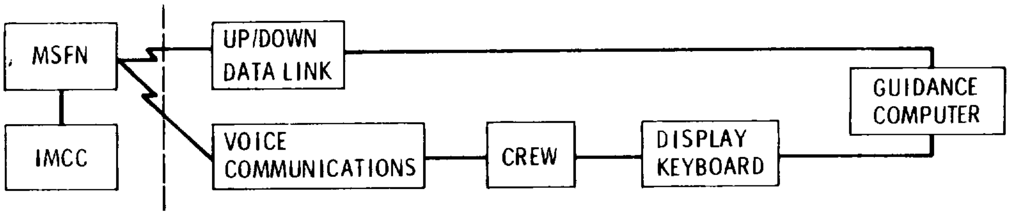


FIGURE 26

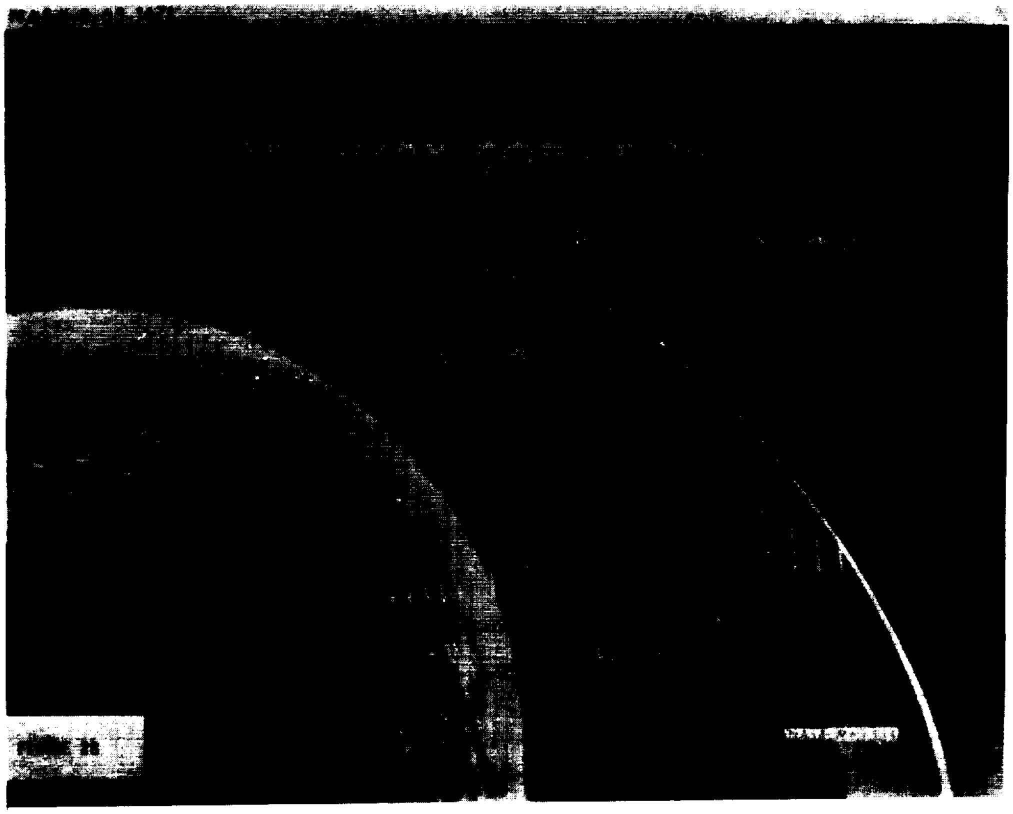
NASA-S-[illegible]-10007

DESCRIPTION OF ON-BOARD/MSFN NAVIGATION INTERFACE


MSFN
IMCC
UP/DOWN DATA LINK
VOICE COMMUNICATIONS
CREW
DISPLAY KEYBOARD
GUIDANCE COMPUTER

- COMPUTER UPDATE RATE - 10 UPLINK WORDS/SEC
- NAVIGATION UPDATE INFORMATION REQUIRED
  - STATE VECTOR - 6 COMPONENTS
  - TIME
  - AIM POINTS
  - COMPUTER TIME ALINE
- UPDATE FREQUENCY REQUIRED - ONCE PRIOR TO EACH GUIDANCE MANEUVER
- TRANSMISSION TIME
  - 60 SECONDS MAXIMUM FOR 99% PROBABILITY OF NO ERRORS
  - NAVIGATION UPDATE NOT TIME CRITICAL

FIGURE 27



NASA-S-65-3769

FACTORS WHICH INFLUENCE DESIGN OF ENTRY GUIDANCE

REQUIREMENTS FOR CREW SAFETY

ENTRY VELOCITY AND ANGLE

PHYSICAL PROPERTIES OF VEHICLE

REQUIRED RANGE CONTROL

PHYSICAL QUANTITIES MEASURED

COMPUTER SIZE AND SPEED

ENTRY MONITORING SYSTEM

FIGURE 29

NASA-S-65-3414

CLASSIFICATION OF AERODYNAMIC PROPERTIES OF ENTRY VEHICLES

L/D	W/C_DS	EXAMPLES
0	CONSTANT	MERCURY
0	VARIABLE	
CONSTANT	CONSTANT	GEMINI, APOLLO
VARIABLE	VARIABLE	

FIGURE 30

NASA-S-66-3574 MAY 12

ERRORS IN LANDING LOCATION OF MANNED GEMINI MISSIONS

MISSION	ASTRONAUTS	LANDING ERROR, N MI
GEMINI II	(UNMANNED)	18
GEMINI III	GRISSOM YOUNG	64
GEMINI IV	MCDIVITT WHITE	47
GEMINI V	COOPER CONRAD	97
GEMINI VI	SCHIRRA STAFFORD	7
GEMINI VII	BORMAN LOVELL	7
GEMINI VIII	ARMSTRONG SCOTT	3

FIGURE 31

NASA-S-65-1414

TYPICAL ENTRY TRAJECTORY

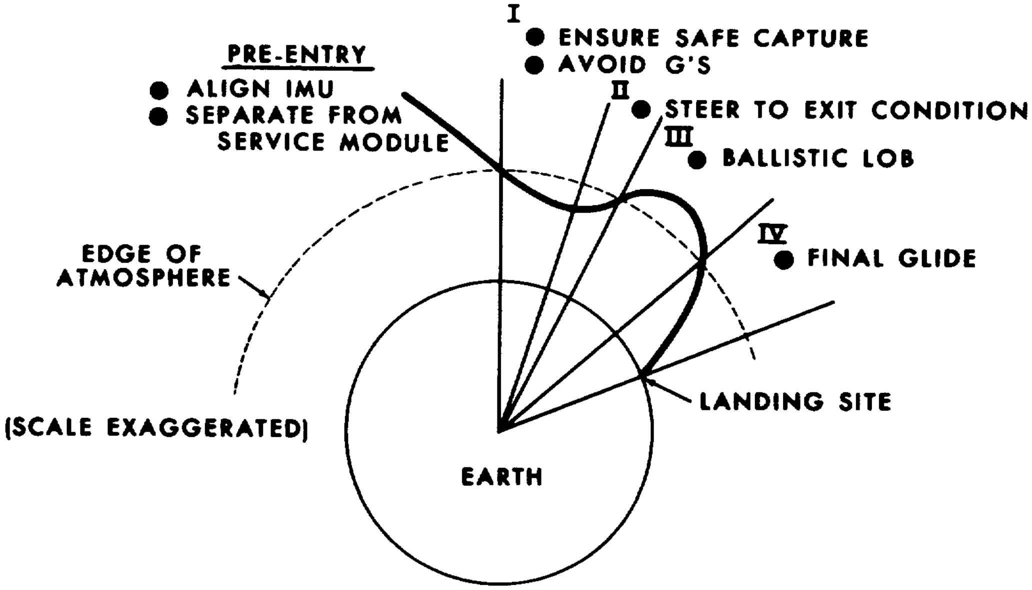

I
● ENSURE SAFE CAPTURE
● AVOID G'S
PRE-ENTRY
● ALIGN IMU
● SEPARATE FROM SERVICE MODULE
II
● STEER TO EXIT CONDITION
III
● BALLISTIC LOB
EDGE OF ATMOSPHERE
IV
● FINAL GLIDE
LANDING SITE
(SCALE EXAGGERATED)
EARTH

FIGURE 32

NASA-S-65-1413

ENTRY STEERING FLOW CHART

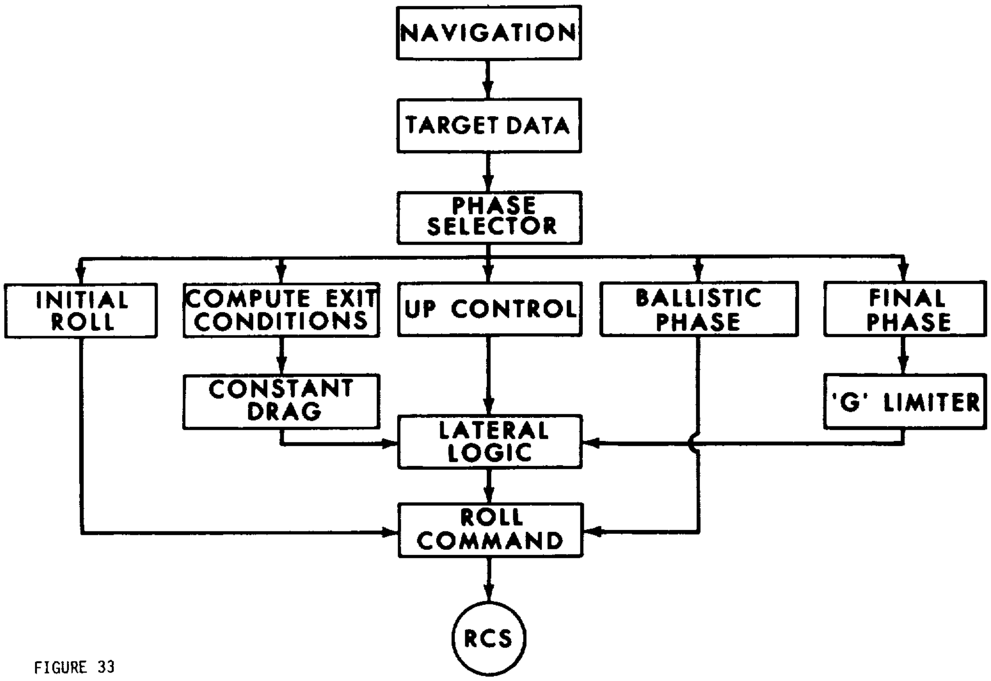

NAVIGATION
TARGET DATA
PHASE SELECTOR
INITIAL ROLL
COMPUTE EXIT CONDITIONS
UP CONTROL
BALLISTIC PHASE
FINAL PHASE
CONSTANT DRAG
'G' LIMITER
LATERAL LOGIC
ROLL COMMAND
RCS

FIGURE 33

NASA-S-65-3768

ENTRY FLIGHT MONITOR

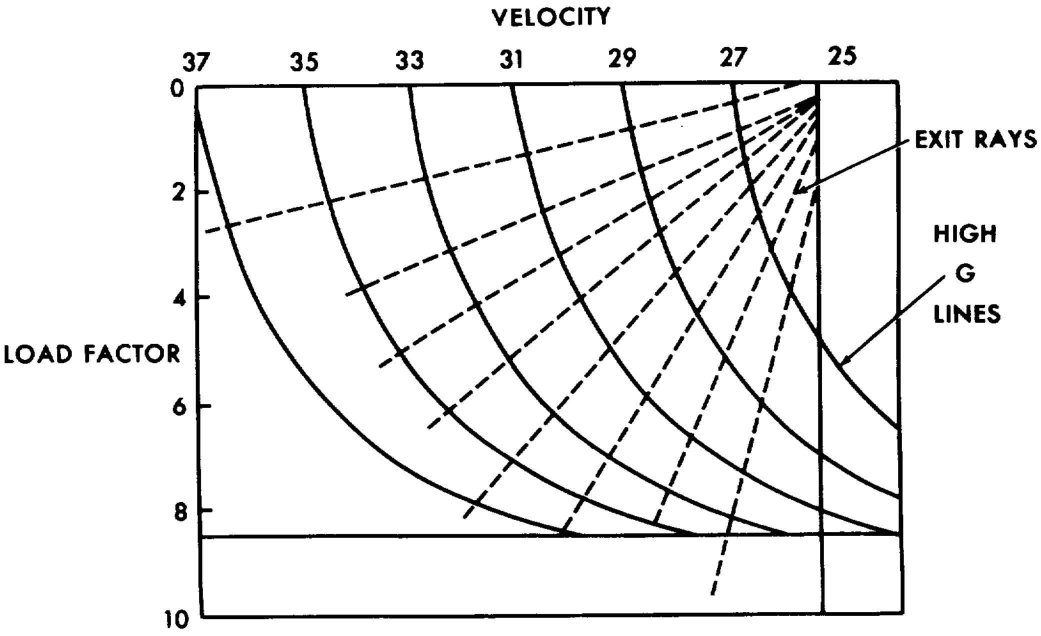


FIGURE 34

NASA-S-65-3793

LEM INTEGRATED GUIDANCE AND CONTROL SYSTEM CONFIGURATION

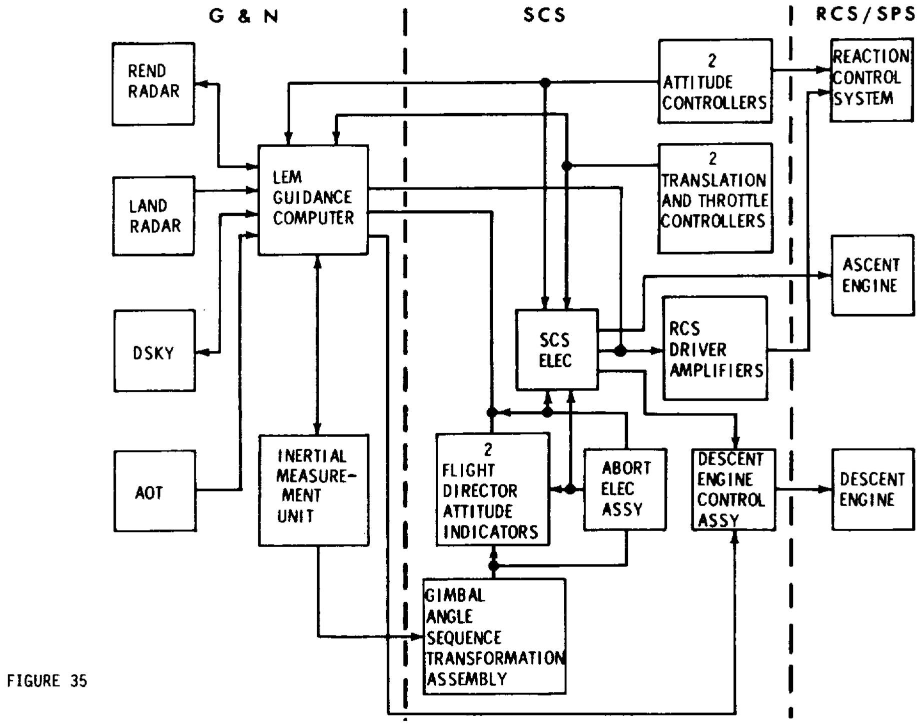

G & N
SCS
RCS/SPS
REND RADAR
LAND RADAR
DSKY
AOT
LEM GUIDANCE COMPUTER
INERTIAL MEASUREMENT UNIT
2 ATTITUDE CONTROLLERS
2 TRANSLATION AND THROTTLE CONTROLLERS
SCS ELEC
RCS DRIVER AMPLIFIERS
2 FLIGHT DIRECTOR ATTITUDE INDICATORS
ABORT ELEC ASSY
DESCENT ENGINE CONTROL ASSY
GIMBAL ANGLE SEQUENCE TRANSFORMATION ASSEMBLY
REACTION CONTROL SYSTEM
ASCENT ENGINE
DESCENT ENGINE

FIGURE 35

NASA-S-65-776

LEM STABILIZATION AND CONTROL SYSTEM BLOCK DIAGRAM

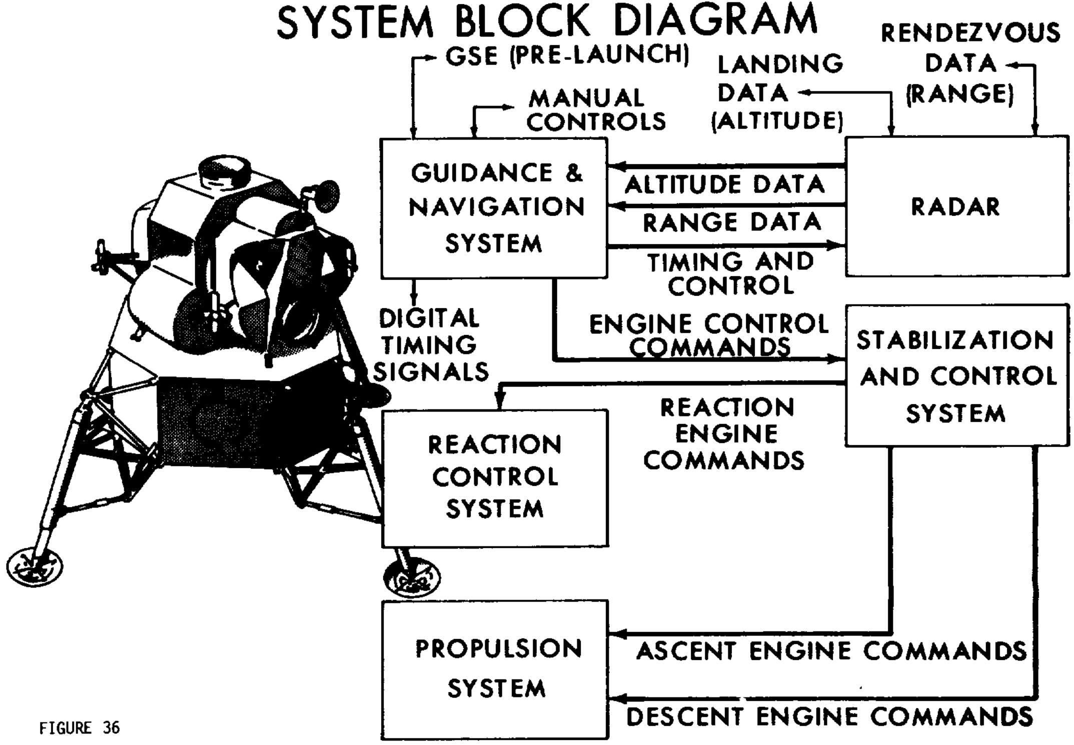

RENDEZVOUS DATA (RANGE)
GSE (PRE-LAUNCH)
LANDING DATA (ALTITUDE)
MANUAL CONTROLS
GUIDANCE & NAVIGATION SYSTEM
ALTITUDE DATA
RANGE DATA
TIMING AND CONTROL
RADAR
DIGITAL TIMING SIGNALS
ENGINE CONTROL COMMANDS
STABILIZATION AND CONTROL SYSTEM
REACTION ENGINE COMMANDS
REACTION CONTROL SYSTEM
PROPULSION SYSTEM
ASCENT ENGINE COMMANDS
DESCENT ENGINE COMMANDS

FIGURE 36

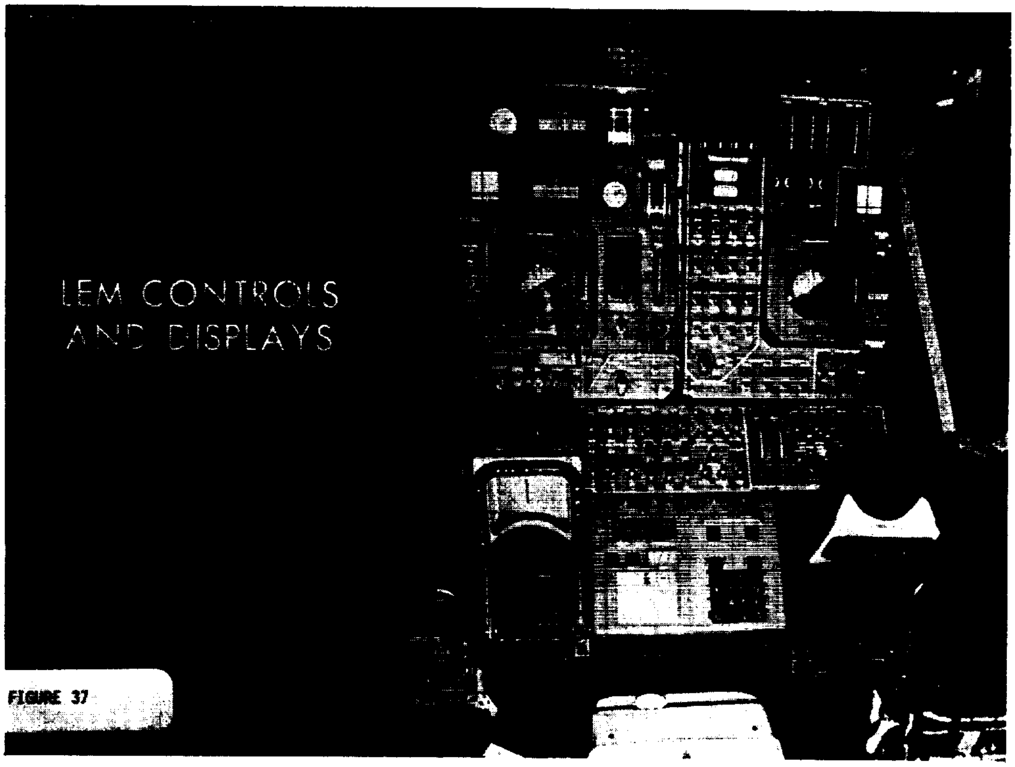

LEM CONTROLS
AND DISPLAYS

FIGURE 37

NASA-S-65-3982

LEM HOHMANN DESCENT ORBIT

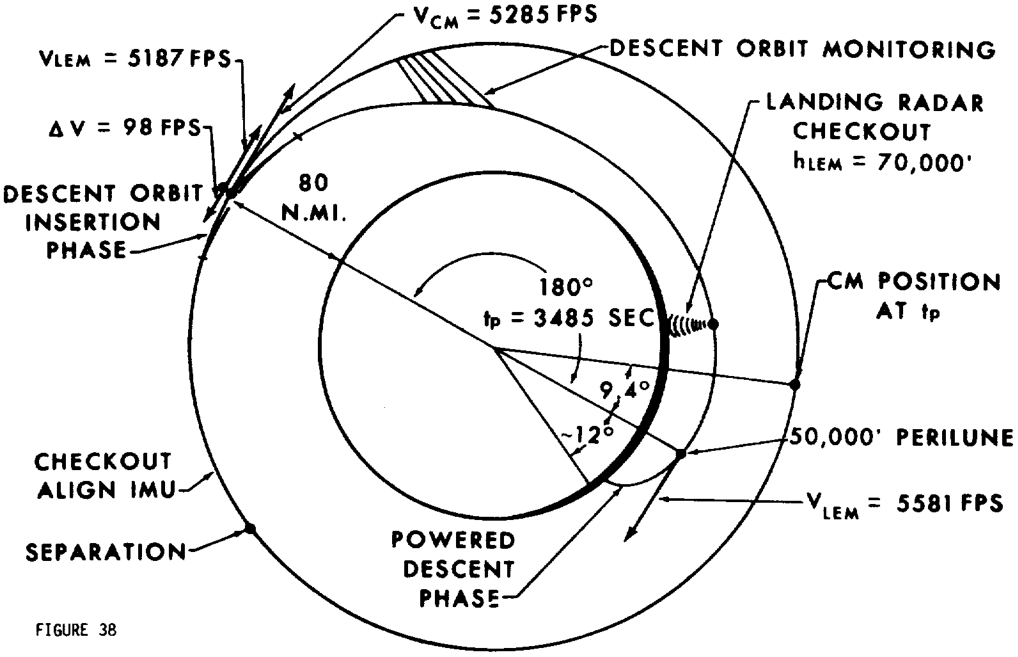

V_{CM} = 5285 FPS
DESCENT ORBIT MONITORING
V_{LEM} = 5187 FPS
LANDING RADAR CHECKOUT
h_{LEM} = 70,000'
Δ V = 98 FPS
80 N.MI.
DESCENT ORBIT INSERTION PHASE
180°
t_p = 3485 SEC
CM POSITION AT t_p
9.4°
~12°
50,000' PERILUNE
CHECKOUT ALIGN IMU
V_{LEM} = 5581 FPS
SEPARATION
POWERED DESCENT PHASE

FIGURE 38

NASA-S-66-3573 MAY 12

LUNAR LANDING MANEUVER PHASES

EXAGGERATED SCALE

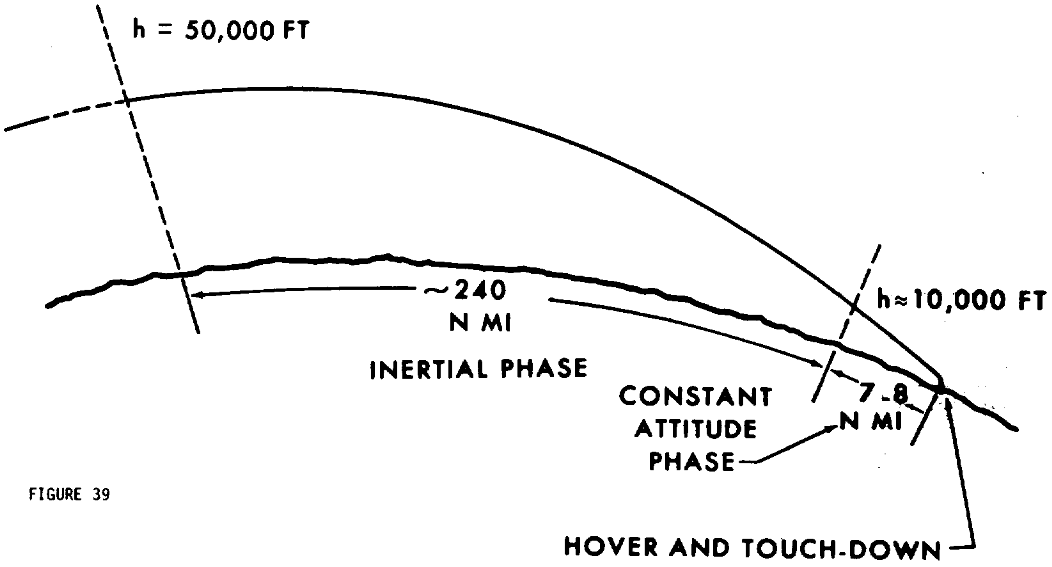

h = 50,000 FT
~240
N MI
INERTIAL PHASE
h≈10,000 FT
CONSTANT
ATTITUDE
PHASE
7-8
N MI
HOVER AND TOUCH-DOWN

FIGURE 39

NASA-S-66-3575 MAY 12

LUNAR LANDING MANEUVER CONSTANT ATTITUDE PHASE

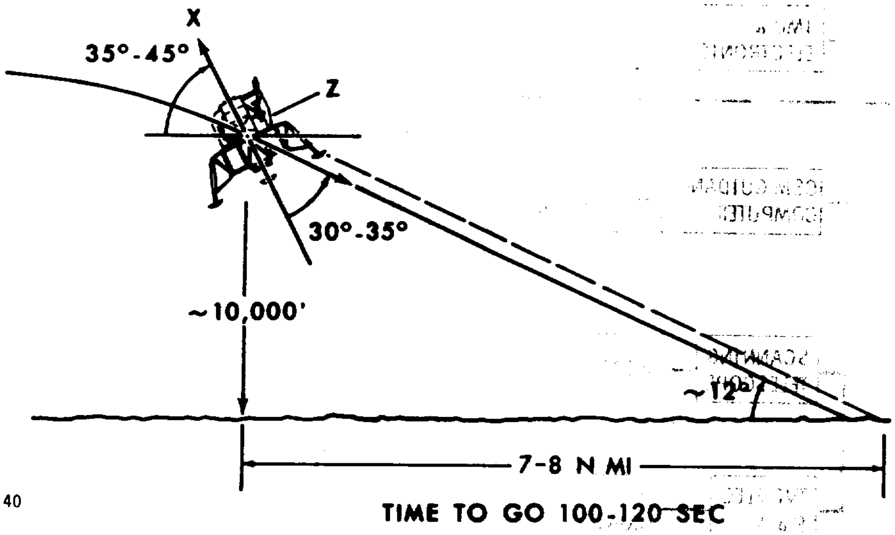

X
35°-45°
Z
30°-35°
~10,000'
~12°
7-8 N MI
TIME TO GO 100-120 SEC

FIGURE 40

NASA-S-65-4482

CSM RELIABILITY
CREW SAFETY - BLK II

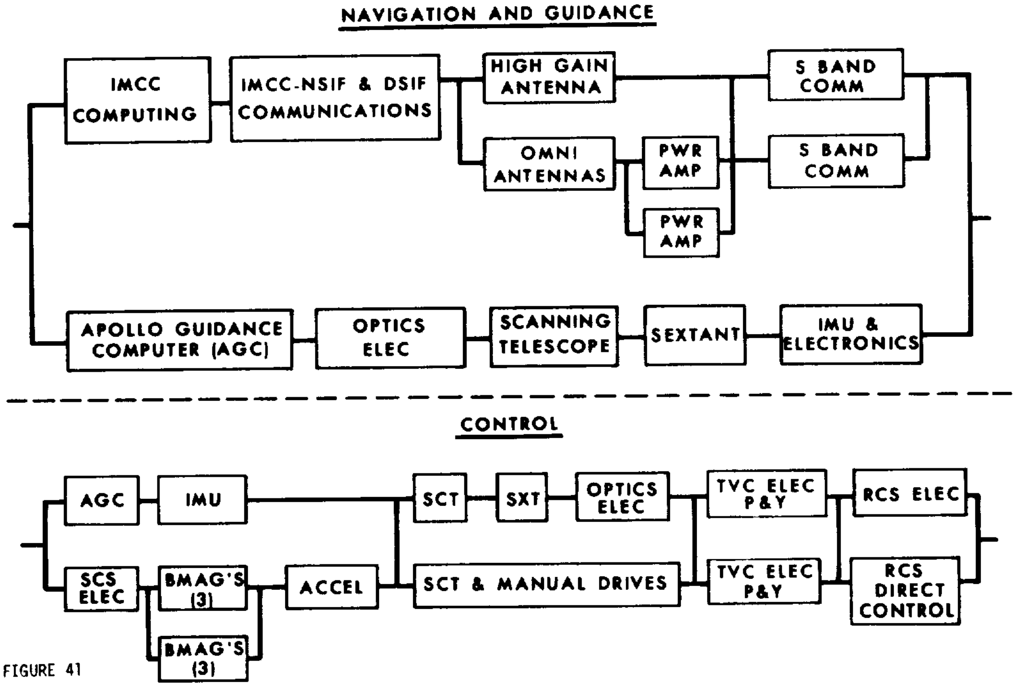

NAVIGATION AND GUIDANCE
IMCC COMPUTING
IMCC-NSIF & DSIF COMMUNICATIONS
HIGH GAIN ANTENNA
OMNI ANTENNAS
PWR AMP
PWR AMP
S BAND COMM
S BAND COMM
APOLLO GUIDANCE COMPUTER (AGC)
OPTICS ELEC
SCANNING TELESCOPE
SEXTANT
IMU & ELECTRONICS
CONTROL
AGC
IMU
SCS ELEC
BMAG'S (3)
BMAG'S (3)
ACCEL
SCT
SXT
OPTICS ELEC
SCT & MANUAL DRIVES
TVC ELEC P&Y
TVC ELEC P&Y
RCS ELEC
RCS DIRECT CONTROL

FIGURE 41

NASA-S-65-9999

CSM RELIABILITY
MISSION SUCCESS-BLOCK II

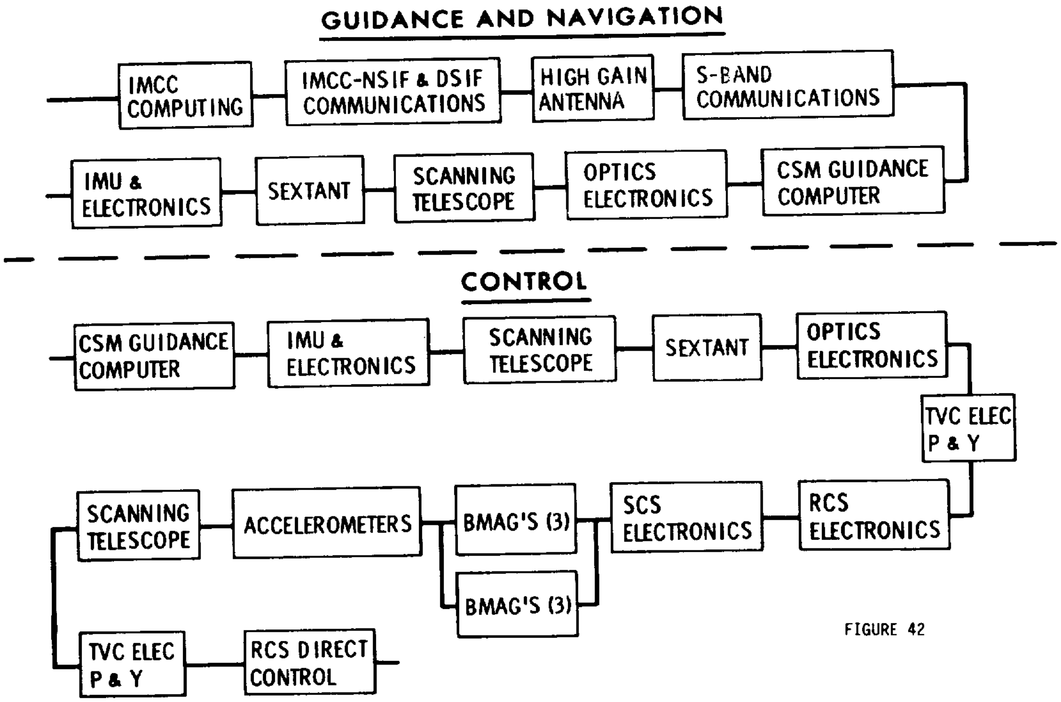

GUIDANCE AND NAVIGATION
IMCC COMPUTING
IMCC-NSIF & DSIF COMMUNICATIONS
HIGH GAIN ANTENNA
S-BAND COMMUNICATIONS
CSM GUIDANCE COMPUTER
OPTICS ELECTRONICS
SCANNING TELESCOPE
SEXTANT
IMU & ELECTRONICS
CONTROL
CSM GUIDANCE COMPUTER
IMU & ELECTRONICS
SCANNING TELESCOPE
SEXTANT
OPTICS ELECTRONICS
TVC ELEC P & Y
RCS ELECTRONICS
SCS ELECTRONICS
BMAG'S (3)
BMAG'S (3)
ACCELEROMETERS
SCANNING TELESCOPE
TVC ELEC P & Y
RCS DIRECT CONTROL

FIGURE 42

DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

by

M. P. Frank

DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

LAUNCH THROUGH EARTH ORBIT INSERTION

The purpose of this paper is to present the various constraints that affect the Lunar Mission planning, primarily in the form of trajectory shaping and the limitation to launch opportunities.

Trajectory geometry constraints and spacecraft performance capability combine to limit the accessible area on the moon. Accessible area limitations combine with operational constraints to limit launch opportunities to certain specifically defined periods. To understand the mission planning considerations and to appreciate the effects of the various constraints, one must closely examine the trajectory characteristics. In this paper an attempt is being made to explain the interrelation of constraints and trajectory shaping. Unfortunately, the explanation becomes quite detailed for some phases of the mission, but this is considered necessary in order to obtain an understanding of the interrelation.

We will begin with the launch and work through the mission, phase by phase. This paper will not describe every trajectory shaping consideration. It will only hit the highlights and discuss the more significant considerations--those that have a major effect on mission planning and the determination of launch windows.

LAUNCH PHASE

The mission planning considerations of the launch phase of the lunar mission are primarily related to launch "windows", booster performance, and contingency planning. Launch windows are defined for two different time intervals. A "daily window" has a duration of a few hours during a given 24-hour period. A "monthly window" consists of a few days during a given month or lunar cycle. The daily window is continuous from opening to closing; but a monthly window may have gaps. For example, a monthly launch window may cover a 7-day period, but a daily window may not exist for some of the intermediate days. A description of the factors that define the launch windows will be presented later in this session. For now, only the effects on the launch phase, of providing a window will be considered.

It is obvious that for operational flexibility it is highly desirable to have as large launch windows as possible, both daily and monthly. A daily window allows delays or holds in the countdown. The mission would not have to be rescheduled to another day if the window was larger than the cumulative delay or hold times. A monthly launch window allows the mission to be quickly rescheduled. If the daily window were missed, the mission would not necessarily be delayed for a month.

Although the duration and frequency of holds is strongly dependent on the actual vehicle, some estimation of the expected holds can be obtained from historical data. This data shows that for programs employing the Atlas, Titan, and Saturn launch vehicles only in rare cases was a hold of greater than 2 hours followed by a successful launch. This indicates that a 2 hour window should be adequate. If the hold exceeded 2 hours, the mission probably would have to be scrubbed anyway.

For the lunar missions, daily launch windows require changes in launch azimuth; the larger the daily window, the larger is the required azimuth change. The mechanism by which variable azimuth provides launch windows, will be described in the translunar injection phase. For now, let's assume it is required. The limitations to the launch azimuths that can be used, are based on considerations or range safety, booster performance and insertion tracking requirements.

Range Safety--In the early days of the lunar mission planning, the range safety limits were defined as 72° and 108°; however, there is some indication now that these could be increased if necessary. The primary concern of these range safety limits is to keep the space vehicle on the range following any aborts during launch.

Booster Performance--A 90° launch azimuth takes maximum advantage of the earth's rotation in achieving orbital velocity. As the azimuth is shifted away from 90°, the booster performance requirements are increased, or its payload capability is decreased as indicated in Figure 1. The Saturn vehicle is capable of providing lunar mission payloads for launch azimuths beyond the 72° to 108° range; however, its flight performance reserves are drastically reduced. For this reason, the mission will be constrained to these launch azimuth limits.

Insertion Tracking--There is an operational requirement to track the space vehicle from orbit insertion to at least 3 minutes following insertion in order to make a GO/NO-GO decision. Since the Apollo space vehicle is inserted into orbit some 1,400 n. mi. downrange, a ship is required to provide this tracking. The ship speed is relatively slow compared with the launch azimuth change during a launch window, and it cannot keep up with the changing ground track. Thus the tracking coverage downrange afforded by one ship limits the range of usable launch azimuths to 26° as shown in Figure 2. A 26° range in usable launch azimuths anywhere between the maximum values of 72° to 108° provides at least a 2.5 hour daily window. The choice of where the 26° range is located within the maximum bounds is left up to the mission planner, and is based on such things as maximizing spacecraft fuel reserve, MSFN tracking coverage, launch window duration, and providing a daylight launch.

Another constraint on the launch phase mission planning is the monthly launch window. A monthly launch window allows the mission to be rescheduled as soon as possible in case it is "scrubbed" for any reason for a given day or in case a hold extends beyond the daily window. It also allows some flexibility in the initial planning of the launch day.

How monthly launch windows are obtained and their limitations other than vehicle systems is the subject of the rest of this session. After describing the various constraining considerations during the rest of this talk we will come back to this particular subject. For now, we will discuss only the effects of vehicle recycle characteristics on the requirements and limitations.

The minimum turnaround time, or the space vehicle recycle time, is a major factor in defining the minimum acceptable duration of the monthly launch window. Studies for NASA Headquarters by Bellcomm have provided some significant data that has been used to develop the minimum launch window philosophy. The most important characteristic is the minimum recycle time for the space vehicle, shown in Figure 3.

If the mission were scrubbed after the countdown had reached T - 6 hours, the minimum time to recycle is in excess of 30 hours, and is as long as 40 hours at T = 0. Thus, a minimum window required to guarantee a recycle capability is 3 days. This does not allow any additional time for repairs or replacing components. If this activity could not be done in parallel with the recycling, 3 days would not be sufficient. Therefore, a window of only 3 days duration is not desirable, but is a minimum. In order to allow time for repairs and still make the monthly launch window, it should be as long as possible. The Bellcomm studies indicate that the probability of a successful launch is between 85% and 90% if a 3-day window is available, and that this increases to about 95% if the launch window is of 5-days duration. Based on this data, the lunar landing mission is being planned only for those periods when at least a 3-day window exists, and every effort is being made to provide 5-day windows.

One final consideration in the launch phase is the desirability of a daylight launch. There are three reasons which make a daylight launch highly desirable--they are all concerned with contingencies.

> Aborts off the pad--The recovery of the crew in the Merritt Island area would be complicated if it had to be performed under conditions of darkness.
>
> Aborts later in the launch require attitude maneuvering of the S/C, and it is desirable to have a sunlit horizon as a backup attitude reference.
>
> Finally, it is desirable to have photographic coverage of the boost phase for post-flight analysis, if a catastrophic failure occurs.

Based on these three considerations, every effort will be made to provide for a daylight launch, although the mission will not be constrained to daylight launches only.

EARTH PARKING ORBIT PHASE

Earth Parking Orbits are required in order to provide launch windows of reasonable duration. Direct lunar injections are possible; however, the launch windows are unacceptably small.

The only major consideration in the earth orbit phase is the duration--or the number of earth orbits. The parking orbit duration is bounded by space vehicle systems considerations. The maximum duration is 4.5 hours from orbit insertion to the beginning of injection and is limited by the launch vehicle capability to provide attitude control and by the battery lifetime. This allows up to three parking orbits prior to the second S-IVB burn. There are other considerations in limiting the parking orbit duration, although they are not "hard" constraints. The S-IVB propellant boil-off and inertail platform drift make it desirable to inject as soon as possible.

The minimum duration of the earth parking orbit phase is limited by the time required to perform system checks and realign the spacecraft platform. Crew timeline analysis indicates that this will require at least 1.5 hours.

There is also a minimum network coverage requirement that states that two tracking stations and a command station must be passed before the GO decision for the second S-IVB burn; however, this is always accomplished in the first orbit. Thus, there is a 3-hour period from 1.5 to 4.5 hours after orbit insertion in which the translunar injection can occur. This means that the injection must occur on the second or third orbit. Figure 4 illustrates the ground tracks for three earth orbits for a typical launch azimuth. The solid line indicates that part of the orbit on which a translunar injection could occur.

TRANSLUNAR INJECTION

The translunar injection position is rigidly constrained by performance considerations. The geometry of the moon's orbit, the energy requirements of the earth to moon transfer trajectory, and the necessity of efficiently burning the S-IVB propellant all combine to place very tight restrictions on the location of this maneuver. An attempt will be made in this section to show how these considerations are interrelated, and how this injection position is determined. It is somewhat involved and will take considerable explanation.

First, let's look at the transfer energy considerations. In order to arrive in the vicinity of the moon, the spacecraft is "aimed" at a position where the moon will be at the time of arrival as illustrated in Figure 5. In order to accomplish this "rendezvous" with a minimum expendature of propellant, the injection must occur very close to the extension of the earth-moon line at the time of arrival. This negative of the unit vector of the moon's position is called the moon's antipode. Something closely akin to a Hohmann transfer is what is being strived for.

This minimum energy transfer would place the perigee on the antipode if the moon's mass did not perturb the trajectory. However, the moon does perturb the trajectory, as shown in the figure, and the perigee must lead the antipode by approximately 8° to compensate. The apogee altitude of the osculating conic trajectory is determined by translunar flight time which defines the trajectory energy required.

To inject to the moon in the most efficient manner, an impulsive velocity would be added along the orbital velocity vector, giving an injection at perigee of the translunar conic. Since an impulsive addition of velocity is not possible, a finite burn time is required, and the actual injection position is on the order of 20° ahead on the antipode. The thrust is directed approximately along the velocity vector, and as the speed increases above orbital, the altitude and flight-path angle increase. For the Apollo configuration, by the time a sufficient energy increase is realized, the altitude increases 60 n. mi. above the orbit and a positive flight-path angle of 6° to 7° has been gained. Since the conic trajectory is very nearly parabolic (eccentricity $\approx .97$), the true anomaly is approximately equal to twice the flight-path angle, so perigee is approximately 12° to 14° behind the burn cut-off position. The burn arc itself is 25°, so that ignition always occurs within a few degrees of the antipode.

The preceeding discussion has shown that the injection position is very closely related to the moon's antipode. To go to the moon efficiently the spacecraft must inject near it, so that we must now address the problem of getting to the antipode from the launch pad at the Cape.

The antipode, being a unit vector from the center of the earth in the direction negative to the moon's position, moves as the moon travels in its orbit. The launch pad is rotating with the earth, and both of these motions must be compensated for in order to rendezvous with the antipode. It is convenient to divide the description of the antipode movements into two categories--a long period cycle and a short period cycle.

The long period cycle is due to the moon's orbital travel about the earth. Figure 6 will be used to illustrate this effect. Assume that the earth is a fixed, motionless sphere and is not rotating about its axis. The moon's orbit plane cuts this sphere as shown. As the moon revolves around the earth, its antipode would trace a great circle in this plane around the surface. Note that the direction of travel is from West to East. The orbital period is some 28 days, and thus at the end of this time the antipode would be back where it began, traveling at the rate of about 0.54° per hour. The latitude of the antipode would have a time history similar to that shown in Figure 6.

The short period motion of the antipode across the surface of the earth is due to the earth's rotation. To illustrate this it is assumed that the moon is fixed at some position in its orbit, and the earth is now allowed to rotate about its polar axis. The antipode travel is illustrated in figure 7. In this case the latitude is constant and the longitude changes from East to West at 15° per hour. The complete picture of the antipode travel across the earth's surface is obtained by combining the long period and the short period motions. The latitude varies sinusoidally with time with an amplitude of 28.5° (in 1968) and a period of 28 days. The longitude variation is at a nearly linear rate of 14.5° per hour.

The launch must occur at a certain time for each launch azimuth in order to intercept the antipode. This correct launch time is defined by the anitpode's position, the time interval from launch to arrival at this position, and the antipode travel during this time interval. Figure 8 illustrates this problem. Consider an inertial sphere of radius equal to the earth. A trace of the launch pad travel as a function of time on this sphere is represented by a fixed latitude completely encircling the sphere. The launch pad completes one revolution per day. The trace of the antipode is given by the intersection of the moon orbit plane and the sphere. The antipode completes a revolution every 28 days. The launch at any given azimuth must be timed so that the inertial plane of the resulting orbit contains the antipode at the time the space vehicle crosses the moon orbit plane. Later launch times require greater launch azimuths. If additional parking orbits are required, the launch must occur later to account for the additional antipode travel. For each 360° travel of the launch pad, there are two launch times for each azimuth which allow interception of the antipode. This is better illustrated in figure 9, which shows the same situation in earth-fixed coordinates.

In this figure, the launch pad is now fixed, and the antipode travels rapidly over the surface of the earth. The antipode position is shown at four different times during the day, corresponding to the positions at intercept for 72° and 180° launch azimuths. The launch must be timed so that the vehicle intercepts the moving antipode. The time required for the antipode to travel from the interception of the 72° launch azimuth trajectory defines the launch window duration. The intersection of the MOP is drawn in for each position. This figure shows how two different launch times for one azimuth can provide intercept with the antipode. One provides injections going south over the Atlantic Ocean and the other provides injections going north over the Pacific Ocean. For the day illustrated in this figure, the Atlantic injection gives a trajectory that is nearly in the moon's orbit plane, and the Pacific injection results in a trajectory that is highly inclined to the moon's orbit plane. Half a lunar cycle later, the Pacific injection would be highly inclined. The magnitude of this relative inclination depends on the lunar declination and is a maximum when the moon is near the equator. When the moon is near maximum declination, both windows provide trajectories with low relative inclinations.

It is of special significance that the Pacific injection always results in a trajectory above the moon orbit plane, regardless of the moon's declination or whether it is ascending or descending. This effects the relative location of available landing areas on the moon from these two injection windows. This effect will be described later.

It can be seen that when the launch azimuth bounds are defined, the proper launch time can be found, allowing for the number of parking orbits to be employed prior to injection.

If for some reason the injection opportunity were missed, it could be attempted one orbit later when the space vehicle again approached the antipode. However, since the antipode is traveling in a plane that is not necessarily the same as the vehicle's orbit, a plane change would be required. This is illustrated in figure 10. It can be seen that the antipode has traveled out of the parking orbit plane when the vehicle returns to the position of injection. The magnitude of the out-of-plane travel is dependent on the relative inclination between the parking orbit and the moon orbit planes. The maximum value is about .6°. This second injection would require a greater propellant expenditure by the S-IVB because of the plane change involved. If two injection opportunities are to be provided, the launch would be timed so that both would require a plane change, because this minimized the propellant required. The launch would occur a little bit later so that the first time the vehicle crosses the moon orbit plane, the antipode has not reached the parking orbit plane. The second time the vehicle crosses the moon orbit plane, the antipode has passed through the parking orbit plane. If three injection opportunities are to be provided, the launch would be timed so that the antipode was in the parking orbit plane for the second one.

Figure 11 illustrates the effects of different targeting methods on the characteristic velocity required to provide additional injection opportunities. Three cases are shown. The first shows additional ΔV required when the launch is timed for the first injection to be coplaner. The second and third opportunities have large additional ΔV requirements.

In the second case, the launch is timed to split the delta azimuth between the first and second injection opportunities. This would be used for two injection opportunities.

In the third case, launch is timed so that second injection opportunity is coplaner. This method would be to provide three injection opportunities.

The penalties shown are only illustrative; the actual values strongly depend on relative inclination between the two planes.

Because injection is limited to the second or third orbit, only two injection opportunities are planned for 504 mission, and the second targeting technique is being used.

The combination of launch azimuth limits, parking orbit duration constraints, and the geometry of the moon's orbit confine the location of the injection positions to two geographical areas. These areas are generally centered over the South Atlantic Ocean and the Pacific Ocean, and for this reason are distinguished by these names. The bounds, as shown as figure 12, are defined by the first orbit for a 72° launch azimuth, the third orbit for a 108° launch azimuth, and the extremes of lunar declination. The areas shown in figure 12 contain all of the possible injection positions.

TRANSLUNAR COAST PHASE

In this, the description will be confined to: the effects the trajectory inclination relative to the moon's-orbit plane; the effects of the "free-return" flight plan, and its relationship to the lunar orbit insertion maneuver; and finally a discussion of some alternatives to the free-return flight plan.

The first point to be made in describing the translunar coast trajectory is in regard to the relative location of the trajectory to the moon's orbit plane. It was stated earlier that Pacific injections always result in translunar trajectories from Atlantic injections are always below it as shown in figure 13. The amount of out-of-planeness is a function of the moon's declination and whether or not it is ascending or descending in its orbit. These parameters influence the magnitude of the effects, however they do not change the general conclusions.

Following a Pacific injection, the spacecraft approaches the moon from above the moon orbit plane. This forces the trajectory below this plane on the far side of the moon, where the lunar orbit insertion maneuver takes place. The resulting lunar orbit then is constrained to be approximately as illustrated in figure 14. A plane change during the orbit insertion can modify the resultant orientation somewhat, but the basic conclusion can still be drawn that to land at northern latitudes on the front side of the moon, a Pacific injection will result in lower propellant costs. Conversely, Atlantic injections favor the southern latitudes. This will be clearly demonstrated later when the accessible lunar areas are defined.

FREE RETURN

One of the most constraining requirements of the lunar landing mission is the free-return trajectory. It incurs sever limitations to the area on the moon to which Apollo missions can be conducted. Although it is costly in terms of spacecraft performance requirements, the inherent safety feature of a free return makes it a highly desirable method of getting to the moon.

A circumlunar free-return trajectory, by definition, is one which circumnavigates the moon and returns to earth as shown in figure 15. The perigee altitude of the return trajectory is of such a magnitude that by using negative lift the reentering vehicle can be prevented from skipping out of the atmosphere, and the aerodynamic decelleration can be kept below 10 g's. Thus, with a complete propulsion system failure following the translunar injection, the spacecraft would return safely to earth.

The range in return perigee altitudes that provide this feature is called the reentry corridor and is primarily a function of the lift-to-drag ratio of the reentry vehicle. For the Apollo vehicle this corridor is approximately ± 12 n. miles centered around a 25 n. mile altitude. The injection velocity accuracy required to achieve a free-return trajectory is less than a tenth of a foot per second. Obviously, this is well beyond the capability of any guidance system when the total ΔV involved is on the order of 10,000 fps. However, it is still valid to plan for a free-return trajectory, because this procedure at least will minimize the ΔV requirements to return to earth should an SPS failure occur. In this situation, there is a good probability that the RCS can provide the necessary velocity corrections to overcome the injection errors.

The free-return trajectory severely limits the accessible area on the moon because of the very small variation in allowable lunar approach conditions and because the energy of the lunar approach trajectory is relatively high. The high approach energy causes the orbit insertion ΔV to be relatively high. However, the main limitations to accessible area are a result of the small range in flight times from earth to moon. Figure 16 illustrates the effect of flight time on the location of perilune. All free-return trajectories have translunar transit between 60 and 80 hours, and it can be seen in figure 16 that perilune is limited to a region within about 10° of the negative of the earth-moon line or approximately 180° longitude. For non-free-return trajectories, the transit time can be anything from 50 hours to 110 hours.

Perilune could be adjusted from 140° W longitude to 140° E longitude merely be selecting the appropriate flight time. This narrow region of perilune position of free-return trajectories combined with a small range of approach inclinations is what limits the accessible area.

The relative inclination between the free-return trajectory and the moon orbit plane is less than 11°. Any trajectory with a greater inclination than this, simply does not return to the entry corridor at earth, regardless of the perilune position. The range of free-return trajectory conditions near the moon is illustrated in figure 17. Note the relatively small cone formed by the locus of perilune positions.

The braking maneuver to decellerate the spacecraft from the hyperbolic approach trajectory to a lunar orbit is performed at or near perilune. For illustrative purposes, it will be assumed that it occurs at perilune. In order to land at a site that is not contained by the approach trajectory plane, a plane change must be made. It is generally more efficient to combine this plane change with the decelleration at orbit insertion. When the landing site is near the mode, however, an excessively large plane change is required to cause the trajectory to pass over the site. This is illustrated in figure 18. Since the approach trajectories have low inclinations and orbit insertion occurs near the 180° longitude, it can be seen that to cause the lunar orbit to pass over sites at high latitudes in the region near 0° longitude large plane changes would be required. The propellant capacity of the spacecraft limits the magnitude of the plane change that can be made.

As was noted in the preceeding figure, there is a locus of perilune positions; it is not that there is not one focal point through which all of the trajectories must pass; there is an area. This tends to relieve the limitations slightly, but the fact remains that a plane change at deboost is relatively ineffective in achieving higher latitudes near the zero longitude. Note also that as the landing site is moved away from the 0° longitude, the plane change requirements become much less.

If the orbit insertion were not made at perilune, the magnitude of the plane change could be reduced in many cases as illustrated in figure 19. In this figure, two lunar orbits resulting from orbit insertion at two different positions along the approach hyperbola illustrated. Both pass over the landing site, and both could be acceptable.

If the insertion was performed at perilune, a much larger plane change would be required, so it appears that if the insertion were made prior to perilune, the ΔV required would be much less. However, there is an additional penalty associated with this pre-perilune braking due to the fact that a flight-path angle change must also be made. Figure 20 shows the in-plane geometry. If the deboost is performed at any position other than perilune, the velocity vector must not only be reduced, but its direction must also be changed if we are to achieve a circular orbit. A flight-path angle change is just as expensive as an azimuth change were required. It is much more efficient to make a small plane change and a small azimuth change than it is to make a large azimuth change. This trade-off is made in the mission trajectory design to obtain the optimum combination.

Another feature of this non-perilune deboost is that the resultant orbit altitude is above the perilune altitude. The perilune must be reduced a certain amount in order to obtain the desired orbit altitude as illustrated in figure 20. The exact amount of reduction depends on the true anomaly of the deboost maneuver, but in no case is a perilune altitude of less than 40 n. miles employed.

Since the free-return flight plan is so constraining on the accessible lunar area, parallel investigations of other techniques are being conducted. The primary goal of these parallel investigations is to develop techniques that retain most of the safety features of the free return, but do not suffer from the performance penalties. An example of this type of mission plan is something termed a Hybrid Flight Plan illustrated in figure 21. The spacecraft is injected into a highly eccentric elliptical orbit which has the free-return characteristic; that is, a return to the entry corridor without any further maneuvers. The launch vehicle energy requirements are reduced, and a greater payload (more SPS propellant) could be carried. Some three to five hours after injection, after the SPS has been checked out, a mid-course maneuver would be performed by the spacecraft to place it on a lunar approach trajectory. This lunar approach trajectory would not be a free return, and hence would not be subject to the same limitations in trajectory geometry. Landing sites at high latitudes could be achieved, with little or no plane change, by approaching the moon on a highly inclined trajectory. This Hybrid Flight Plan offers large improvements in performance over the free return plan and still retains most of the safety features. The spacecraft does not depart from the free-return ellipse until the LM docking has been completed (providing a second propulsion system for returning to earth) and then only if the SPS checks out O.K.

One of the difficulties in flight planning the Hybrid mission is that the initial trajectory is not ammendable to conic approximations. So much time is spent milling around out near the moon that conics or patched conics do not provide accurate simulations. It is extremely important that rapid calculation procedures be available because of the large number of iterations required to "search in" or design a mission trajectory. And if all of this must be done with precision integrating trajectory programs, the computer time becomes excessively large. Work is continuing in an effort to develop this Hybrid Flight plan capability. There are variations on this Hybrid Plan which look very promising, and these are also being investigated.

A comparison of accessible area available for Hybrid and free-return flight plans is given in figure 22. Only the area between 45° E and 45° W longitudes is shown, as this is the primary zone of interest. The area available with free-return trajectories is limited near the equator. While the area attainable with the Hybrid mission, which is essentially the same as that for a non-free-return mission, is much larger. It includes all of the area available with the free-return, and extends to much higher latitudes at the smaller longitudes.

LUNAR ORBIT PHASE

There are only two parameters of interest in this phase. These are the orientation of the lunar orbit and the number of parking orbits required. The orientation of the plane of the lunar orbit is selected to minimize the Δ V requirements. There are three maneuvers that must be considered in this optimization.

The three maneuvers are: the lunar orbit insertion, the transearth injection, and a lunar orbit plane change performed by the SM during the LM stay on the lunar surface.

The moon's relatively slow rotation rate, combined with the low orbit inclinations result in a small out-of-plane motion of the landing site. The LOI maneuver is planned so that the resulting parking orbit plane contains the landing site at the nominal time of landing, as illustrated in figure 23. Position 1 represents the location of the landing site at the time of lunar orbit insertion. At the time of landing it has rotated to position 2.

During the lunar surface stay, the landing site continues to rotate out-of-plane to position 3, and in order to reduce the LM maneuvering requirements, the SM makes a plane change maneuver prior to LM launch. This maneuver is planned so that the landing site is in the new parking orbit plane at the nominal time of launch.

The transearth injection maneuver is performed from this final parking orbit orientation and, in general, a plane change is required. The SM performance requirements are minimized by selecting the best orientation of the lunar parking orbit consistant with the location of the lunar landing site and the lunar surface stay time.

The number of parking orbits both prior to LM descent and after rendezvous are dictated by crew procedure timelines and MSFN tracking considerations. After the LOI, three orbits are required for the crew to activate and checkout the LM. After rendezvous, two orbits are required to prepare for transearth injection. This allows sufficient tracking for orbit determination by the MSFN.

At this point, the sequential description of the mission planning considerations and constraints by mission phase will be interrupted. The trajectory shaping parameters have been described in sufficient detail to show the effect on launch opportunities.

TOTAL LAUNCH WINDOW CONSIDERATIONS

There are at least six major considerations that in one way or another limit the times at which the Apollo lunar landing mission can be launched. These constraining factors are due to either the characteristics of the moon's orbit about the earth, operational requirements, or spacecraft performance capability.

The constraints are listed below in the order they will be discussed.

1. Launch azimuth
2. Lighting conditions at lunar landing
3. Lunar landing site location
4. Spacecraft performance
5. Daylight launch
6. Minimum number of launch opportunities during a month

LAUNCH AZIMUTH

The mechanism of the launch azimuth effect on launch time was described earlier. It was shown that a specified launch azimuth and a specified number of earth parking orbits defined the required launch time that provided a rendezvous with the moon antipode. The fact that either of two distinct launch times would provide this rendezvous was illustrated. It was also pointed out that one of these launch times resulted in translunar injections approximately over the Atlantic Ocean, and the other resulted in injections approximately over the Pacific Ocean.

This characteristic of discrete launch azimuths defining discrete launch times can be expanded to show that a range of launch azimuths define a range of launch times. And for a given range, such as 72° to 108°, the launch times for each injection window can be readily determined. These daily windows, as limited only by launch azimuth, are shown for the year 1968 in figure 24. In this figure, the unshaded areas represent allowable launch times. The letters "P" and "A" denote the Pacific and Atlantic injection windows. Each window is opened at a 72° launch azimuth and closed at 108°.

The difference in launch time for the two windows varies throughout each month. In some periods the closing of one window is followed immediately by the opening of the other window. At other times there is as much as 14 hours between the closing of one window and the opening of the other.

Note, also, that the time of opening of a given window is later for each successive day. The rate of change per day is quite rapid at times and at other times is almost negligible. The relatively flat period for a particular window corresponds to the part of the lunar month when translunar injection is in the moon orbit plane. This daily shift in the time of launch and differences in launch time between the two windows are important characteristics to keep in mind in the subsequent discussion.

LIGHTING CONDITIONS AND LUNAR LANDING SITE LOCATION

These two considerations are inseperable in their effects on launch windows. The effects of lighting constraints can only be evaluated in conjunction with the landing site longitudinal location as will be seen in the following discussion.

In order to provide the LM crew with the best possible visibility conditions during landing, the sun elevation angle at the landing site during the powered descent must be between 7° and 20° above the eastern horizon. This is to allow the crew to visually evaluate the possible landing points and select a favorable one within the LM "footprint". The derivation of this range of sun elevation angles will be presented later in "site selection" session.

The magnitude of the allowable range in sun elevation angle has a major effect on the determination of launch windows. Figure 25 illustrates the lighting geometry. In this figure the sunrise terminator is located approximately at 0° longitude. For this condition a lunar landing could only be accomplished in the region enclosed by the dotted lines. This region of acceptable lighting moves across the face of the moon, following the sunrise terminator from East to West at a rate of approximately 13° per day, so that the days of acceptable lighting conditions as a function of landing site longitude can be readily determined from lunar ephemeris data. The effects of latitude can be neglected in the region near the lunar equator. Figure 26 provides an example of this variation for two typical months in the first quarter of 1968. The most striking feature of this figure is the fact that for approximately 60% of the month there is no area with acceptable lighting conditions anywhere between the longitudes of 45° E to 45° W. No landing is possible from February 12 to March 3, even if the only restriction on landing site were that it must be between 45° E to 45° W.

The effect of restricting landings to specific sites can be illustrated on this same figure. For example, suppose that the only available landing site were located at 25° E; from figure 26 it can be seen that in the months of February and March of 1968 there are only two days on which a lunar landing could be accomplished--February 5 and March 5. One landing day is avaiable during each 28-day lunar cycle. In order to provide multiple landing opportunities, several sites must be available.

The number of landing opportunities is directly related to the number of launch opportunities. For lunar landing missions employing free-return trajectories, the relationship between landing time and launch time has only a slight variation. Therefore, arrival (or landing) times can also be shown on this same figure. The landing times associated with the acceptable lighting period in February 1968 are presented in figure 27. For simplicity, only the Pacific injections are shown. Arrival times for Atlantic injections would be shifted by the difference in launch times. The launch date is noted for each band of arrival times. The variation in arrival times within a given launch window is about 10 hours and is due to: the variations in launch time between launch azimuths of 72° to 108°; the possibility of injecting on the first or second injection opportunity; and the variation in the translunar transit time for different energy trajectories available in the free-return family. For a $2\frac{1}{2}$ hour earth launch window, this band of arrival times would be reduced to about 8 hours. The shaded areas represent the range of landing site longitudes that have acceptable lighting for each of the launch days. Note that for each launch day, the region of available longitudes for landing is different. This region moves westward at about 13° per day, and there is virtually no overlap. No one longitude is available for more than one launch day when the requirement for a $2\frac{1}{2}$ hour launch window is considered. This clearly illustrates the constraining effects of lighting requirements and landing site location on the launch window. In order to provide multiple launch opportunities during the month, several lunar landing sites must be available. One additional site is required for each additional launch opportunity. It can also be seen from this figure that to avoid duplication and gaps in the launch window the longitudinal spacing of these sites must be increments of approximately 8° to 13°. This longitudinal spacing minimizes the probability of having 2 sites available on the same day, or what would be worse of having no site available on one day.

We can determine from this figure, the landing site distribution required to provide monthly launch windows. For example, to guarantee that a launch opportunity exists on consecutive days the landing sites must be located between 8° and 13° apart in longitude. One site provides one launch opportunity. To provide an alternate day launch window configuration, the sites must be located some 20° to 26° apart. To get a launch window of 5 days duration, the first and last site must be about 50° to 60° apart.

So far, we have only described the lighting effects on lunar landing site longitudinal location and the effect on launch dates. The latitudinal location of the landing site also has an effect on available launch dates because of its effects on Service Module performance requirements. The lighting may be acceptable, but if the landing site is outside the latitude bounds of spacecraft performance,.then a launch opportunity still does not exist. The latitude limits which can be attained are defined by SPS propellant available are a function of selenographic longitude, lunar declination and librations, and the translunar injection window.

The accessible area for a typical day is illustrated in figure 28. Note that the area available from the Pacific injection window is somewhat north of that available from the Atlantic window. These areas shift to the south as the moon travels to northern declinations and vice versa. In order for an acceptable launch window to exist on any given day, the landing site must be within the area defined by these latitude bounds and within the longitude region defined by the lighting bounds. The latitudinal shift in the accessible area boundaries is cyclic, with a period approximately equal to the lunar orbit period.

If a region on the lunar surface lies in the accessible area throughout the month, regardless of the daily shifting of these areas, it is said to be 100% accessible. A 100% accessible area has great significance in the selection of lunar areas to be examined for possible landing sites. If the candidate sites can be located in a region that is always accessible, then the mission planner is releived of one very troublesome constraint, namely spacecraft performance. It can be guaranteed that no mater what the lunar declination or libration, when the lighting is acceptable, the landing site is attainable.

The area available every day of the month presents a pessimistic picture, in that it does not consider the fact that only about 8 days are really usable because of the lighting constraint. A more realistic picture of the area available for a month, would be obtained if the latitude limits were defined for the longitudinal regions on the days when the lighting was acceptable in those regions. That is, select only those days when the sun elevation is between 7° and 20° for the longitudes between 45° E and 45° W; and furthermore on any one of these days define the latitude limits only for the longitude region which had acceptable lighting. The area available during the month of February 1968 under these conditions is illustrated in figure 29. The available area defined in figure 29 cannot be extrapolated from month to month because the lighting cycle and the declination and libration cycles do not have the same period. So that one of these figures must be made for every month of interest.

The accessible areas for each month during a year can be combined to define an area available for the entire year. This is illustrated in figure 30.

The purpose of all of this discussion of performance limitations has been to show that the landing site location has a major effect on launch opportunities; not only through lighting conditions and longitude interactions but also through latitude and performance interactions. This total interaction can be summarized as follows: Given a landing site location, a launch is possible only on the day that the lighting is acceptable and then only if the landing site is within the latitude bounds attainable for that longitude on that day.

If lunar landing sites could be selected entirely on the basis of performance and lighting constraints, they could be located so that there would be no restrictions on launch windows. Unfortunately, there are many factors that must be considered in the selection of lunar landing sites. These other factors force compromises to be made, with the result being that launch windows are in fact constrained by the available lunar landing sites. A complete description of the lunar landing site criteria, other than the launch window considerations will be presented in a later session.

We have seen that lunar landing sites, lighting requirements, and spacecraft performance are all very effective in constraining the launch opportunities. Just how constraining, will be illustrated shortly. In order to illustrate the limitations on launch opportunities, it will be necessary to assume certain lunar landing sites. For the purposes of this illustration, we will assume that the sites to be photographed on the lunar Orbiter A and B missions are found to be acceptable for Apollo landings.

First, let's consider the seven sites to be photographed by Orbiter A. The launch opportunities provided by these 7 sites throughout the year 1968 are summarized in figure 31. The interesting features of this figure are the pattern and frequency of launch opportunities, and the frequency of night launches.

During some of the months, many launch opportunities are not shown for the Pacific injection window. This is because some or all of the sites were outside of the performance boundaries and were not available when the lighting was acceptable. The Pacific injections are the hardest hit by this performance limitation because the western sights to be photographed by Orbiter A are generally south of the equator, away from the best performance region of the Pacific windows. In the early part of the year, the moon's declination and lighting are in a favorable phase, and the southern sites are well within the performance capability of the Pacific injection. However in the latter part of the year, the moon is at a northern declination when the lighting is acceptable and the southern sites are outside of the accessible area.

Since the best performance region for the Atlantic injection window is south of that for the Pacific, the lighting and libration combination is favorable for this window in the latter part of the year. The result is that for those months when the Pacific injections are unavailable, the Atlantic injections are. So launch windows exist all year.

The effect of constraining the mission to be launched only in daylight can also be determined from figure 31. Although this is not considered to be a firm constraint for all lunar missions, it is highly desirable and every effort will be made to have a daylight launch at least for the first one.

From figure 31, it can be seen that to limit the launch to daylight hours eliminated the Atlantic injection window for the entire year. In addition, several of the launches using the Pacific injection window occur at night during the winter months and would also be lost.

The net result of the combination of performance limitations and a daylight launch constraint would be to virtually eliminate all launch windows in the last quarter of 1968.

The effect of a minimum launch window duration constraint is demonstrated in figure 32. This figure shows the remaining launch opportunities if only those launch windows of 5-days duration were considered. It is clear that the launch opportunities afforded by the seven sites used in this analysis are entirely satisfactory during the first half of 1968. And that the situation rapidly changes from marginal to unsatisfactory during the third quarter and remains that way for the rest of the year because night launches would be required. Five-day daylight windows are available in March, April, May, and early June. However, beginning in late June and continuing throughout the rest of year, a night launch would be required.

To improve the daylight launch capability, the sites to be photographed on the Orbiter B mission were located north of the A sites. This provided sites in a area more favorable from the Pacific injection window. The resulting 5-day launch windows are summarized in figure 33. It can be seen from this figure that 5-day windows with a daylight launch could be obtained throughout the year if the Orbiter B sites were available.

Questions and Answers

LAUNCH THROUGH LUNAR ORBIT

Speaker: M. P. Frank

1. Does the launch azimuth change through the launch window?

   ANSWER - Yes; however, the launch azimuth is changed in discrete steps.

2. Dr. Reiffel - At sufficiently high altitudes both the solar cycle and individual solar effects can markedly affect the scale heights and density. In view of the mechanization of the re-entry monitor being planned which senses, as I understand it, very low G forces and rates of change to decide whether the spacecraft is in the corridor - what would be the effects of a flare heating the high atmosphere (and perhaps also requiring a radiation induced abort) and what data are really available on atmospheric structure for the proper time in the solar cycle on which to base the corridor design? Is significant solar-induced dispersion possible in other words?

   ANSWER - MSC will investigate the effects of solar flares on the entry conditions.

NASA-S-66-6460 JUN

EFFECT OF LAUNCH AZIMUTH ON PAYLOAD CAPABILITY

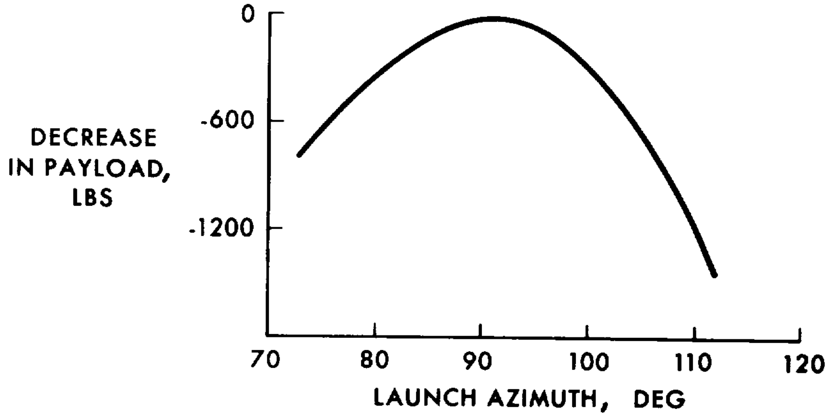


Figure 1

NASA-S-66-6017 MAY

ORBIT INSERTION

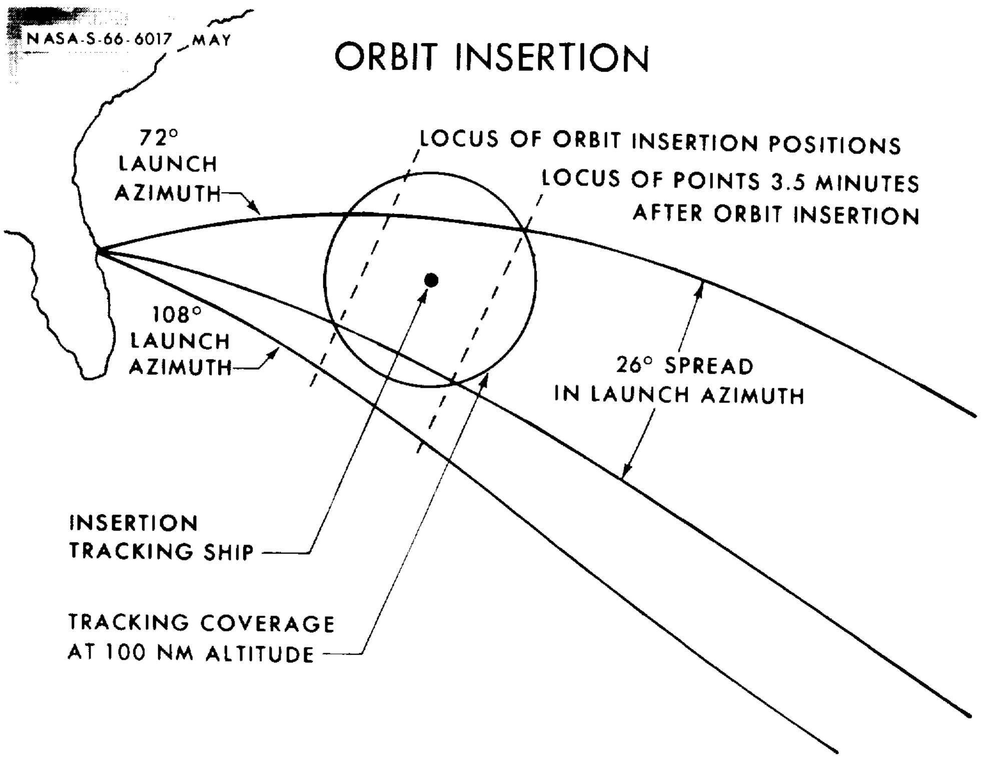

LOCUS OF ORBIT INSERTION POSITIONS
LOCUS OF POINTS 3.5 MINUTES AFTER ORBIT INSERTION
72° LAUNCH AZIMUTH
108° LAUNCH AZIMUTH
26° SPREAD IN LAUNCH AZIMUTH
INSERTION TRACKING SHIP
TRACKING COVERAGE AT 100 NM ALTITUDE

Figure 2

NASA-S-66-6003 MAY

LAUNCH ABILITY BY HOLD OR RECYCLE

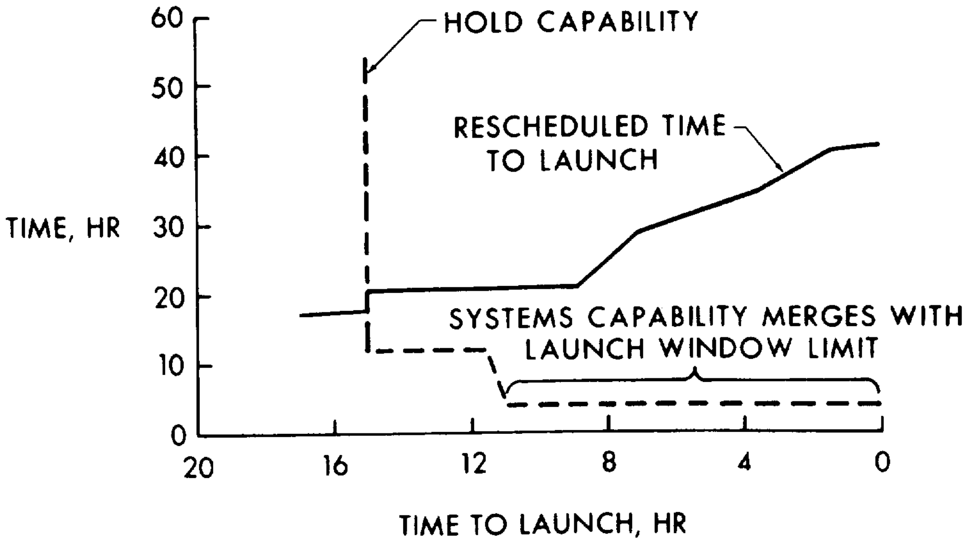


Figure 3

NASA-S-66-6471 JUN

EARTH PARKING ORBIT GROUND TRACK FOR THREE ORBITS FOR A 90 DEGREE LAUNCH AZIMUTH

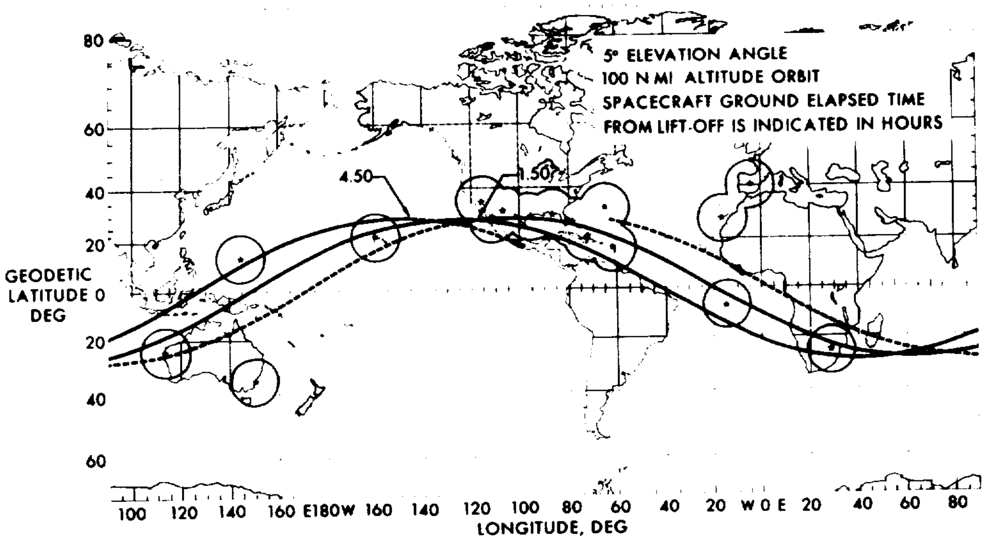

5° ELEVATION ANGLE
100 N MI ALTITUDE ORBIT
SPACECRAFT GROUND ELAPSED TIME FROM LIFT-OFF IS INDICATED IN HOURS
4.50
1.50
80
60
40
20
GEODETIC LATITUDE 0 DEG
20
40
60
100 120 140 160 E180W 160 140 120 100 80 60 40 20 W 0 E 20 40 60 80
LONGITUDE, DEG

Figure 4

NASA-S-66-6011 MAY

TRANSLUNAR INJECTION GEOMETRY

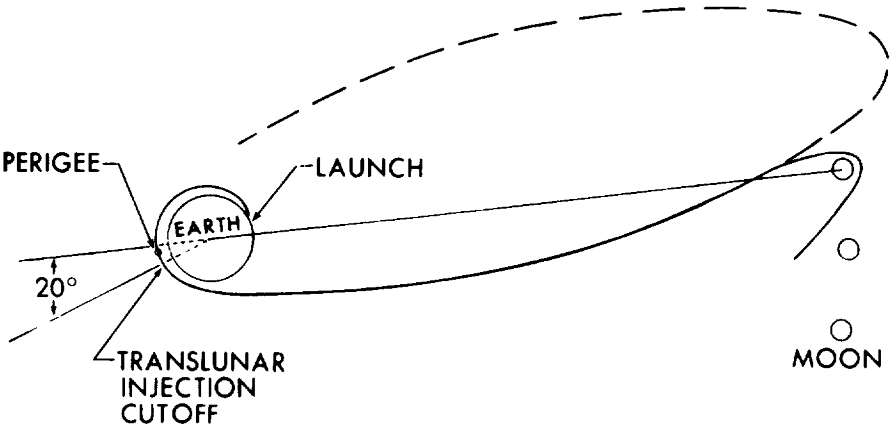

PERIGEE
LAUNCH
EARTH
20°
TRANSLUNAR
INJECTION
CUTOFF
MOON

Figure 5

NASA-S-66-6015 MAY

LONG PERIOD MOTION OF ANTIPODE

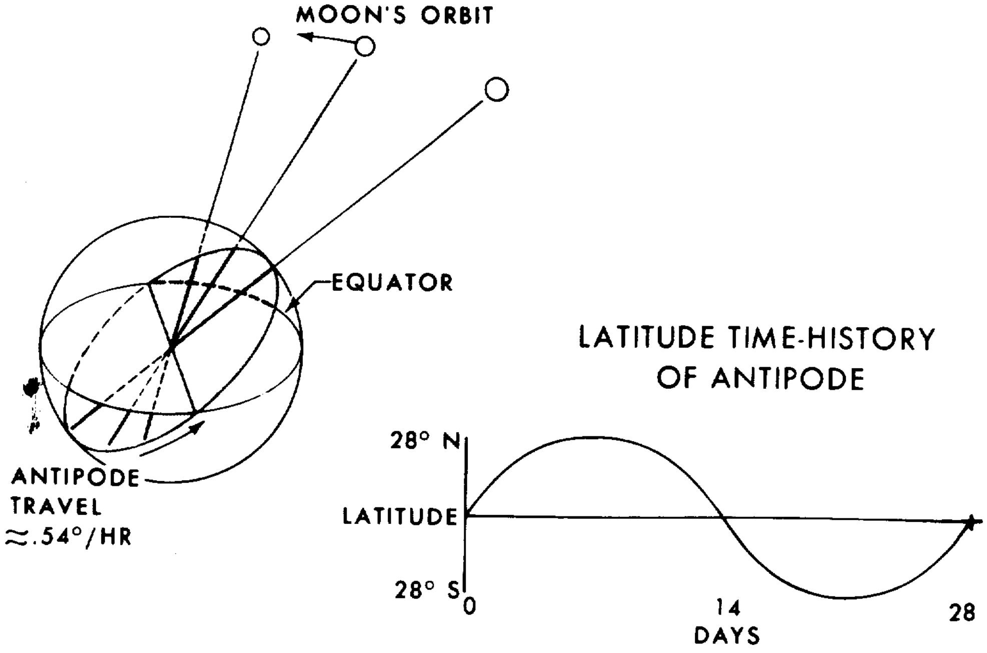

MOON'S ORBIT
EQUATOR
ANTIPODE
TRAVEL
≈.54°/HR
LATITUDE TIME-HISTORY
OF ANTIPODE
28° N
LATITUDE
28° S
0
14
28
DAYS

Figure 6

NASA-S-66-6023 MAY

SHORT-PERIOD MOTION OF ANTIPODE

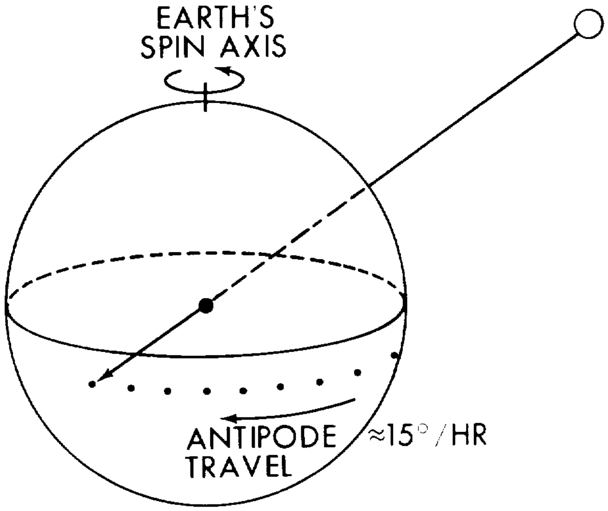
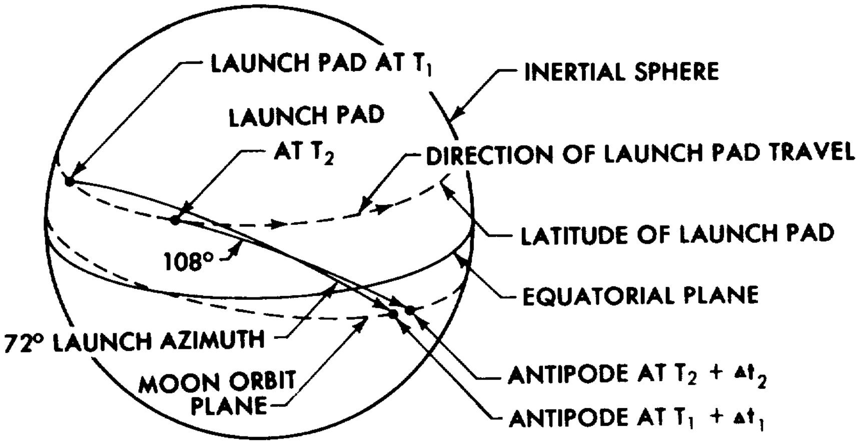

EARTH'S SPIN AXIS
ANTIPODE TRAVEL
≈15°/HR

Figure 7

NASA-S-66-6419 JUN

ANTIPODE RENDEZVOUS GEOMETRY


LAUNCH PAD AT T_1
INERTIAL SPHERE
LAUNCH PAD AT T_2
DIRECTION OF LAUNCH PAD TRAVEL
LATITUDE OF LAUNCH PAD
108°
EQUATORIAL PLANE
72° LAUNCH AZIMUTH
MOON ORBIT PLANE
ANTIPODE AT $T_2 + \Delta t_2$
ANTIPODE AT $T_1 + \Delta t_1$

Δt_1 = TIME FROM LAUNCH TO FIRST CROSSING OF MOP FOR 72° LAUNCH AZIMUTH

Δt_2 = TIME FROM LAUNCH TO FIRST CROSSING OF MOP FOR 108° LAUNCH AZIMUTH

Figure 8

NASA-S-66-6016 MAY

TRANSLUNAR INJECTION CHARACTERISTICS

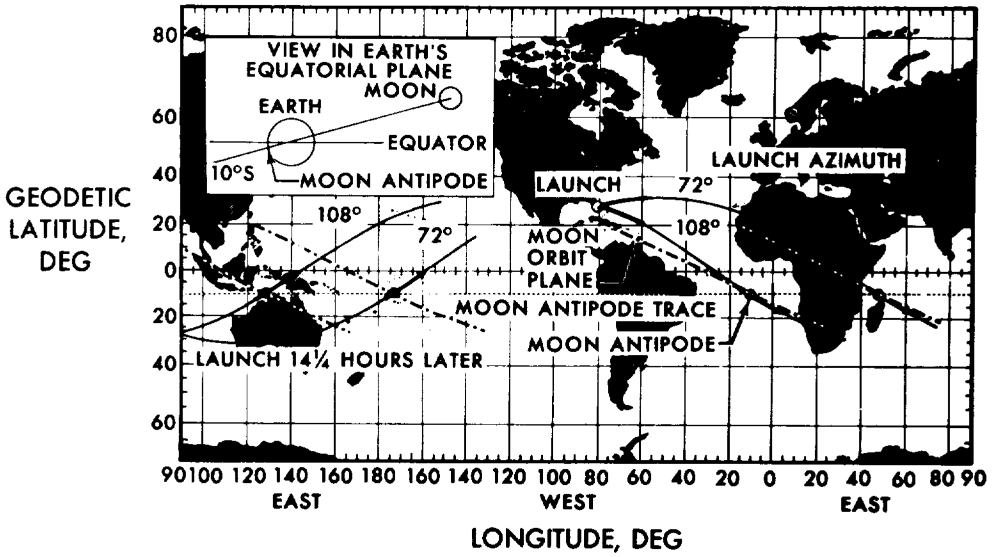

VIEW IN EARTH'S EQUATORIAL PLANE
MOON
EARTH
EQUATOR
10°S
MOON ANTIPODE
LAUNCH
72°
LAUNCH AZIMUTH
108°
MOON ORBIT PLANE
108°
72°
MOON ANTIPODE TRACE
MOON ANTIPODE
LAUNCH 14¼ HOURS LATER
GEODETIC LATITUDE, DEG
80
60
40
20
0
20
40
60
90 100 120 140 160 180 160 140 120 100 80 60 40 20 0 20 40 60 80 90
EAST
WEST
EAST
LONGITUDE, DEG

Figure 9

NASA-S-66-6012 MAY

EFFECT OF MISSING 1ST INJECTION OPPORTUNITY

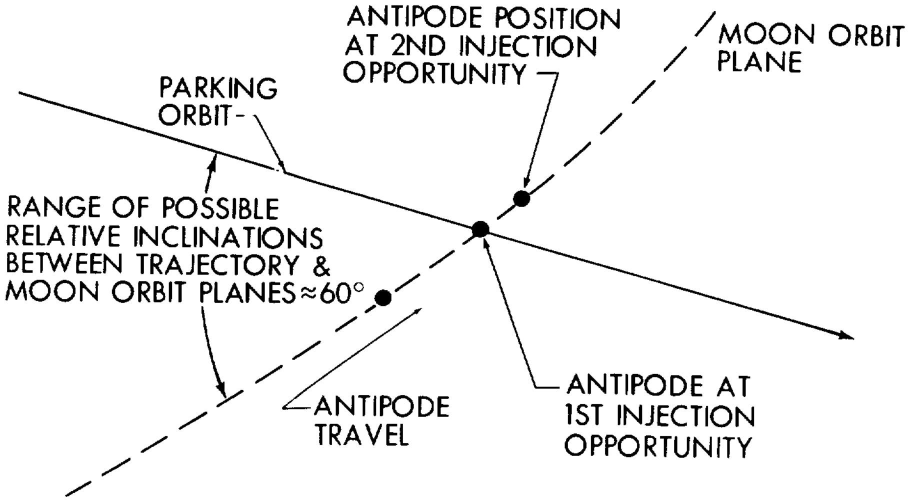

ANTIPODE POSITION AT 2ND INJECTION OPPORTUNITY
MOON ORBIT PLANE
PARKING ORBIT
RANGE OF POSSIBLE RELATIVE INCLINATIONS BETWEEN TRAJECTORY & MOON ORBIT PLANES≈60°
ANTIPODE TRAVEL
ANTIPODE AT 1ST INJECTION OPPORTUNITY

Figure 10

NASA-S-66-6013 MAY

INJECTION OPPORTUNITY

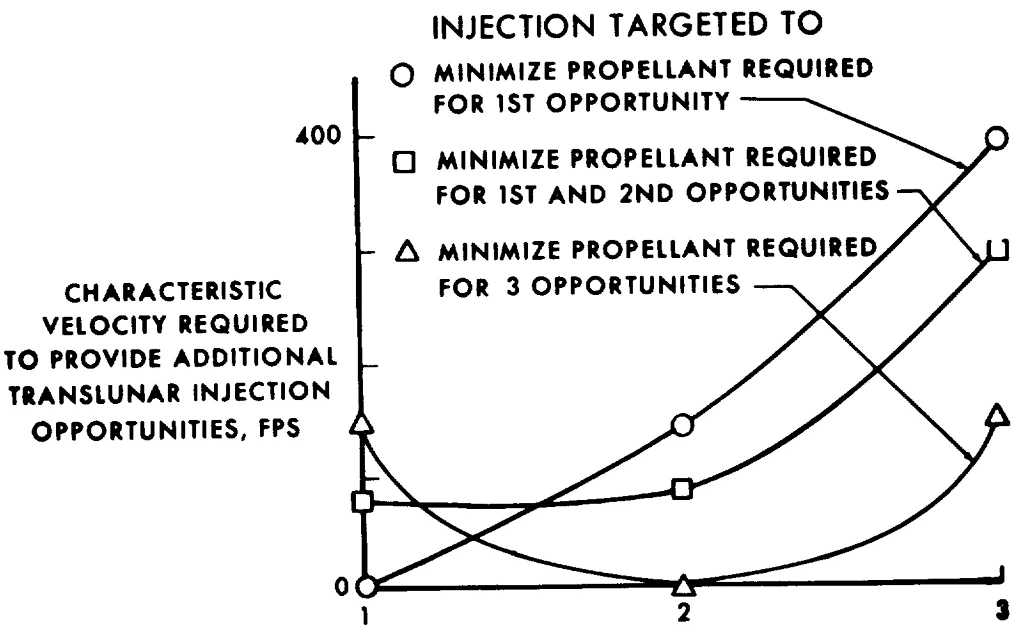


Figure 11

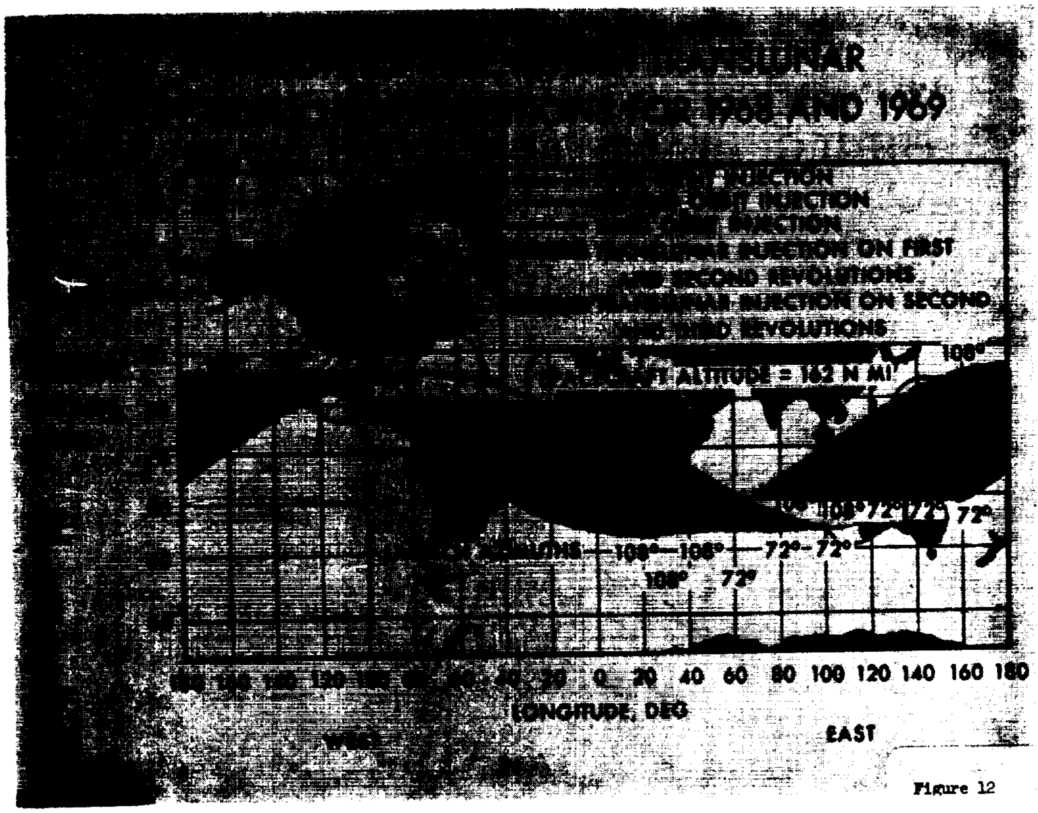

…SLUNAR
…FOR 1968 AND 1969
…INJECTION
…INJECTION
…INJECTION
…INJECTION ON FIRST AND SECOND REVOLUTIONS
…INJECTION ON SECOND AND THIRD REVOLUTIONS
…ALTITUDE = 162 N MI
108°
108° 72° 72° 72°
…108°—108°—72°—72°
108° 72°
0
20
40
60
80
100
120
140
160
180
LONGITUDE, DEG
EAST

Figure 12

NASA-S-66-5022 MAY 31

EFFECT OF INJECTION WINDOW ON TRANSLUNAR TRAJECTORY GEOMETRY

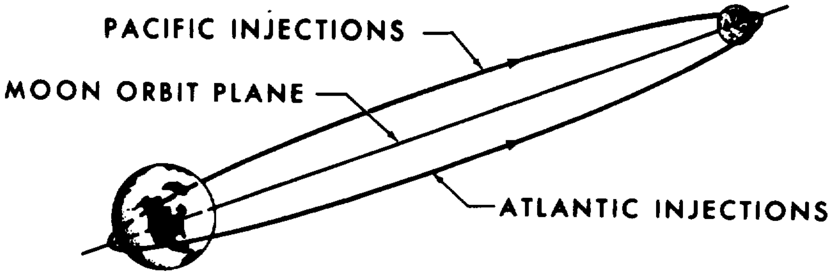

PACIFIC INJECTIONS
MOON ORBIT PLANE
ATLANTIC INJECTIONS

Figure 13

NASA-S-66-6390 JUN

EFFECT OF INJECTION WINDOW ON LUNAR APPROACH CONDITIONS

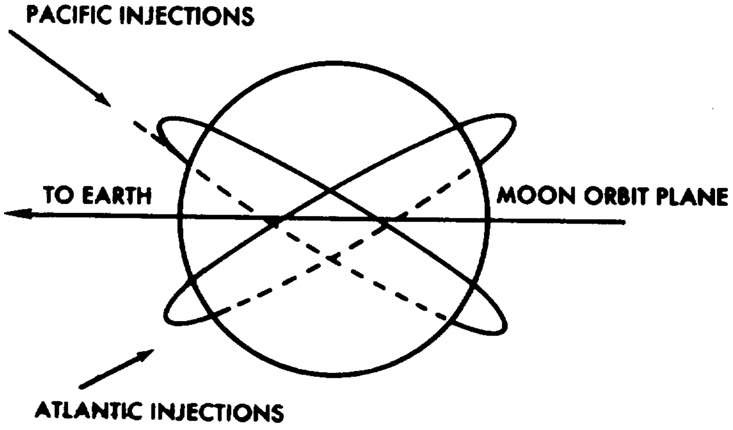

PACIFIC INJECTIONS
TO EARTH
MOON ORBIT PLANE
ATLANTIC INJECTIONS

Figure 14

NASA-S-66-6388 JUN

FREE RETURN TRAJECTORY

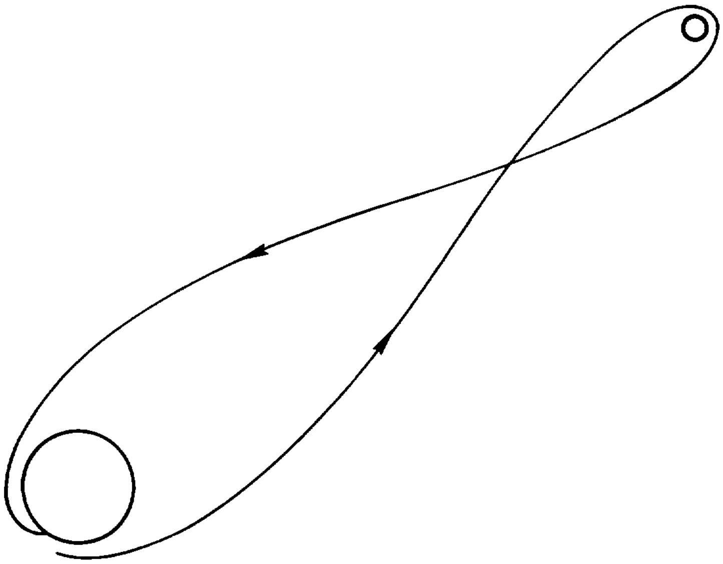


Figure 15

NASA-S-66-6007 MAY

PERILUNE POSITION FOR DIFFERENT TRANSLUNAR TRANSIT TIME
(ALTITUDE OF PERILUNE ≈80 NM)

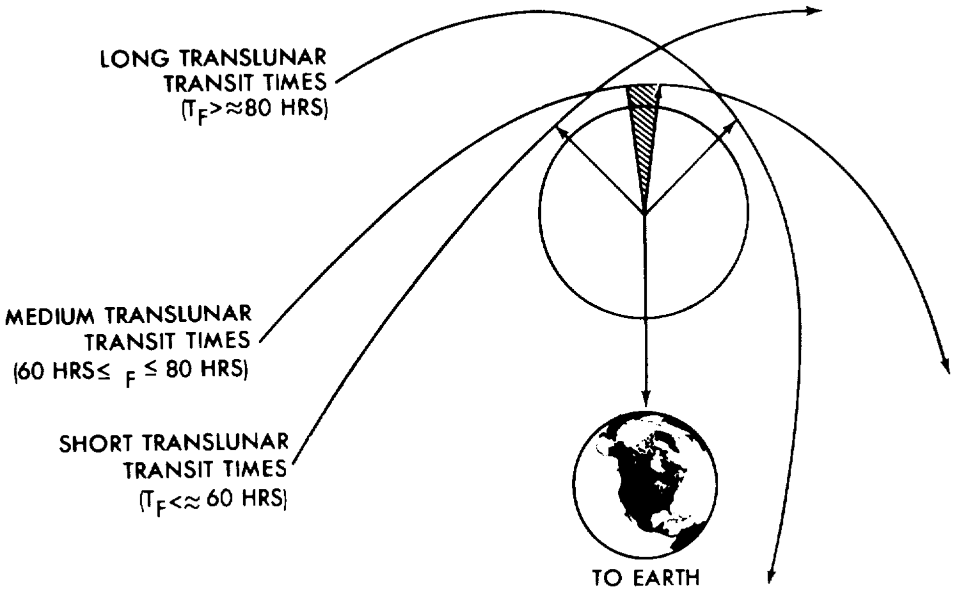

LONG TRANSLUNAR TRANSIT TIMES (T_F>≈80 HRS)
MEDIUM TRANSLUNAR TRANSIT TIMES (60 HRS≤ $_F$ ≤ 80 HRS)
SHORT TRANSLUNAR TRANSIT TIMES (T_F<≈ 60 HRS)
TO EARTH

Figure 16

NASA-S-66-5010 MAY 31

LUNAR APPROACH GEOMETRY OF FREE-RETURN TRAJECTORIES

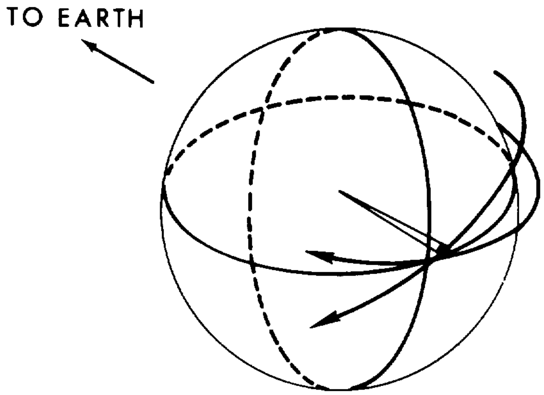


Figure 17

NASA-S-66-5015 MAY 31

EFFECTIVENESS OF PLANE CHANGE AT LUNAR ORBIT INSERTION ON INCREASING ACCESSIBLE LANDING SITES

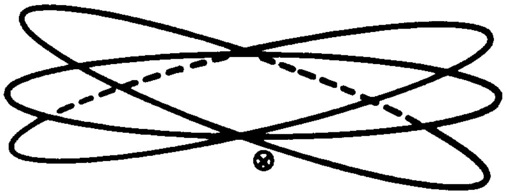


Figure 18

NASA-S-66-5014 MAY 31

RELATIVE TRAJECTORY GEOMETRY OF LUNAR ORBIT INSERTION MANEUVER

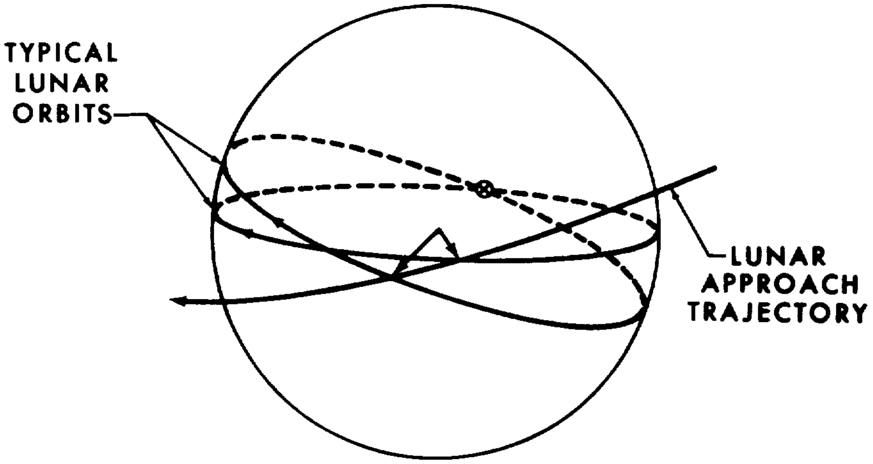

TYPICAL LUNAR ORBITS
LUNAR APPROACH TRAJECTORY

Figure 19

NASA-S-66-6393 JUN

GEOMETRY OF LUNAR ORBIT INSERTION

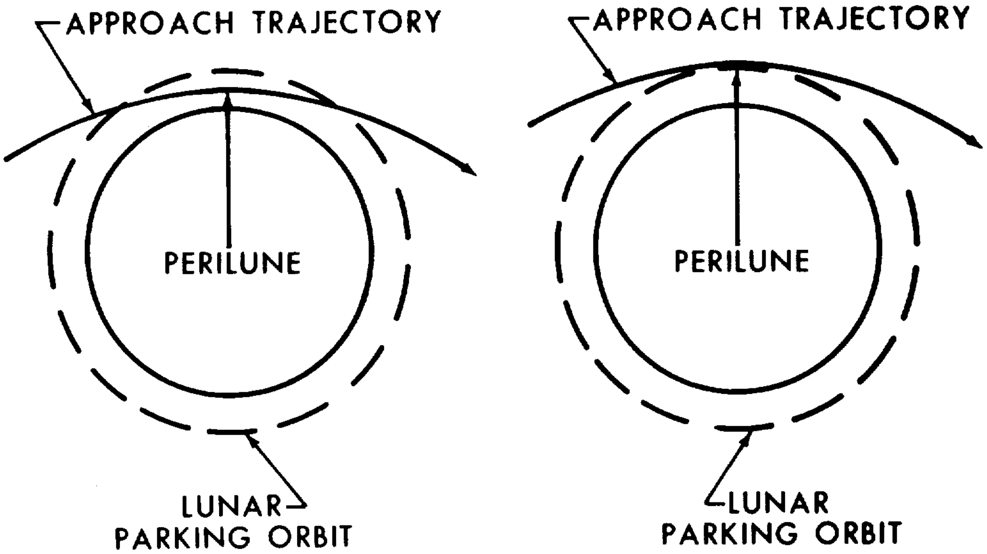

APPROACH TRAJECTORY
PERILUNE
LUNAR PARKING ORBIT
APPROACH TRAJECTORY
PERILUNE
LUNAR PARKING ORBIT

Figure 20

NASA-S-66-6022 MAY

HYBRID FLIGHT PLAN

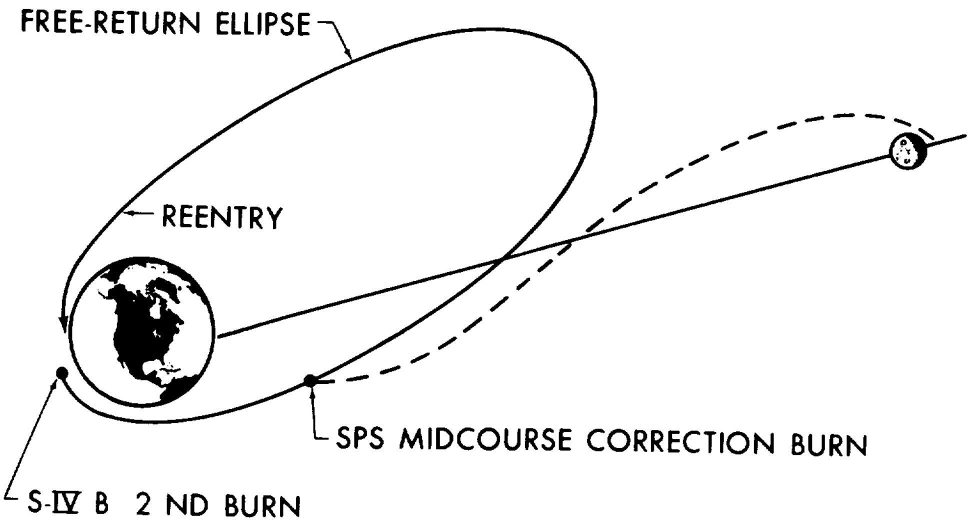

FREE-RETURN ELLIPSE
REENTRY
SPS MIDCOURSE CORRECTION BURN
S-IV B 2 ND BURN

Figure 21

NASA-S-66-5013 MAY 31

EFFECT OF NON FREE RETURN TRAJECTORIES ON 100% ACCESSIBLE AREAS

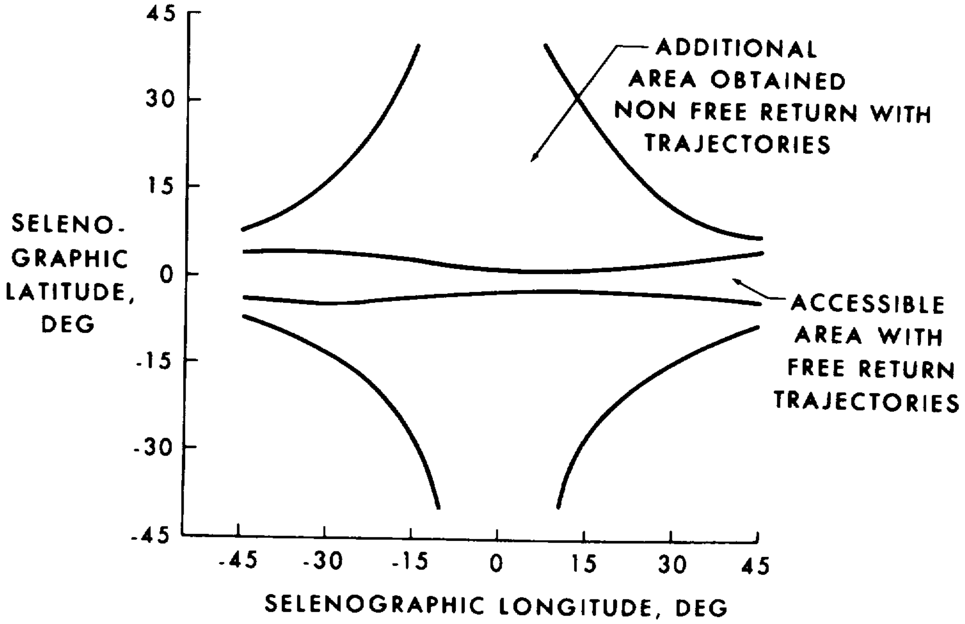


Figure 22

NASA-S-66-6459 JUN

ROTATION OF THE LUNAR LANDING SITE RELATIVE TO LUNAR PARKING ORBIT PLANE

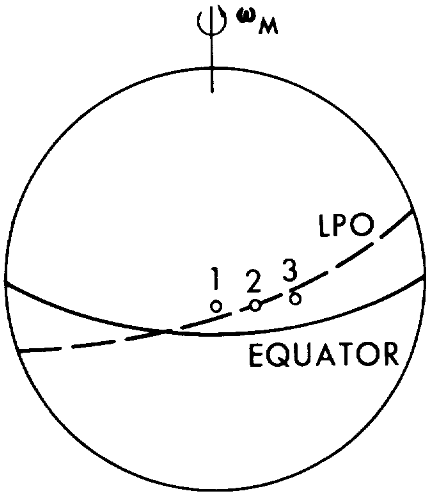

ω_M
LPO
1
2
3
EQUATOR

Figure 23

NASA-S-66-5016 MAY 31

EFFECT OF LAUNCH AZIMUTH CONSTRAINT

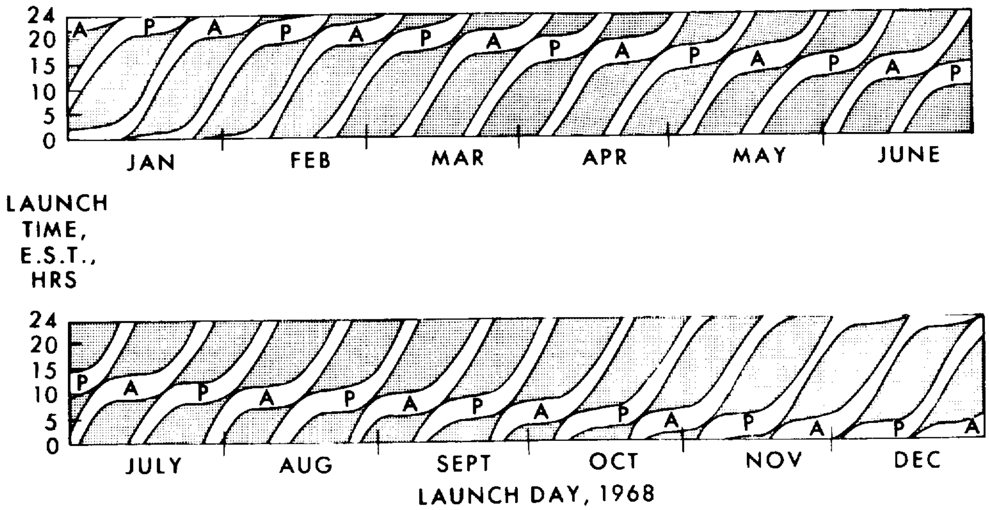


Figure 24

NASA-S-66-5938 JUL 1

LAUNCH AZIMUTH CONSTRAINTS

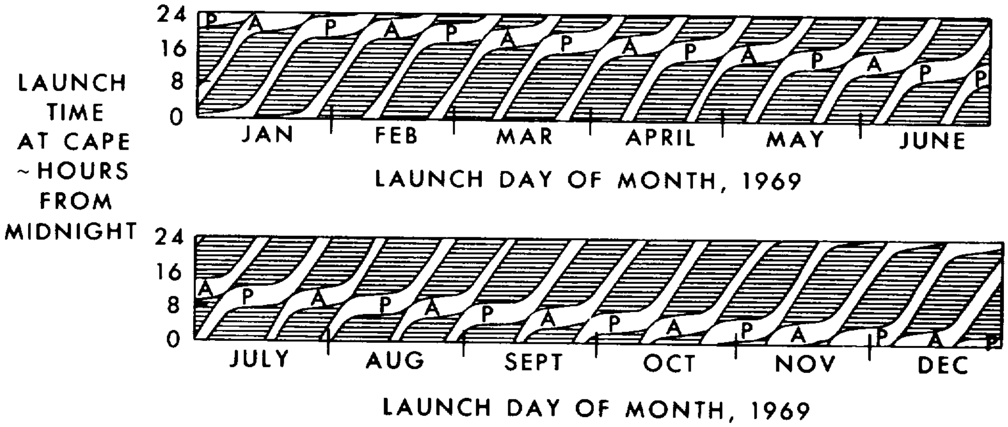
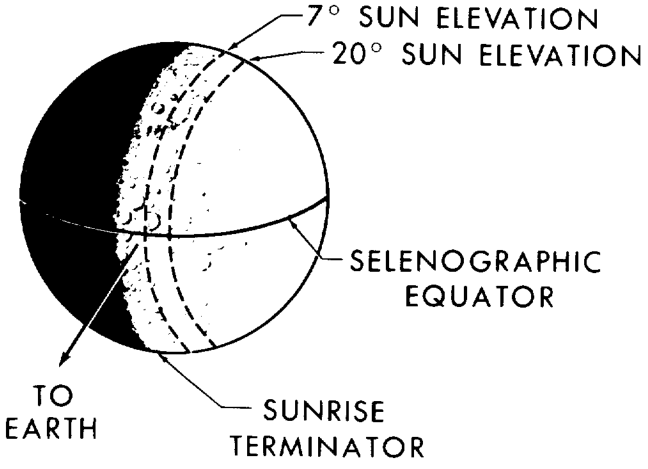


Figure 24A

NASA-S-66-5011 MAY 31

SUN LIGHT CONDITIONS FOR LUNAR LANDING



7° SUN ELEVATION
20° SUN ELEVATION
SELENOGRAPHIC EQUATOR
TO EARTH
SUNRISE TERMINATOR

Figure 25

NASA-S-66-5008 MAY 31

EFFECT OF LIGHTING CONSTRAINT ON AVAILABLE LANDING DAYS

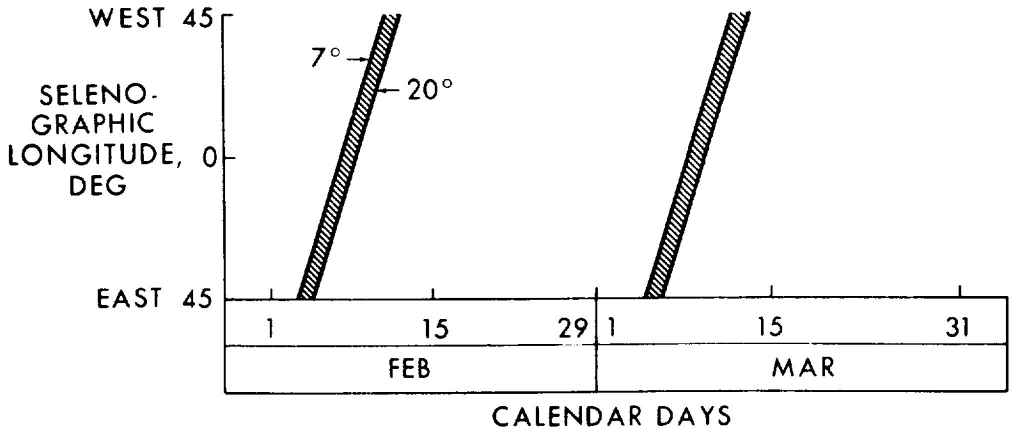


Figure 26

NASA-S-66-5021 MAY 31

LUNAR LANDING OPPORTUNITIES

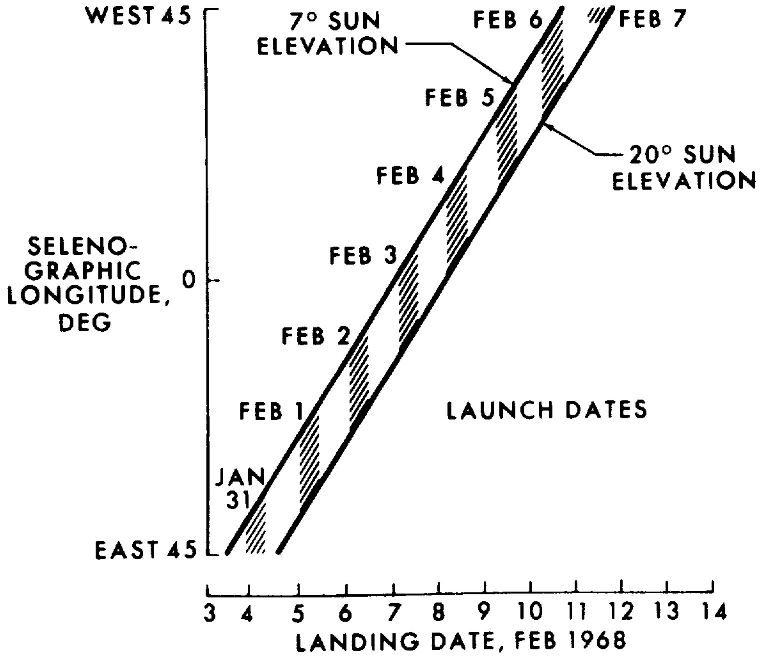


Figure 27

NASA-S-66-5941 JUL 1

LUNAR LANDING OPPORTUNITIES

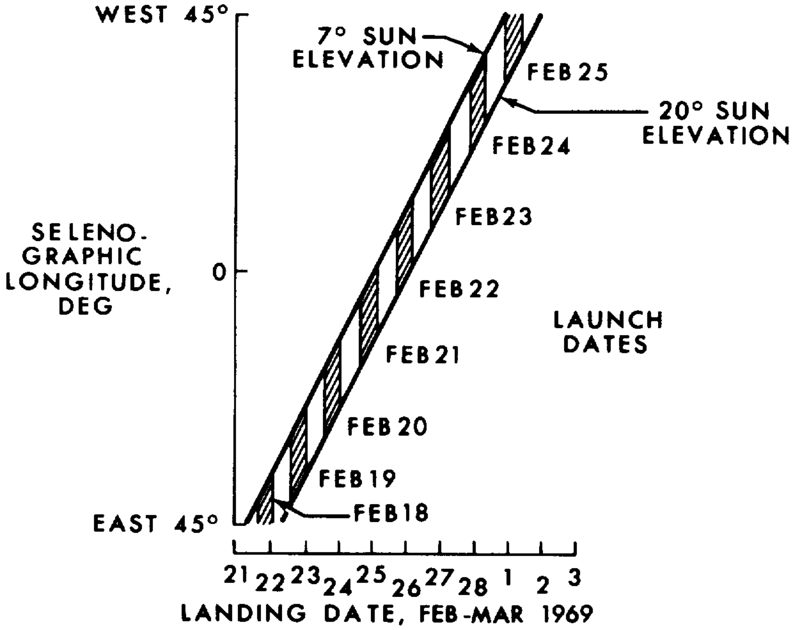


Figure 27 A

NASA-S-66-5017 MAY 31

ACCESSIBLE LUNAR LANDING AREA FOR FEB 1, 1968

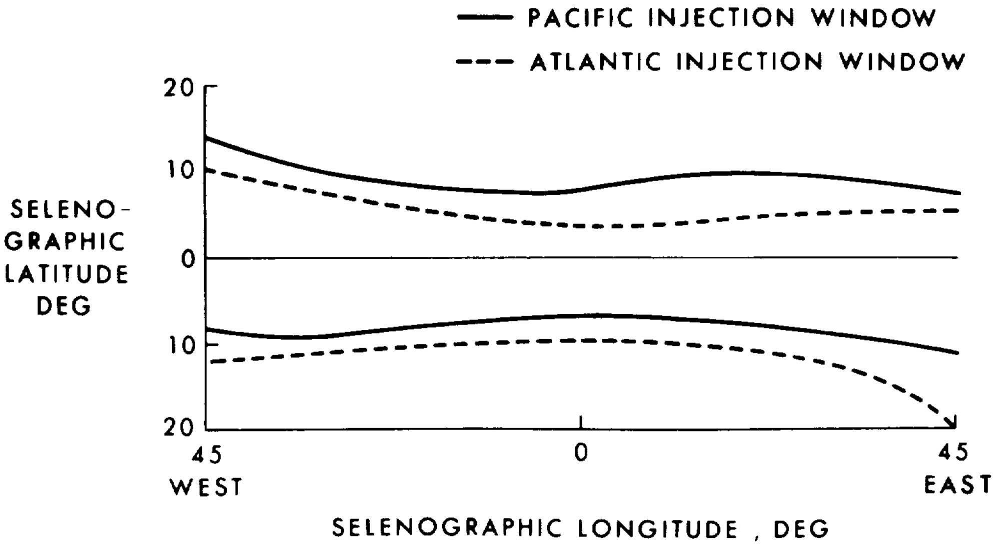


Figure 28

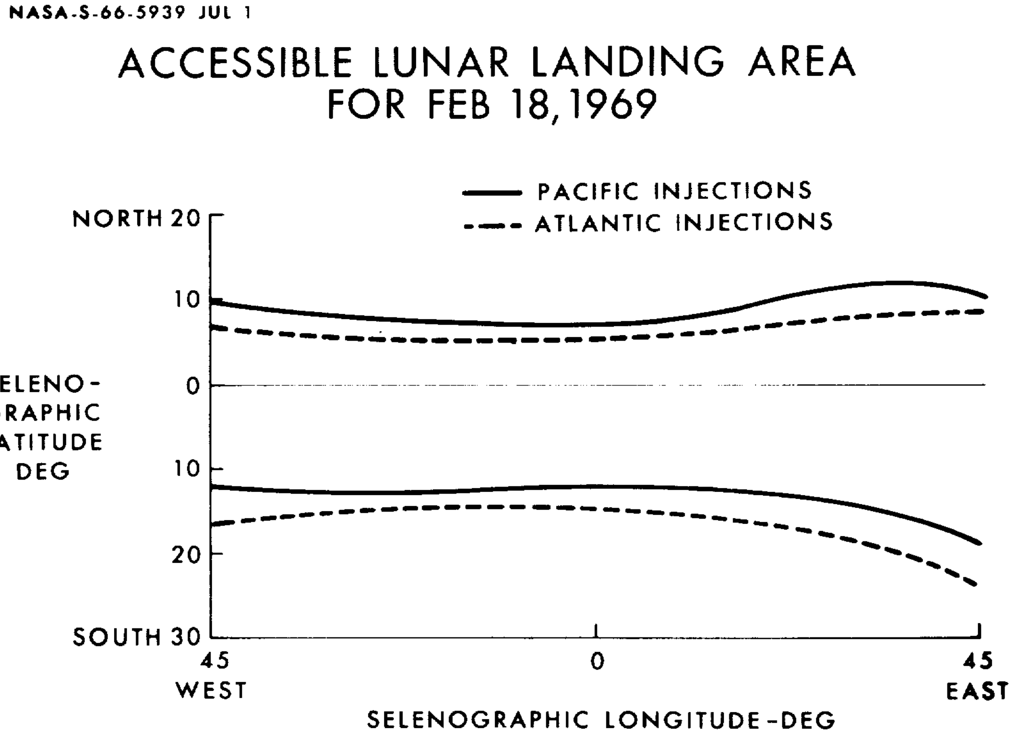


Figure 28 A

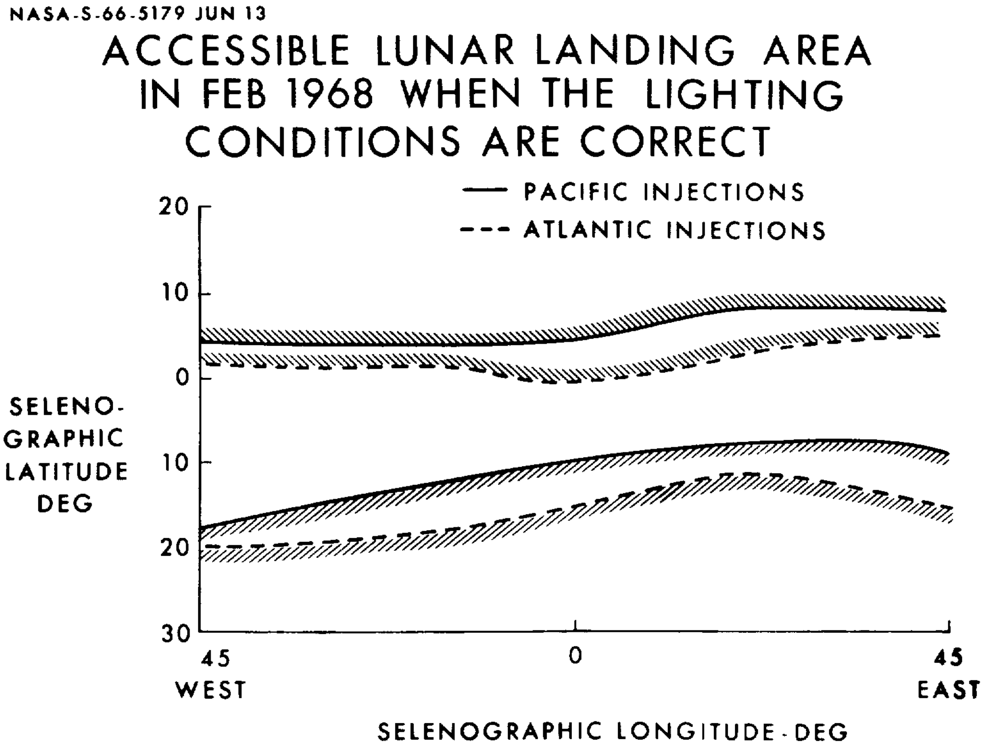


Figure 29

NASA-S-66-5940 JUL 1

ACCESSIBLE LUNAR LANDING AREA IN FEB 1969 WHEN THE LIGHTING CONDITIONS ARE CORRECT

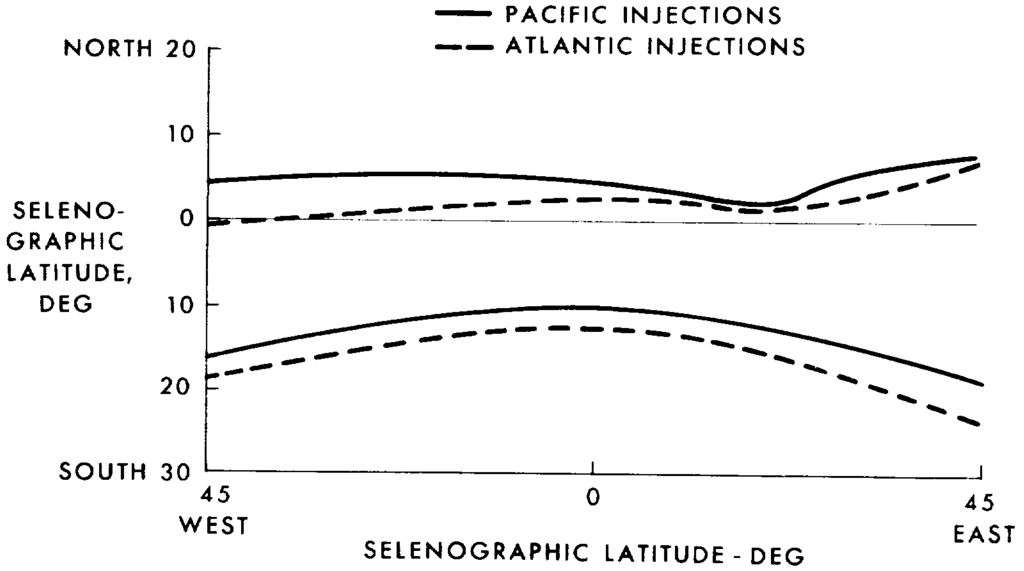


Figure [illegible]A

NASA-S-66-5182 JUN

100% LUNAR SURFACE ACCESSIBILITY FOR 1968

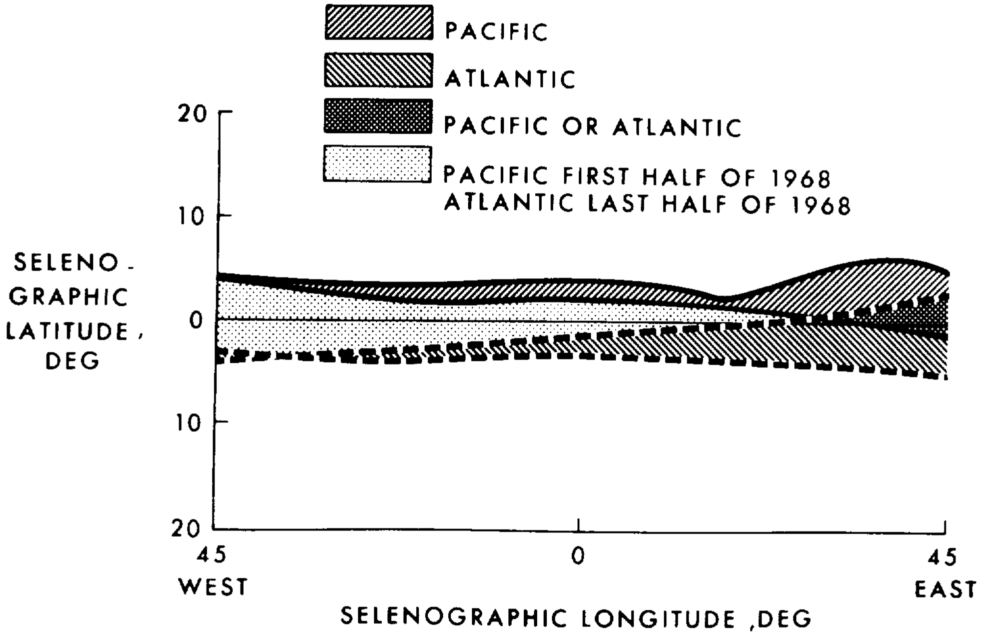


Figure 30

NASA-S-66-5943 JUL 1

100% LUNAR SURFACE ACCESSIBILITY FOR 1969

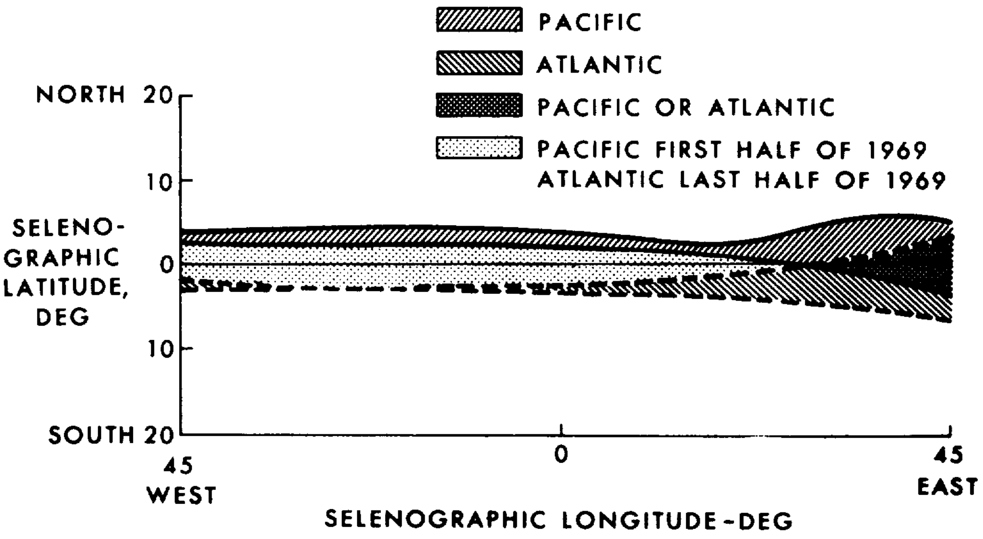


Figure 30 A

NASA-S-66-5005 MAY 31

APOLLO LAUNCH OPPORTUNITIES IN 1968

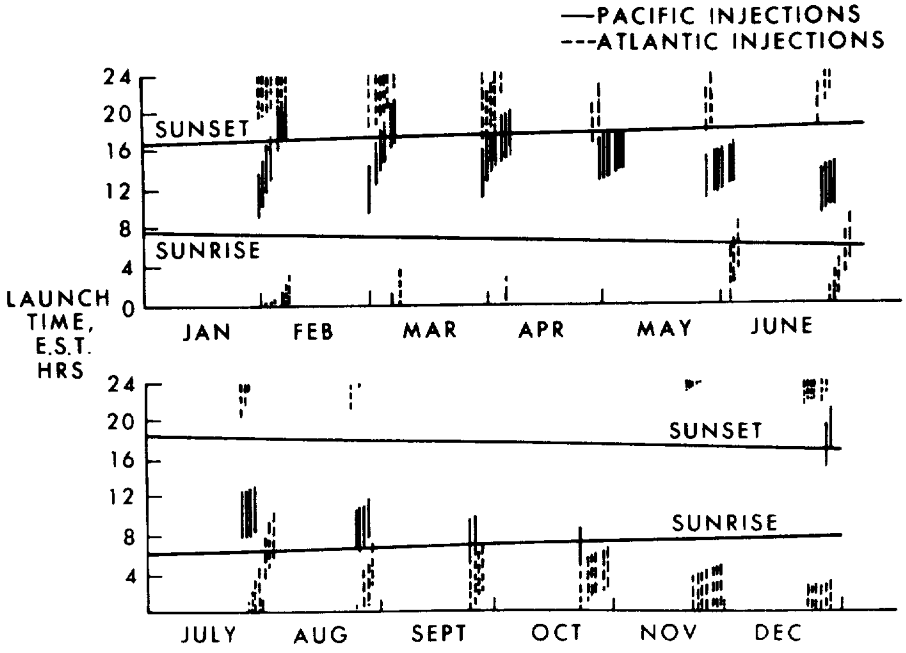


Figure 31

NASA-S-66-5937 JUL 1

APOLLO LAUNCH OPPORTUNITIES
IN 1969

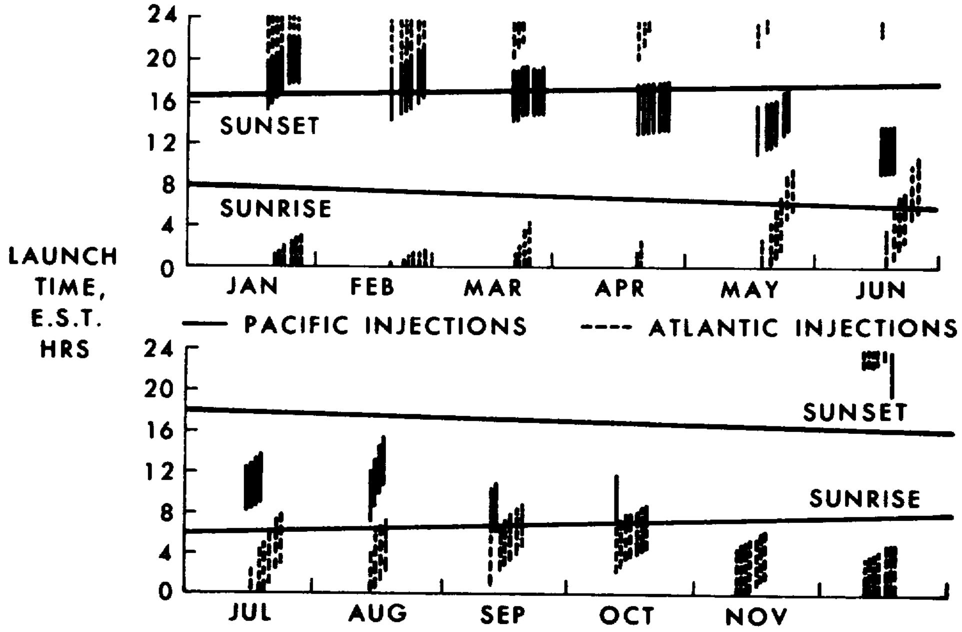


Figure 31 A

NASA-S-66-5006 MAY 31

APOLLO LAUNCH OPPORTUNITIES IN 1968
THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW

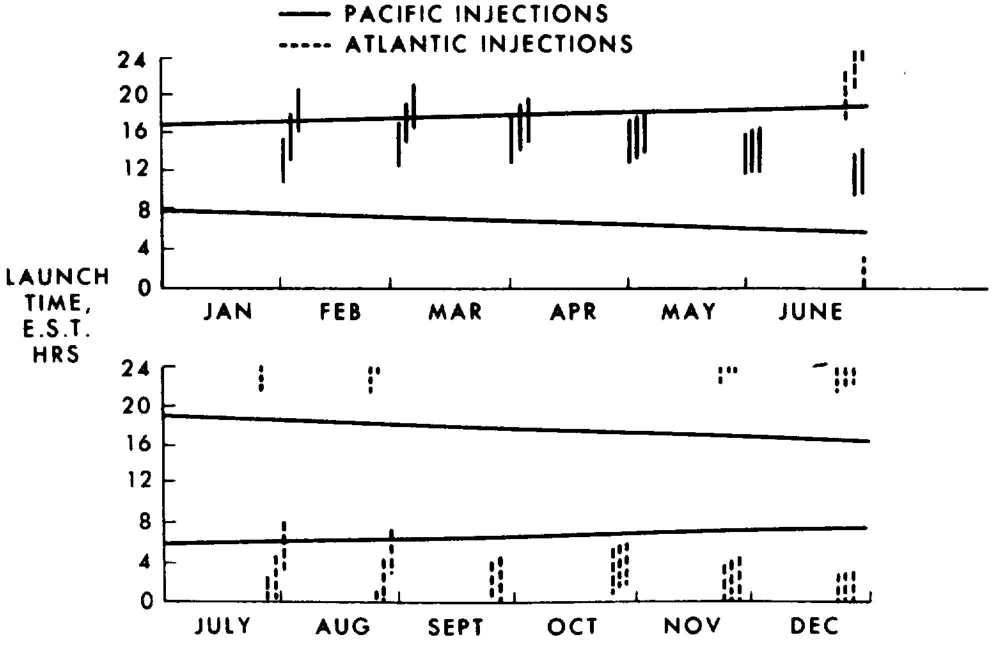


Figure 32

NASA-S-66-5944 JUL 1

APOLLO LAUNCH OPPORTUNITIES IN 1969 THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW FOR ORBITER B SITES

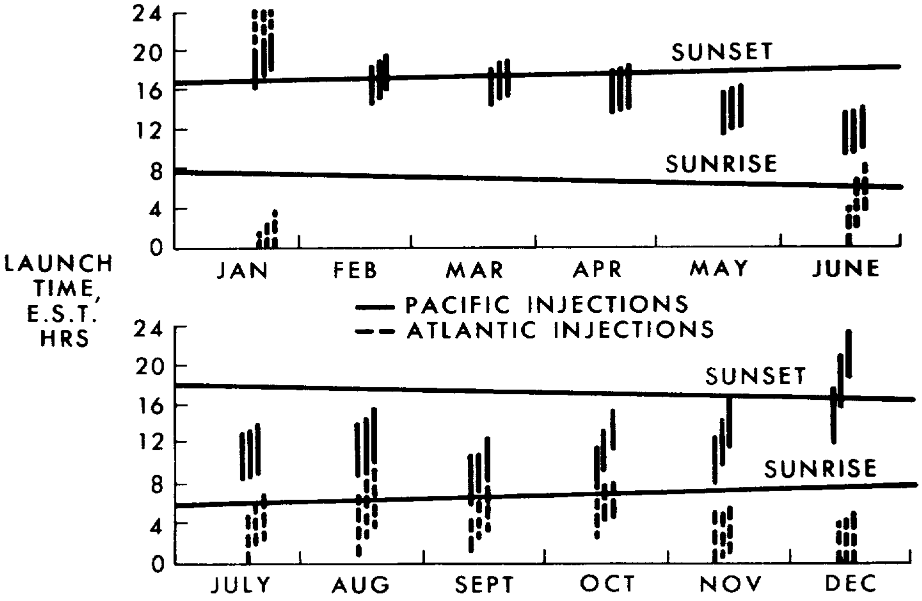


Figure 32 A

NASA-S-66-5206 JUN

APOLLO LAUNCH OPPORTUNITIES IN 1968 THAT PROVIDE A 1-3-5 DAY LAUNCH WINDOW FOR ORBITER B SITES

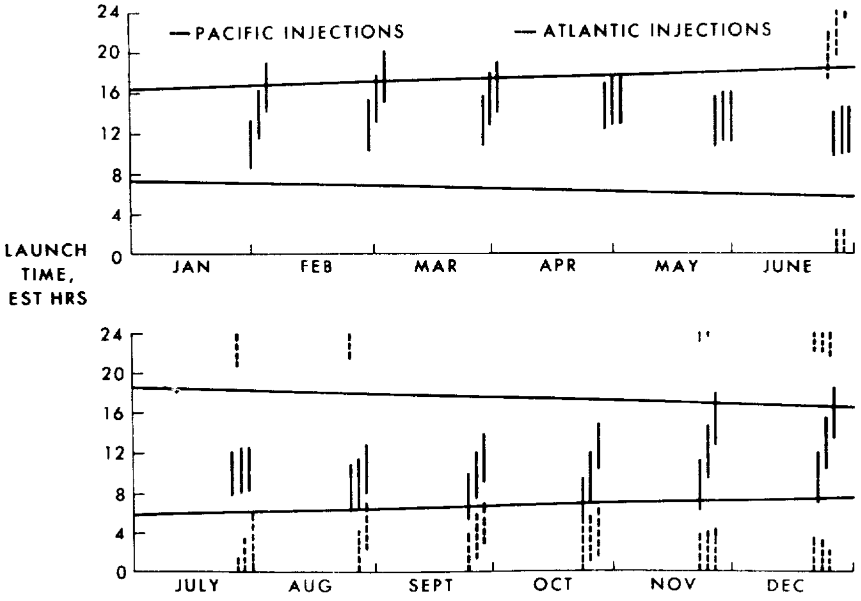


Figure 33

APOLLO LUNAR MODULE LANDING STRATEGY

Donald C. Cheatham
Assistant Chief for
Engineering and Development
Guidance and Control Division

Floyd V. Bennett
Assistant Chief
Theoretical Mechanics Branch
Guidance and Control Division

APOLLO LM LANDING STRATEGY

- INTRODUCTION
- STRATEGY CONSIDERATIONS
  - Spacecraft Systems
    - Guidance and Control System
    - Landing Radar System
    - LM Window System
    - Descent Propulsion System
  - Mission Landing Position Requirement
- POWERED DESCENT DESIGN
  - Braking Phase
    - Objectives and Constraints
    - Ignition Logic
    - Guidance with Limited Throttle
    - Landing Radar Updating
    - Delta V Budget
    - Descent Guidance Monitoring
    - Summary of Braking Phase
  - Final Approach Phase
    - Objectives and Constraints
    - Determination of Hi-gate
    - Parameter Tradeoffs
    - Redesignation Footprint
    - Landing Point Designator Utilization

    - Delta V Budget
    - Summary of Final Approach Phase
  - Landing Phase
    - Objectives and Constraints
    - Nominal Trajectory
    - Delta V Budget
- LUNAR LANDING TOUCHDOWN CONTROL
  - Objective and Constraints - Modes of control
  - Sequence of events
  - Descent engine shut-off
  - Manual control of landing velocities
  - Automatic control of landing velocities
- ABORT AFTER TOUCHDOWN
- SUMMARY

APOLLO LUNAR MODULE LANDING STRATEGY

1.0 INTRODUCTION

The landing of the Lunar Module (LM) upon the surface of the moon will be the climax of the Apollo mission, although the importance of the return phases is not to be de-emphasized. The LM landing approach will be the first time that the complete LM system will have been operated in the lunar environment. This also will be man's initial face-to-face encounter with the exact nature of the terrain in the landing area and of the problems of visibility as they may affect the ability to land the LM; although, these aspects of the landing will be simulated many times in fixed-based simulators and partial preflight simulators. These simulations are extremely important in the preparations for the mission; but only after the mission is completed will it be known how adequate the simulations have been.

Considering the entire LM descent after separation from the Command Module in lunar orbit, a theoretical landing maneuver could consist of a Hohmann transfer impulse on the back side of the moon with a delta V, or change in velocity, of 109 ft/sec, followed 180° later by an impulsive velocity change of about 5622 ft/sec as the LM approaches the lunar surface, as illustrated in figure 1. The flight path angle in the final portion of the approach would be zero degrees. Such a theoretical approach would require infinite thrust-to-weight ratio by the descent engine. This, of course, is an impossible and impractical approach. A finite thrust-to-weight ratio of the descent engine must be used and the approach path must account for lunar terrain variations and uncertainties in the guidance system. Since lunar terrain variations of as much as $\pm$ 20,000 ft. could be expected, and, also, uncertainties in the value of the lunar reference radius, coupled with guidance dispersions, could add another 15,000 ft. to the uncertainty, a conservative safe value of 50,000 ft. was chosen as a pericynthion altitude. From a performance standpoint, the choice of 50,000 ft. as opposed to either 40,000 or 60,000 ft. was quite arbitrary because the difference from the standpoint of fuel requirements was very slight, as indicated in figure 2. The initial thrust-to-weight of the LM descent engine will be about three-tenths. Combining this thrust-to-weight with a perigee altitude of 50,000 ft. leads to the descent profile, as shown in figure 3. The separation and Hohmann transfer maneuver requires slightly less delta V due to the pericynthion altitude increase. The powered descent portion approaching the landing area, however, requires a delta V of 5925 ft/sec, which is a considerable increase over the infinite thrust requirement. A scaled trajectory profile of this theoretical LM powered descent is shown in figure 4, indicating that the entire descent takes approximately 220 n. m. The LM velocity and attitude is shown periodically along the flight profile. This trajectory has the predominant characteristics of a low flat profile terminating with a flight path angle of about 9 degrees. An obvious feature is that the crew, considering the location of the LM window, never have the opportunity to see where they are going. They can look either directly up, or, if the LM is rotated about its thrust axis, can look down at the surface, but they are never able to see in the direction they are going. If the crew is to perform any assessment of the landing area or out-the-window safety of flight during the approach, it is obvious that the latter portion of the trajectory must be shaped so that a different attitude of the LM can be used during the approach. Shaping the trajectory away from the fuel optimum approach will result in a penalty in fuel requirements. Both the amount of time the crew will require to assess the landing area, and the range from which the landing area can be adequately assessed must be traded off against the amount of fuel involved in the penalty of the shaping. It soon becomes obvious that a strategy is needed that will trade off the system capabilities of the spacecraft and the crew capabilities against the unknowns of the lunar environment encountered during the descent from the orbit, in order to insure that proper utilization of the onboard systems can be made to greatest advantage. The development of this strategy, then, is the subject of this paper.

2.0 STRATEGY CONSIDERATIONS

The LM landing strategy can be defined as the science and and art of spacecraft mission planning exercised to meet the lunar environmental problems under advantageous conditions. In order to plan strategy, the objectives, the problems to be faced, and the characteristic performance of available systems need to be well known. As indicated in figure 5, the objectives of the LM landing planning strategy are to anticipate the lunar environmental problems and to plan the landing approach so that the combined spacecraft systems, including the crew, will most effectively improve the probability of attaining a safe landing. The major factors that must be considered in this strategy are the problems brought about by the orbital mechanics of the landing maneuver, the limitations of the spacecraft systems (including limitations in fuel capacity and payload capability), and the constraints of the lunar environment (including terrain uncertainties, visibility, and determination of suitable landing positions). The orbital mechanics aspects have been discussed in the preceeding section. The lunar environmental constraints will be discussed in a subsequent section. The remainder of this section is concerned with descriptions of the spacecraft systems and the mission landing position requirements. Although all of the LM systems are important to attain the lunar landing, those affecting the strategy are (a) the guidance and control system, (b) the landing radar, (c) the spacecraft window, and (d) the descent propulsion system.

Spacecraft Systems

Guidance and control system - The guidance and control system is important to the landing strategy in that it has a direct effect upon the area over which the landing may be accomplished and on the problems of landing at a desired point. The functional description and accuracies of this system have been discussed in a preceeding paper. The effect of the guidance, navigation, and control system of the LM on the landing begins with navigation in the lunar orbit. The accuracy of this navigation, whether performed by the onboard system or by the Manned Space Flight Network, determines the uncertainties at the start of the powered descent. Assuming that the guidance system will be updated by landing radar to eliminate the altitude dispersions, the landing dispersions will be a function of the initial condition uncertainties brought about from lunar orbit navigation coupled with the inertial system drift during the powered descent. A summary of the guidance system capability for attaining a given landing point on the moon is presented in figure 6a and the associated assumptions in figure 6b. Both the MSFN and the spacecraft onboard navigation in lunar orbit are considered. The Apollo system specification of a landing CEP of 3000 ft. is met in either case when the inertial system performs within specification.

The 3σ landing dispersion ellipses are shown in figure 7 for cases where the lunar orbit navigation was done by the MSFN and also onboard the CSM. The ellipses are quite similar with the major axis for the MSFN case being slightly shorter and the minor axis for the MSFN being slightly longer than that for the case utilizing CSM onboard navigation. A special case in which the downrange distance was allowed to be unconstrained is also shown on figure 7. In this case the downrange or major axis of the ellipse is primarily a function of the thrust uncertainties of the fixed-throttle position of the descent engine that will be discussed subsequently. The crossrange axis is equal to that of the 3σ ellipses for guidance to a specific point and is determined by the method of lunar orbit navigation.

Landing radar system - The control of the LM during the descent to the surface can be provided automatically through steering commands generated by the guidance system and also manually by the crew by inputs through an attitude controller. The primary control system stabilization utilizes digital autopilot mode of the guidance computer. Figure 8 shows the attitude thruster firing combinations to create control moments. The engines are located on an axes system rotated about the LM descent engine thrust area 45° from the spacecraft axes. They are operated as control couples for three-axis attitude control. As can be seen in figure 8, two pairs of control couples are available for each axis. The method of providing translational control while in the hovering condition is to tilt the spacecraft by means of the attitude control system. This produces a lateral component of acceleration from the descent engine thrust in the desired direction which is stopped by returning to vertical and reversed by tilting in the opposite direction. During the descent the attitude control system is also coupled to a slow moving gimbal actuator system of the descent engine to enable a means of trimming the descent engine thrust direction so that it passes through the LM center of gravity. The trimming system reduces undesirable torques from the descent engine in order to conserve RCS propellant. The LM landing radar system is important in landing strategy. As indicated earlier, it is used to eliminate the guidance system altitude dispersions and, also, the uncertainties of knowing the altitude from the lunar surface prior to beginning the descent. The LM landing radar is a 4-beam dopple system with the beam configuration shown in figure 9. The center beam measures the altitude, and the other three beams measure the three components of velocity. Two positions of the landing radar antenna provide both altitude and velocity measurements over a wide range of spacecraft attitudes. The first antenna position is tilted back from the thrust axis by approximately forty-three degrees so that the altitude beam will be nearly vertical during the early portions of the descent and, hence, will still provide accurate altitude information. As the LM approaches the landing maneuver, the antenna is physically switched to the second position making the altitude beam parallel to the X-axis of the LM. The landing radar will begin to provide altitude measurements at an approximate altitude of 40,000 ft. These altitude measurements will be used to update the inertial system starting at an altitude of about 25,000 ft. The radar velocity updates will begin at approximately 15,000 ft. The landing radar accuracy is given in figure 10.

LM window system - The LM window, although perhaps not normally considered a system, is a very important part of the landing strategy because it is through this window that the crew must observe the landing area to confirm the adequacy of the surface for touchdown. The physical configuration of the LM window is shown in figure 11. This photograph was made from within the LM cockpit showing the left hand, or the command pilot's, window. The window is triangular in shape and skewed so that it provides maximum viewing angles for the landing approach maneuver. Although the window is not large in size, the pilot's eye position is normally very close to the window so that the angular limits provided are quite wide. These angular limits are displayed in figure 12, showing the limits as viewed from the commander's design eye position. The plot shows the azimuth and elevation variations of possible viewing limits referenced from a point where the pilot would be looking dead ahead, with respect to LM body axes (parallel to the Z-body axis), for the zero point. It is possible for the pilot to see downward at an angle of about 65° from the normal eye position and to the left side by approximately 80°. If the pilot moves his head either closer to the window, or further back, these limitations change slightly.

The guidance system is coupled with the window system through grid markings so that the pilot can observe the intended landing area by aligning his line-of-sight with the grid marking according to information displayed from the guidance system. Figure 11 in addition to showing the window system, shows the location of the Display and Keyboard, which among other things provide digital readout information from the guidance system. The procedures for utilizing these integrated systems for landing site designation and redesignation will be discussed later in this paper.

Descent propulsion system - The descent engine is an extremely important system to the design of the LM descent strategy. Initially, the descent engine was capable of being throttled over a range from 10 to 1. Design considerations, however, have made it necessary to limit the throttle capability to that shown in figure 13. This figure shows that at the start of powered flight, there is an upper fixed position of the throttle which would nominally provide about 9700 lb. of thrust. As long as the throttle is maintained in this fixed position, thrust magnitude will vary according to the nominal solid line. At the start of the powered flight, there is expected to be approximately ± 1 percent uncertainty in the thrust at this fixed-throttle setting. The uncertainty grows up to ± 2 percent after approximately 300 seconds of fixed-throttle usage. The descent engine is always throttleable, in the region of 6300 lb. of thrust, to approximately 1050 lb. of thrust. The change from a fully throttleable engine in the upper region of the thrust level to a fixed-throttle position affects the guidance procedures during the initial powered descent, as will be explained later.

Mission Landing Position Requirement

Important strategy considerations are the types of requirements that are placed on the landing position, as indicated in figure 14. The first consideration is a requirement to land at any suitable point within a specified area, with the implication that the area could be quite large. Obviously, if the area is large enough, the requirements on the guidance system would be diminished considerably. The second type of requirement is that of landing at any suitable point within a reasonably small area, constrained in size primarily by the guidance dispersions. This would, of course, dictate that the size of the area chosen will be compatible with the capabilities of the guidance and navigation system. The third consideration is that of landing at a prespecified point, such as landing with 100 ft. of the position of a surveyor spacecraft, or perhaps another type of spacecraft. It is obvious that this latter consideration imposes the greatest requirements on the strategy and also the guidance system, and would require some means of establishing contact with the intended landing position during the approach. The present strategy is primarily based upon the second consideration, that of landing in areas of the size compatible with the guidance system dispersions. If, however, the landing area can be increased in size to the point that downrange position control is not of primary importance, the associated strategy is not greatly different than that for the requirement assumed because the trajectory shaping requirements would be the same for the terminal portion of the trajectory. The subsequent discussions of this paper will be based primarily upon a landing area size compatible with guidance system dispersions.

3. POWERED DESCENT DESIGN

After consideration of all the trade off's that could be identified as worthy of consideration during the LM powered descent, a three-phase trajectory design logic was chosen. The logic of this three-phase trajectory design will be discussed in the subsequent sections, but, the general logic is indicated in figure 15. The first phase following powered descent initiation at 50,000 ft. is termed the Braking Phase. This phase is terminated at what is called a Hi-gate position. The second phase is termed the Final Approach Phase, and is terminated at what is called the Lo-gate position, the start of the Landing Phase. The total trajectory covers on the order of 250 n.m. The logic of the braking phase is designed for the efficient reduction of velocity. That is, since there is no necessity for pilot visibility of the landing area in this phase, the attitudes can be chosen so that the spacecraft would have efficient utilization of descent engine thrust for reducing velocity. During the final approach phase, the trajectory is shaped to allow an attitude from which the pilot can visually acquire and assess the landing site. An additional requirement met by this phase is to provide the pilot with a view of the terrain at such a time that he can confirm the flight safety of the trajectory prior to committing to a landing. The landing phase is flown very much as a VTOL type of aircraft would be flown on the earth to allow the pilot vernier control of the position and velocities at touchdown. The attitude chosen is flown so as to provide the crew with visibility for a detailed assessment of the landing site. The scaled profile of the design descent trajectory is shown in figure 16 a) and b), and includes an indication of the spacecraft attitude at various milestones along the trajectory. The final approach and landing phases together cover only about 2 per cent of the total trajectory range, although the time spent within these phases will be about 30 per cent of the total time. The following sections will discuss in detail the logic of the design of the three phases and will summarize the delta V budget for the descent.

Braking Phase

Objectives and constraints - The objective of the braking phase, as indicated in figure 17, is to provide efficient reduction of the horizontal velocity existing at the initiation of the powered descent. During most of this phase, the altitude is high enough so that the pilot does not have to worry about the terrain variations, and he can conduct the reduction in velocity at attitudes that allow great efficiency. The major constraint of this trajectory phase is limitations imposed by the fixed-throttle-position thrust of the descent engine. It is desirable to use the maximum thrust of the descent engine as long as possible in order to provide efficient utilization of the fuel. There is, however, an initial segment of the powered descent which is flown at reduced throttle to insure that the descent engine gimbal trim mechanism has nulled out of trim moments due to center-of-gravity offsets.

Ignition Logic - The logic for igniting the descent engine for initiation of the braking phase is as follows. First, the LM state (position and velocity) is integrated ahead in time. Next, the guidance problem for the braking phase is solved, but not implemented, continuously with the advanced LM states as initial conditions. When the guidance solution requires the level of thrust equal to the expected thrust of the fixed-throttle position, see figure 18, that solution is chosen for initiation of braking. Finally, when the LM reaches the position and velocity state that yielded the proper thrust solution, the guidance computer sends the engine on signal to the descent propulsion solution. In order to prevent large moments due to c. g. offset, the engine is ignited at the low 10 percent level, instead of maximum thrust. This level is held for some 28 seconds to trim the engine gimbal through the c.g. before increasing thrust to the maximum, or fixed-throttle, setting. This low level of thrusting is accounted for in the ignition logic.

Guidance with Limited Throttle - The general approach of the braking phase, from the standpoint of the guidance system, is to utilize the same type of guidance equations that are appropriate for the throttled phases which follow. Thus, modifications in the targeting are required to allow for the utilization of the fixed-throttle position during this phase. It is still desired to vary the state vector of the LM from its value at the start of powered descent to the state specified at the hi-gate position of the trajectory. The guidance equations would normally determine the thrust level or acceleration level and attitude required in order to make an efficient change in the state. Prior knowledge of the initial thrust-to-weight of the descent engine allows choice of initial conditions and the guidance equations to be utilized in such a way as to select a time to go for the entire phase that will use the approximate thrust-to-weight of the upper limit of the descent engine. In actual operation, the LM system during this phase will respond to commands of attitude change, but as long as the guidance system is calling for a thrust above 6300 lb., the descent engine will remain in its fixed or upper limit position. If the thrust variation of the descent engine at this fixed throttle position were known exactly, the trajectory could be preplanned to obtain the desired hi-gate state vector. In view of the uncertainties of the descent engine, however, the trajectory must be planned so that the guidance system will begin to call for thrust levels in the region in which the descent engine can be throttled (below 6300 lbs.) prior to reaching Hi-gate position. This is to provide control over the velocities when the Hi-gate position is reached. The logic of this guidance scheme is shown in figure 19. The figure shows the profile of the trajectories as a function of range, and also a profile of the descent engine thrust, both the nominal value and that commanded by the guidance system as a function of range. The nominal thrust-to-weight case is shown first, and the trajectory is essentially preplanned by flying backward from the hi-gate position, first of all, using a thrust in the throttleable range to go back for a period of time; the period of time being determined by the possible magnitude of the uncertainty of the descent engine. This, in effect, determines the fictitious target that can be used in the guidance system in the first portion of the trajectory. The fictitious target is based upon the nominal thrust profile when the descent engine is in the fixed-thrust position. The logic of the guidance is perhaps best explained by comparing the actual value of thrust with that commanded by the guidance system, even though in the upper thrust region the descent engine is not responding to these commands.

Initially, the guidance system is targeted to a fictitious target upstream of the hi-gate state. The nominal thrust-to-weight variation follows the solid line, and the guidance system computes the commanded variation of thrust-to-weight shown on the figure. At an intermediate position, the guidance targeting is switched from that of the fictitious target to that of the hi-gate target. The discontinuity seen in the commanded position has no effect on the system, since, in this region, the descent engine throttle is not responding to the guidance system. If the thrust-to-weight does remain nominal, the commanded thrust-to-weight magnitude will gradually decrease until it is within the region in which the descent engine can be throttled. This will nominally occur at the fictitious target position. The guidance system then has a number of seconds, prior to the hi-gate position, to match both the velocity and the position desired at hi-gate. From hi-gate on, the commanded thrust will be at or below the maximum in the throttleable range. Figure 20 illustrates the thrust profiles (commanded and actual) for low and high thrust-to-weight ratios. In the case of the low thrust-to-weight ratio where the actual value of the thrust is below that of the expected nominal, it is seen that the initial commanded thrust has the same type of variation as the nominal, prior to the switchover point; but, after the switchover point, there is a delay in time and range in getting down to the region where the commanded thrust reaches the throttleable region. This point, then, is only a few seconds prior to hi-gate. The extreme low thrust-to-weight, then, would be that in which the commanded thrust would reach the throttleable region thrust exactly at the time the hi-gate position was reached. For the case where the thrust-to-weight is higher than nominal, the commanded thrust will reach the throttleable position a number of seconds prior to that for nominal thrust. This allows a much longer time to affect the desired velocity condition at the hi-gate position. This, however, means that the region prior to hi-gate is being flown at a much lower thrust-to-weight ratio for a longer period of time than would be desirable from a standpoint of fuel efficiency. This is the case that involves the greatest penalty in fuel. Figure 21 shows the delta V penalty variation due to fixed-thrust uncertainties. The left-hand scale indicates the delta V penalty, the horizontal scale shows the bias time of the fictitious target back from the hi-gate target, and the right-hand scale is the thrust-to-weight uncertainty expressed in ± percentages. The figure indicates that the ± 2 percent uncertainty of the descent engine will require a bias time of approximately 65 seconds and will invoke a bias delta V penalty on the order of 45 ft/sec. In effect, the 45 ft/sec. of fuel is the penalty paid for reducing the landing dispersions from that associated with the range-free type of guidance, to that in which a desired position at hi-gate is reached. The magnitude of additional variation in the landing point that would be associated with range-free type of guidance is essentially the percentage uncertainty thrust-to-weight value times the total range travel. For the case of ± 2 percent average thrust uncertainty and a nominal range of 250 n. m., this results in approximately ± 5 n. m. of range uncertainty which can be eliminated at the cost of 45 ft/sec. of fuel penalty.

Landing Radar Updating - The effect of landing radar (LR) updating on the guidance commands is important from the standpoint of eliminating altitude uncertainties, and the resulting changes in attitude and throttle required by the change in solution of the guidance equations. The effect of landing radar update is a continuing effect throughout the trajectory once the initial update altitude is reached; and, therefore, some aspects of the following discussions will touch on the final approach phase as well as the braking phase.

The altitude update is initiated at 25,000 ft., as determined by the primary guidance system, and is continued at each two-second interval for the remainder of the approach. Velocity updates are initiated at about 15,000 ft., when the velocity is reduced to about 1550 ft/sec. The velocity is updated a a single component at a time, in two-second intervals (6 seconds for a complete update). The altitude updating is continued along with the velocity components. After each complete (3components) velocity updating, an altitude update only is performed, then the velocity updating is continued. The weighting factors for LR altitude and velocity updates are illustrated in figure 22 as linear functions of the parametcr being updated. These are linear approximations to optimum weighting based upon least-squares estimation.

The guidance commands for an ideal descent (no initial condition errors, no IMU errors, no LR errors, no terrain variations, no DPS uncertainties) are shown in figure 23. The trajectory presented in the figure is not the nominal design trajectory, but is adequate to illustrate the effects of landing radar update. This particular trajectory has a hi-gate altitude of 6100 ft. and a throttle period of 80 sec. prior to hi-gate. The pitch profile exhibits a slope discontinuity at the fictitious target point (TF) for throttling the engine, as shown in part (b) of the figure.

At the hi-gate target point (HG), the pitch angle undergoes the rapid pitchup to the constant attitude desired for final near constant (about 35° of the vertical). At the low-gate target (TLG, about 500 ft. altitude), the attitude begins to change (nearly linear) to satisfy the near vertical attitude desired just prior to the vertical descent target (TVD, about 100 ft. altitude), 10° off the vertical. The profile is terminated at this point.

The same trajectory has been analyzed for cases with initial condition errors, descent engine thrust uncertainties, IMU errors, landing radar errors and a typical terrain profile approaching the landing site. The terrain profile used is shown in figure 24 and is applicable for an approach to a site at 0°20'N latitude and 12°30'E longitude. Both a properly scaled profile and an expanded altitude scale profile are shown.

An example effect of the terrain, initial condition and system errors is shown in figure 25. In addition to the effect of the terrain the other initial predominent error included was an altitude uncertainty of about minus 1600 feet. This case is considered somewhat extreme in that the altitude uncertainty of -1600 feet is about a 3σ magnitude if CSM landmark type sightings have been made on the landing site and in a directive such the terrain effects are additive with the inertial system altitude uncertainty tending to accentuate the pitch angle and thrust variations from the ideal case. The time histories of pitch angle and thrust magnitude are presented in figure 25 and include the ideal case to provide a basis for comparison.

The pitch angle varies by slightly more than 10 degrees at a maximum prior to hi-gate and is about equivalent after hi-gate. The thrust level shows generally the same level of command. The pitch angle deviations are of concern because of possible effect upon landing radar operation and because of increased expenditure of descent engine propellants.

In the event that no landmark sightings near the landing site are performed in lunar orbit, large uncertainties (up to 10,000 ft. on 3 σ basis) in the braking altitude can exist. Investigations of the ability of the LR to update these large altitude uncertainties have indicated that 100 fps of additional delta V is required. Furthermore, throttle commands above 60 percent and large attitude deviations (up to 70°) occur in some instances in the throttle down region prior to hi-gate. Further investigation of this problem is proceeding.

Delta V Budget - The nominal fuel expenditure during the braking phase is 5206 ft/sec. To this an additional 15 ft/sec. is added to account for possible mission changes that would raise the CSM altitude 10 n.m. For the random thrust uncertainties of the descent engine a 3σ random fuel expenditure of ± 20 ft/sec. is budgeted. In addition, analysis has shown that navigation uncertainties in altitude, although eventually eliminated by the landing radar, will change fuel consumption by about 60 ft/sec. for a 3000 ft. uncertainty. To account for this, a 3σ random fuel expenditure of ± 60 ft/sec. has been allotted on the fuel budget.

Descent Guidance Monitoring - An important function of the crew during the braking phase is to monitor the performance of the guidance system onboard. This is done by checking the solution of the primary guidance system with the solution of position and velocity obtained from the abort guidance system. As indicated in figure 26, this is accomplished by periodic differencing of the primary and abort guidance solutions of altitude, altitude rate, and lateral velocities. The altitude rate parameter is perhaps the most significant parameter to monitor because this is the one that cal lead to a trajectory that violates the flight safety considerations. Analysis has shown, however, that it will take greater than the extremes of 3σ performance of the abort and primary guidance solutions to lead to an unsafe trajectory prior to the hi-gate position. Because the Manned Space Flight Network will be very effective in measuring the altitude rate of the spacecraft, it also will be very effective in providing an independent vote in the event that onboard differencing indicates the possibility of a guidance failure. The total procedures for this guidance monitoring are still in the formative stages and are currently being investigated in simulations conducted by the Manned Spacecraft Center.

Summary of Braking Phase - The braking phase, lasting about 450 seconds, covers some 243 nautical miles during which the velocity is reduced from 5500 ft/sec. to approximately 600 ft/sec., and the altitude from 50,000 feet to about 9,000 feet. The attitude during the phase is normally such that the thrust vector is close to being aligned with the flight path angle. In this attitude, the pilot is not able to look in the direction of the intended landing area. In the first portion of this phase, the LM could assume any desired roll attitude about the X or thrust axis. Mission planning will determine if the initial attitude will allow the crew to look down on the lunar surface to check the progress over the terrain. As the LM approaches the position at which landing radar will begin operating, the roll attitude will be such that the windows will be oriented away from the surface in order to provide a more favorable attitude for the landing radar operation and to prepare for the pitch-up maneuver at the hi-gate position that will allow a view forward to the landing area.

Final Approach Phase

Objectives and Constraints - The final approach phase is perhaps the most important phase, from the standpoint of the strategy. It is primarily in this phase that the trajectory is shaped at a cost of fuel, in order to provide the crew with visibility of the landing area. In this phase, the crew begins to be confronted with some of the possible unknowns of the lunar environment, such as the possibility of reduced visibility. The objectives of the final approach phase are enumerated in figure 27. The first objective is to provide the crew with out-the-window visibility, and to provide adequate time to assess the landing area. The second is to provide the crew with an opportunity to assess the flight safety of the trajectory before committing the continuation of the landing. And thirdly, to provide a relatively stable viewing platform in order to best accomplish the first and second objectives. In other words, maneuvering should be kept to a minimum. The primary constraints on the strategy in this phase are again the desire to keep the fuel expenditure to a minimum and the limitation of the LM window. In the event that the ascent engine must be used for abort during this approach to the surface, the difference in thrust-to-weight between the descent and ascent engines must also be considered as a constraint. The ascent engines thrust-to-weight initially is only about one-half of that of the descent engine in this phase. The altitude loss during vertical velocity nulling as a function of nominal trajectory altitude and velocity must be included in the consideration for a safe staged abort. The other constraints that must be considered are the problems of the lighting of the lunar terrain, and its inherent contrast properties which may make it difficult for the pilot to see and assess the terrain features. The primary variables that may be traded-off during this approach phase include the pitch attitude, the altitude at which hi-gate or the transition altitude is chosen, the flight path angle of the trajectory, and the variation of look angle to the landing area (referenced to the spacecraft thrust axis). This again considers the limitation of the LM window.

Determination of HI-gate - Perhaps the first factor that must be chosen, in order to design the final approach phase, is the hi-gate altitude. Figure 28 lists the factors affecting the choice of the hi-gate altitude. The first factor is the range from which the landing area can be assessed adequately. If this were the only factor to be considered, it would of course be unwise to waste fuel to provide this ability, if the viewing range to the target landing area was so great that the detail of the area could not be observed. The second factor is the time that the crew will require to adequately assess the landing area. A third consideration is that of flight safety requirements with regard to the undertainties of the terrain altitude considering the operating reliability of the landing radar and its ability to update the guidance system (the inertial system), and also considering the abort boundaries associated with the ascent engine (see figure 29). Preliminary estimates were made of all these factors and considering a desire to be able to get to hi-gate, even if the landing radar is not updating the guidance system, the third requirement predominates, and flight safety dictates the choice of hi-gate altitude. If further analysis of the landing radar operations indicates a high system reliability, then the flight safety requirements will be satisfied and the hi-gate altitude would be selected on the basis of the first two considerations.

The flight safety of the final approach trajectory will be largely governed by the magnitude of the uncertainties in altitude above the terrain. Figure 30 lists the present expected uncertainties. These uncertainties include that of the guidance and navigation system which considering that onboard lunar orbit navigation is accomplished, there will be an approximate 1500 ft of altitude uncertainty on a one sigma basis. If lunar navigation is conducted by the Manned Space Flight Network, the uncertainty will be approximately 500 ft less. At the present time, and largely as a result of some of the data from the Ranger spacecraft missions, there is a large undertainty in the lunar radius magnitude, both the bias and the random uncertainties.

Both of these quantities are established as one kilometer or approximately 3200 ft, 1σ basis at this time. Lunar Surface Technology personnel have indicated that their present capability in determining the slopes in the areas of the maria is limited to an uncertainty of approximately $\pm 3^{\circ}$ on a 3σ basis so this is equivalent to a 700 ft, 1σ uncertainty, considering the ranges of uncertainty of the landing position. In addition, our present mission planning allows for a terrain profile along the approach path limited to a general slope of $\pm 2^{\circ}$ with local variations not to exceed ± 5 percent of the nominal LM trajectory altitude. This results in altitude biases of 700 to 800 fr (3σ) over the ranges of uncertainty of the landing position.

The minimum hi-gate altitude can be determined by combining the altitude 3σ uncertainties and biases previously discussed. The manner in which these factors are combined, however, depends upon the navigational updating in orbit (with CSM optics or MSFN) and during the powered descent (with LR). Results for the various combinations are given in figure 31. The first case is based upon MSFN orbit navigation and no LR updating and represents the largest hi-gate altitude, 32,600 ft. This extreme and impractical hi-gate results from the fact that no terrain updating occurs anytime during the mission; and therefore all of the uncertainties and biases are maximum.

The second case differs from the first only in that two sighting from orbit on a landmark, in the proximity of the landing site, are provided in order to update the position (radius) of the landing site. This case assumes that orbit navigation of the CSM state is accomplished by MSFN and LR updating during the powered descent is not available. The minimum hi-gate for this case is 6700 ft, a substantial reduction over case 1. This is because the landing site update eliminates the lunar radius bias and reduces the random uncertainties in radius significantly.

The third case shows a moderate increase in hi-gate altitude over case 2 due to the moderate increase in PGNCS uncertainties from onboard navigation (which includes the landing site update) as opposed to MSFN navigation. The minimum hi-gate for this case is 7500 ft.

The preceding analysis has assumed that the crew would immediately assess a collision situation and take the appropriate action. Allowing a finite time, on the order of 10 seconds, for assessing the situation, an operational hi-gate altitude satisfying crew safety without LR is approximately 9000 ft.

Parameter Trade-Offs - Considering that the hi-gate altitude requirement has been set at approximately 9000 ft, the major trade-offs that are still needed to be established include the flight path angle, the acceptable look angle to the landing area, and the time required to assess the landing area. Each trade-off may affect the state vector that is specified at hi-gate, and this change must be taken into account in the total landing descent profile planning. Figure 32 shows the penalty of fuel as a function of hi-gate altitude. The selection of about 9000 ft as the hi-gate altitude costs about 250 ft/sec of delta V. Because the LM pilot can only see down 65° from his straight ahead viewing position, it is desirable for the look angle to be greater than 25° above the thrust axis. Considering the variations in attitude that may come about through the guidance system caused by flying over variable terrain, a desired look angle of 35° has been chosen providing a margin of 10° over the lower limit of the window. The flight path angle is also important. The angle must not be too shallow in order to get the proper perspective of the landing area as it is approached, and, on the contrary, it must not be too steep, purely from the standpoint of the pilot being better able to judge the safety of the approach path. In figure 33, the delta V penalty for variations in flight path angle for various look angles is illustrated. As can be seen from the figure, the major delta V penalty is incurred for increasing the look angle. Little penalty is paid for varying the flight path angle from 10° up to 20° for a given look angle. The sum total of the trade-off is that the hi-gate altitude will be approximately 9000 ft, the look angle to the target approximately 10° above the lower limit of the window, and the flight path angle will be in the order of 13° to 15° throughout the major portion of the final approach phase.

The shaping accomplished in the final approach phase costs approximately 270 ft/sec of equivalent fuel. In order to see what this has provided, figure 34 shows a comparison of the selected trajectory with that of the fuel optimum showing the variations of horizontal and vertical velocity as a function of time to go. Figure 33 shows that the time to go from 9000 ft altitude down to the lo-gate position has been increased by approximately 45 seconds. In addition, the vertical velocity has been cut by approximately a third for equivalent altitudes; however, the primary difference shows up in the comparison of horizontal velocity at equivalent altitudes, noting that at 5000 ft the fuel optimum trajectory has a velocity of about 1000 ft/sec, whereas the selected trajectory has a horizontal velocity of about 450 ft/sec.

Redesignation Footprint - Even though an adequate perspective of the landing area and adequate viewing time are provided by the selection of the flight path angle, the line-of-sight angle, and the hi-gate altitude, it is still pertinent to determine how much of the area the pilot needs to survey. This, in turn, is a function of how much fuel the pilot will have in order to change his landing site if he decides that the point to which the guidance system is taking him is unacceptable. Assuming that it will take the pilot a few seconds to get oriented to the view in front, it appears that the maximum altitude from which he could consider a redesignation would probably be less than 8000 feet. Figure 35 shows the available footprint as a function of fuel required for this purpose. The perspective of the figure is that of looking directly from overhead the spacecraft perpendicular to the surface where the spacecraft position is at the apex of the lines. The nominal landing point, or that point to which the spacecraft is being guided by the automatic system, is the zero-zero range position. The solid contour lines are the ranges that could be reached provided that the indicated amount of fuel could be expended. For a delta V expenditure of approximately 100 ft/sec, an additional 8000 ft downrange could be obtained, and approximately 10,000 ft in either direction crossrange. The horizontal line at the bottom of the figure indicates the lower window limit, and the second line indicates the position 5° above the lower window limit. The other lines indicate the side window view limitations experienced by the pilot or command pilot, on the left. The copilot would have a similar limitation of side vision toward the direction of the pilot, therefore, only the region bounded by the inboard side window limits would be common to the field of view of both crew members.

The variation of footprint capability as the altitude is decreased during the descent is indicated in figure 36. Contours of footprint capability are shown for an expenditure of 100 ft/sec of fuel at altitudes of 8000 ft, 5000 ft, and 3000 ft. The footprint capability naturally shrinks the closer the approach is made to the landing area. However, a given budgeted amount of fuel provides an area that subtends very closely to the same angular view from the pilot's viewing position. The present strategy is based upon having a high probability that the intended landing area will be generally suitable, and, for this reason, there will be a low probability of requiring large redesignations of the landing position.

It has been assumed that a maximum capability of designating 3000 ft downrange will be required and this provision of fuel is allotted for redesignation at 5000 ft of altitude. Approximately 45 ft/sec of fuel is required for this redesignation capability. Figure 37 shows the footprint available for this fuel allotment.

The LM pilot does not have the opportunity to see the footprint as viewed here, but instead from the perspective provided by the approach flight path angle. The pilot view from the hi-gate altitude is indicated in figure 38. During this phase, the spacecraft is pitched back approximately 40°, thus, the horizon is very near the - 40° elevation depression angle. The landing site is at approximately 55° depression or about 10° above the lower limit of the window. For reference purposes a 3000 ft circle has been drawn about the landing position and the landing footprint associated with a delta V of 100 ft/sec is shown.

Landing Point Designator - The pilot will know where to look to find the intended landing area, or the area which the guidance system is taking him, by information coming from the guidance system display and keyboard (DSKY). This information will be in the form of a digital readout that allows him to locate the correct grid number on the window, commonly called the landing point designator (LPD). After proper alinement of the grid, the pilot merely has to look beyond the number corresponding to the DSKY readout to find where on the lunar surface the automatic system is guiding the spacecraft. The proposed grid configuration for the landing point designator is shown in figure 39.

The process of landing point designation and redesignation is illustrated in figure 40. The guidance system always believes that it is following the correct path to the landing site. It has the capability at any time to determine the proper look angle or line-of-sight to the intended landing site. Because of orbital navigation errors and also drifts of the inertial system during the powered descent, the actual position of the spacecraft will not be the correct position. Thus, if the pilot looks along the calculated line-of-sight he would see an area different from that of the desired landing area. Should the desired landing area appear in another portion of the window, then the pilot, by taking a measurement of the angle formed by the line-of-sight readout from the guidance system and the new line-of-sight (to the desired point), can input the change in line-of-sight into the guidance computer.

will be a cooperative task between the pilot and the copilot where the copilot will read the DSKY and call out to the pilot the numbers corresponding to the landing point designator. The pilot will then orient his line-of-sight so that he can look beyond the proper number on the landing point designator and see where the guidance system is taking him. If he is not satisfied with this position, then he can instruct changes in the guidance system by incrementing his attitude hand controller. During this portion of the approach, the guidance system is flying the spacecraft automatically so that the pilot's attitude hand controller is not effective in making attitude changes. With each increment that the pilot makes in moving the hand controller in a pitching motion, there is an instruction to the guidance system to change the landing point by the equivalent of a half-degree of elevation viewing angle. Lateral changes in the landing position would be made by incrementing the hand controller to the side in a motion that would normally create rolling motion of the spacecraft. Each increment of a hand controller in this direction causes a 2° line-of-sight change laterally to the landing area. When the guidance system receives these discrete instructions it recalculates the position of the desired landing area and commands the pitch or roll attitude in combination with a throttle command required to reach the desired position. This results in a transient response from the spacecraft until the new attitude and throttle setting commands are responded to. After the transient has settled out, the copilot would normally read the DSKY again and inform the pilot what new number to look for to find the desired landing area. The pilot would then orient himself to look at this number and check to see if his instructions to the guidance system had been fully correct. If not, some refinement in landing site selection would then be made.

The response of the spacecraft to redesignations of landing position is important. For example, if the new site selected is further downrange, the spacecraft will pitch closer to the vertical and reduction in throttle will be made so that the new position will be more closely centered in the pilot's window. If, however, the site chosen is short of the original landing site, then the spacecraft would have to pitch back and increase throttle in order to slow down and obtain the new desired position. These attitude motions affect the line-of-sight and become important because of the danger of losing sight of the target. Some typical responses to changes in the landing point are shown in figure 43. The variation of the line-of-sight to the landing site (looking angle) with time from hi-gate is shown for the nominal case, a redesignation

The guidance system will then recompute the location of the desired landing area. When this occurs the guidance system, in effect, begins a period of relative navigation where the new landing point is calculated in the present reference frame and is not significantly affected by whatever inertial system or other navigational errors that may have occurred.

The accuracy with which the landing point designation or the redesignation process can be made is a function of how accurately the line-of-sight can be interpreted, or correctly displayed to the pilot.

There are several sources of redesignation errors, as indicated in figure 41. These include the variations in terrain along the approach to the landing site, the guidance dispersion effect upon altitude (provided the landing radar updating is not complete), boresight installation, the inertial measuring unit reference misalinement, and the errors of application by the spacecraft crew. The effect of the altitude errors whether from the terrain, or from the guidance system altitude uncertainties, are shown graphically in figure 42. In this case, the guidance system assumes the landing site is at the same elevation as the terrain over which the spacecraft is flying; and, therefore, determines the line-of-sight through that point. However, when the crew views this line-of-sight the intercept point with the lunar surface is at an entirely different point than the intended landing position. For flight path angles of about 14°, this ratio of downrange error to altitude error is approximately 4 to 1. Altitude errors do not affect the lateral dispersions. It is obvious that although the landing radar performs a very vital function in reducing the altitude dispersions of the guidance system, there is probability that the same landing radar function will update the inertial system with a false indication of the landing position altitude.

The errors other than the altitude type errors (the installation IMU and the pilot application errors) all tend to be biases. Preliminary testing indicates that these errors could be of the order of one-half degree. Again for typical flight path angles of about 14° this half degree of application boresight error will lead to redesignation errors downrange on the order of 800 ft for redesignations occurring in the altitude range of 5000 to 8000 ft. These downrange errors will reduce to the order of 100 ft when the redesignations are made at altitudes of 1000 ft or less. Thus, there is a trade-off with regard to the probable magnitude of the errors that vary with altitude, especially if the approach terrain is likely to have large variations of altitudes. The process of redesignation downrange and redesignation uprange. The redesignations occur at an altitude of 5000 ft. For the nominal landing site, the line-of-sight look angle is maintained between 35° and 30° throughout the phase. For the 3000 ft long redesignation the look angle is increased over the nominal case varying between 45° and 35° (after the resulting transient response is completed). For the 3000 ft short redesignation the pitchback motion of the spacecraft causes the line-of-sight angle to the very target area to be decreased to approximately 20° initially, increasing to about 28° for a short-time interval. Thus, for this case, visibility of the landing area would be lost for a portion of time since the lower window limit is 25°. For this reason, it would be the normal procedure not to redesignate short by more than the equivalent of about 2000 ft at this altitude. At lower altitudes, shorter range redesignations should be limited to proportionally less magnitude. For crossrange redesignations, the effect on the look angle is slight for redesignations up to 3000 ft; however, the spacecraft will require a new bank attitude (which is nominally zero for in-plane redesignations). Thus, this figure does not present the total attitude response transients for the effect of site redesignations.

An important aspect of the redesignation process is the problem of how to account for the propellant fuel expenditure. There is no accurate procedure to account for this fuel other than to interrogate the guidance system for the amount of fuel remaining.

The guidance computer load is quite heavy at this time, therefore, it is probable that a rule of thumb approach may be utilized, which, in effect, informs the pilot that so many units of elevation and azimuth redesignation capability can be utilized. Sufficient conservatism can be placed on this number to insure that the pilot does not waste fuel to the extent that the landing could not be completed. At the same time, this would allow the pilot a rough assessment of whether or not the new landing area would be within the fuel budget.

Delta V Budget - The fuel expenditure during the nominal final approach phase will be an equivalent to 889 ft/sec characteristic velocity. To this number is added, for budget purposes, a bias allowance of 45 ft/sec for the landing point redesignation capability, and a 3σ random allowance of 15 ft/sec for refinements in the landing site designation.

Summary of Final Approach Phase - The final approach phase covers about 5 1/2 nautical miles during which the altitude is decreased from 9000 ft to 500 ft, and the velocity from 600 ft/sec to 50 ft/sec. The time required normally will be about 105 seconds during which time the pilot will have a continuous view of the landing area. It is during this time that assessments of the landing area will be made, and required redesignations of the landing position to more favorable landing terrain will be accomplished.

The Landing Phase

Objectives and Constraints - The basic purpose of the landing phase is to provide a portion of flight at low velocities and at pitch attitudes close to the vertical so that the pilot can provide vernier control of the touchdown maneuver, and also to have the opportunity for detailed assessment of the area prior to the touchdown. In order to accomplish this, the trajectory is further shaped after the final approach phase. The guidance system is targeted so that the design constraints of the lo-gate position are met, but the actual target point will be at or near the position where the vertical descent begins. The final approach phase and the landing phase are then combined with regard to the manner in which the guidance system is targeted. The targeting design would satisfy the constraints of both the terminal portion of the final approach phase and the landing phase by proper selection of the targeting parameters. There will be a smooth transition from the extreme pitch-back attitude with associated with the final approach phase and the near vertical attitude of the landing phase.

In the final approach phase, the trajectory was shaped in order to pitch the attitude more toward the vertical so that the approach conditions would allow the pilot to view the landing site. The resulting pitch attitude, approximately 40° back from the vertical is, however, still quite extreme for approaching the lunar surface at low altitudes, hence, it is necessary to provide additional shaping in order to effect a more nearly vertical attitude at the termination of the total descent. Figure 44 shows a comparison of the nominal attitudes for those two phases. The objectives and constraints of the landing phase design are presented in figure 45. The first objective is to allow the crew to make the detailed assessment, and a final selection, of the exact landing point. In order to accomplish this, there will be some flexibility in the propellant budget to allow other than a rigid following of the design trajectory. This leads to objective number two, in which it is desired to allow some maneuvering capability and adjustment of the landing point. The constraints are familiar ones including the fuel utilization, the physical limitations of the window, and in turn, the lighting and associated visibility of the surface, the visibility associated with the lighting, the actual terrain, and the possibility of blowing dust maneuvering within the desired attitude limits in order to retain the advantages of a fairly stable platform, and last, what is termed the staged abort limiting boundary. This boundary defines the circustances under which an abort maneuver cannot be performed without the ascent stage hitting the surface. This curve is based upon a combination of vertical velocities, altitudes, and the pilot-abort-staging system reaction time.

Nominal Trajectory - The variables that are available to try to satisfy all of these constraints and objectives include variations in the approach flight path and the velocities involved, the attitude of the spacecraft, and the actual touchdown control procedures. The landing phase profile which has resulted from almost $2\frac{1}{2}$ years of simulating the maneuver is illustrated in figure 46. The lo-gate point is at an altitude of approximately 500 ft., at a position about 1200 ft back from the intended landing spot. The landing phase flight path is a continuation of the final approach phase flight path so that there is no discontinuity at the lo-gate position. At the start of this phase, the horizontal velocity is approximately 50 ft/sec and the vertical velocity is 15 ft/sec. The pitch attitude is nominally 10 to 11° throughout this phase, but rigid adherence to this pitch attitude is not a requirement. The effect of the pitch attitude is to gradually reduce the velocities as the flight path is followed in order to reach the desired position at an altitude of 100 ft from which a vertical descent can be made. Modification of this trajectory can be accomplished simply by modifying the profile of pitch attitude in order to effect a landing at slightly different points than that associated with the nominal descent path. No actual hover position is shown in the approach porfile because the vertical velocity or descent rate nominally does not come to zero. The approach is a continuous maneuver in which forward and lateral velocities would be zeroed at approximately the 100 ft altitude position and the descent velocity allowed to continue at approximately 5 ft/sec. This allows a very expeditious type of landing, however, if a hover condition is desired near the 100 ft altitude mark. It is a very simple matter for the pilot to effect such a hover maneuver. The only disadvantage of the hover maneuver is the expenditure of fuel. The total maneuver from the lo-gate position will normally take approximately 80 seconds. If flown according to the profile, the descent propellant utilized will be equivalent to about 390 ft/sec of characteristic velocity. During the landing approach, the pilot has good visibility of the landing position until just before the final vertical descent phase. Figure 46 also shows a nominal sequence of pilot views of a 100 ft radius circular area around the landing point. However, even during the vertical descent, the area immediately in front of and to the side of the exact landing position will be visible. The LM front landing pad is visible to the pilot. In addition to being able to observe the intended landing site, the pilot has ample view of much of the lunar surface around him so that if the original site is not suitable he can deviate to the other landing position, provided that the new landing position is obtainable with the fuel available. The basic system design will allow the entire maneuver to be conducted automatically. However, the LM handling qualities make it a satisfactory vehicle for the pilot to control manually. The satisfactory nature of the LM manual control handling qualities has been demonstrated by fixed base simulation and by flight simulation at the Flight Research Center using the Lunar Landing Research Vehicle and the Langley Research Center using the Lunar Landing Research Facility. Simulations have shown that here should be no problems involved if the pilot decides to take over from manual control at any time during the terminal portion of the final approach phase or the landing phase.

Much concern has been generated with regard to the problem of visibility during the landing approach. This factor has led to a constraint upon the sun angle at the landing site, as will be discussed by the paper on Site Selection. In the event that the pilot has some misgivings about the area on which he desires to land, the landing phase can be flexible enough to accommodate a dog-leg type maneuver that will give the pilot improved viewing perspective of the intended landing position. Manual control of this maneuver should present no problem and could be executed at the option of the pilot. At the present time, trajectory is not planned for an approach in order to maintain simplicity of trajectory design, because of the expected ease in which the maneuver could be accomplished manually should the need be present. Should, however, the dog-leg be identified as a requirement for an automatic approach, it will be incorporated.

A profile of the altitude and altitude rate of the landing phase is shown in figure 47. The altitude rate is gradually decreased to a value of about 5 ft/sec at the 100 ft altitude position for vertical descent. The descent rate of 5 ft/sec is maintained at this point in order to expedite the landing. At approximately 50 ft of altitude (± 10 ft), the descent rate would be decreased to the design touchdown velocity of $3\frac{1}{2}$ ft/sec. It is not necessary for this to be done at exactly 50 ft so that uncertainties in the altitude of the order of 5 to 10 ft would not significantly affect the approach design. The value of $3\frac{1}{2}$ ft/sec descent rate is then maintained all the way until contact with the surface is effected and procedures initiated for cutoff of the descent engine. The curve labeled staged-abort boundary shown in figure 47 is applicable to the situation in which the descent engine has to be cut off and the vehicle staged to abort on the ascent engine. It is obvious that this boundary must be violated prior to effecting a normal landing on the surface. However, with the current design, this boundary is avoided until the pilot is ready to commit himself to a landing so that it is only in the region of below 100 ft that he is in violation of the boundary.

Delta V Budget - A summary of the landing phase fuel budget is given in figure 48. The budget reflects allowances for several possible contingencies. For example, the pilot may wish to proceed to the landing site and spend some time inspecting it before he finally descends to the surface. This would require that the spacecraft hesitate during the approach, and the penalty involved is the amount of fuel expended. A period of 15 seconds of hover time will cost about 80 ft/sec of fuel equivalent. There is also the possibility that the performance of the landing radar may be doubtful, in which case the spacecraft crew might want to hover in order to visually observe and null out the velocities. It has been found by means of flight tests in a helicopter, that velocities can be nulled in this manner within 1 ft/sec after less than 15 seconds of hover time (another 80 ft/sec of fuel expenditure). It would be possible to update the inertial system in this manner and allow the spacecraft to proceed and land on the surface with degraded landing radar performance during the final portion of the descent. If there are errors in the radar vertical velocity, there will be a direct effect upon the time required to complete descent and a random ± 65 ft/sec of equivalent fuel has been allotted in the fuel budget. Another descent engine fuel contingency that must be accounted for is the possible variations in the pilot control technique including the deviations from the planned flight profile the pilot might employ. Simulation experience has indicated a need for an average addition of 80 ft/sec of fuel and a random ± 100 ft/sec. It is noteworthy that only 30 seconds of hover time has been budgeted and that for specifically designated purposes.

Fuel Budget Summary

A summary of the total LM descent fuel budget is given in figure 49. The budget is divided into that required by the baseline trajectory requirement totaling 6582 ft/sec, and items, described as contingencies, totaling 353 ft/sec mean requirement with an additional ± 143 ft/sec random requirement. This leads to a total 7050. The inclusion of the RSS random contingencies as a fuel requirement is considered a conservative approach in that each of the random congingencies could lead to a fuel savings as well as a feul expenditure. The present tankage would provide up to 7332 ft/sec of fuel or about 282 ft/sec more than the budget. Thus, the possibility of additional landing flexibility can be provided by fuel tanks, or in the interest of weight savings, some off-loading of fuel can be considered. The addition flexibility is equivalent to a hover time of about one minute or to additional downrange landing redesignation capability of almost 20,000 feet for a redesignation at 8,000 ft altitude.

The fuel budget summary is presented in figure 50b as a How-Goes-It plot of the expenditure of fuel both in equivalent characteristic velocity and pounds as a function of time and events during the descent. The solid line give the baseline trajectory and results in a fuel remaining of 778 ft/sec at touchdown. Adding the utilization of all of the budgeted contingency mean values of fuel is represented by the dashed line. When these contingencies are utilized the time basis of the plot will be incorrect, particularly for the time between Lo-Gate and Touchdown. The total time could extend to as much as 12½ minutes (735 seconds) in the event that all of the contingency fuel were utilized for hovering over the landing site.

3.0 LUNAR LANDING TOUCHDOWN CONTROL, AUTOMATIC AND MANUAL

Perhaps the most important single operation in the lunar landing mission is the actual touchdown maneuver. It is during this maneuver that the uncertanities of the lunar surface become a real problem. A recommended procedure for controlling the approach has been developed. This procedure, developed partly through simulation, involves reaching a position at about 100 ft above the landing site and descending vertically to the lunar surface, as previously described. During the vertical descent, the lateral velocities are nulled and the vertical velocity controlled to a prescribed value until the descent engine is cut off just prior to touchdown. The procedures for effecting descent engine shutdown will be discussed in detail.

There are two control modes by which the landing operation can be performed, as indicated in figure 51. The first is completely automatic. In this mode, while the pilot may have used the landing point designator to select the touchdown point, he is not active in the actual control loop. The second mode is manual, but is aided by automatic control loops, that is, the pilot has taken over direct control but he has stabilization loops to provide favorable control response. In addition, the manual mode normally will be used in conjunction with a rate-of-descent, command mode to further aid the pilot in control of the touchdown velocities. Within the manual landing mode, the pilot has two options; (1) land visually, which would require that there be no visual obscuration as might come from dust or lunar lighting constraints, or (2) because of such obscurations he would control the landing through reference to flight instruments. Because of the expected good handling qualities of the LM, the manual visual mode should be very similar to flight of a VTOL aircraft here on earth. No landing attitude or velocity control problem is anticipated and the control should be within one foot per second lateral velocities. Manual-instrument mode of control does have sources of error, however, that may degrade control and those that have been considered include the following: control system response, landing radar velocity measurement, landing radar altitude measurement, IMU accelerometer bias, IMU misalignment, display system for manual only, the pilot, for manual only, and the center-of-gravity (c.g.) position. Several of these parameters are listed in figure 51 as being of prime importance.

In considering the control of the landing, emphasis has been placed on the method of timing of shutting off the descent engine. Because of possible unsymmetrical nozzle failure due to shock ingestion and a desire to limit erosion of the landing surface, an operating constraint of having the descent engine off at touchdown has been accepted. Probable errors in altitude information from either the inertial system or from the landing radar preclude the use of this information for the engine cutoff function, even though the accuracy may be of the order of five feet, because of the deleterious effect on touchdown vertical velocities. The need for an accurate, discrete indication of the proper altitude to cut the descent engine off led to the adoption of probes extending beneath the landing pads rigged to cause a light in the cockpit to turn on upon probe contact with the lunar surface. The light-on signal informs the pilot that the proper altitude has been reached for engine cutoff. The probe length must be determined from a consideration of delay times in pilot response, descent engine shutoff valve closures, and tail-off and the nominal descent velocities. The sequence of events is shown in figure 52.

The variation of descent rate at touchdown as a function of descent rate at probe contact is shown in figure 53, and includes the effect of pilot reaction time. The curves are representative of a 53-inch foot probe being used, coupled with a 0.25 second total engine shutoff delay time. This engine delay time includes that time required for the electronic signal to be generated, the shutoff valves to close, and the thrust tail-off to be essentially completed. The heavy dashed line on the chart going up on a 45 degree angle indicated a combination of descent rate at probe contact, plus system delay and pilot reaction times, that would cause the engine to still be on at touchdown. If the desired final rate of descent has been achieved, up to 1.0 second pilot delay time can be tolerated and still have the descent engine off at touchdown. As shown in figure 53, the actual touchdown velocity is just slightly more than the descent rate at probe contact, or about four feet per second. Faster reaction time would increase the final touchdown velocity, but not beyond present landing gear impact limit. If manual control allowed a slightly higher than desired final descent rate, and radar errors at the time of final update also allowed a slightly higher descent rate, these compounded increases might yield descent rates on the order of 5 to 6 ft/sec. These increased rates coupled with the 0.6 second reaction time would mean not meeting the criteria of having the descent engine off at touchdown. One solution for this situation would be to extend the probes to allow more leeway in pilot reaction time. However, the advantages of longer probes must be traded off against a probable decrease in reliability and an increased probability of touching down with greater than acceptable vertical velocities. A simulation study of this maneuver with the pilot cutoff of the descent engine showed that pilot reaction times averaged about 0.3 seconds, as shown in figure 54.

Pilot-in-the-loop and automatic control simulation studies have been conducted of the landing control maneuver. The pilot-in-the-loop studies were made using a simulated LM cockpit including all the control actuators (attitude, throttle and descent engine cutoff). The simulation included the major sources of system errors, such as platform misalignment, accelerometer bias, instrument display resolution, center-of-gravity offsets, and landing radar errors. The landing radar errors are a prime factor in the touchdown control process and the models assumed for the analysis are shown in figure 55. The specification performance of the landing radar calls for each of the three components of velocity to be measured within 1.5 ft/sec on a 3σ basis. Currect predictions are that this specification will be met in lateral and forward directions and bettered by 3/4 ft/sec vertically. For a conservative analysis, the predicted performance has been degraded by a factor or two.

The simulation results of landing velocity manual control with specification performance by the landing radar are shown in figure 56. The dashed lines indicate the present design criteria for the landing gear. The 0.9, 0.99 and 0.999 probability contours are shown and are well within the design envelope. The effect of changing the length of the landing probes is to adjust the vertical velocity bias velocity approximately 1 ft/sec per foot change in probe length.

The effect of landing radar performance upon the landing velocity envelope is shown in figure 57. The 0.99 probability contours are shown for the cases of no radar errors, specification performance, predicted performance, and degraded (predicted) performance. The resulting contours show the almost direct dependence of touchdown velocity error upon the landing radar velocity performance.

The comparative results between automatic and manual control of the landing touchdown velocities are shown in figure 58. The 0.99 contours show that automatic control results in lower touchdown velocities, but the difference is much less pronounced for the degraded radar performance as compared with the predicted radar performance. The figure does not, of course, reflect the advantage that manual control provides in closer selection of the actual touchdown position in the event that the terrain is not uniformly satisfactory.

Additional analysis of these same results for the control performance for attitude and attitude rates indicated that control within the present criteria of 6 degrees and 2 degrees per second can be expected on a 3σ probability.

4.0 ABORT AFTER TOUCHDOWN

Although analysis and simulation tests indicate a high probability that the landing touchdown maneuver will be within the landing gear design criteria, there is still an interest in the ability to abort should the landing dynamics become unstable. The ability to abort will be a function of when the need for the abort is recognized, the time required to initiate abort, the time involved in separation of the ascent stage, the thrust buildup time of the ascent stage, the attitude and the attitude rate at separation, and the control power and control rate limitations of the ascent stage.

At staging, the control power of the ascent stage is about 35 deg/sec^2 for pitch and roll attitude maneuvers. Under emergency manual control where the pilot deflects his attitude hand controller hard-over, there is no attitude rate limitation. Normal manual control commands are limited to 20°/sec and automatic control limited to 10°/sec in pitch and 5°/sec in roll. These attitude rate limitations are important from the standpoint of determining how quickly the ascent stage attitude can be returned to the vertical in the event of an impending tipover.

An analysis was made of the boundary of over-turn conditions from which a successful staged abort could be made. The results of this analysis are shown in figure 59. Two boundaries are shown; one for emergency manual attitude control which requires the pilot to put his hand controller hard over and the other for a rate limit consistent with automatic roll response (5°/sec). Both boundaries apply to the conditions under which an abort action must be recognized as being required. The boundaries allow a total of 1.4 seconds for the time required for the pilot to actuate the abort control, the staging to take place, and the ascent thrust to build up to 90 percent of rated thrust.

In addition to the boundaries, there is also a line indicating the neutral stability boundary or the sets of condition under which the spacecraft would just reach the tipover balance point of about 40 degrees. The curve labeled Landing Gear Design Envelope Maximum Enegry applies to the improbable, if not impossible, case where the landing was made at the corner of the velocity criteria envelope 7 ft/sec vertical and 4 ft/sec horizontal, and all of the energy was converted to rotational motion. It is, therefore, highly improbable that conditions will be encountered that lie to the right of this curve.

For the emergency manual control, the boundary indicated an abort can be made at an altitude of about 60 degrees if the rate is not greater than 10 deg/sec. This condition would take more than 4 seconds to develop after the initial contact with the lunar surface. For the other extreme of attitude rate limit (5°/sec) applicable only to automatic roll attitude control, the boundary is reduced about 10 degrees in attitude.

The pilot will have indication of attitude from his window view and from the attitude instrument display (FDAI). Both of these are considered adequate sources of attitude information in the event that the spacecraft passes a 40 degree deviation from the vertical and an abort becomes necessary.

Considering the improbability of landing contact that would result in an unstable post-landing attitude and the probability that even in such an event the pilot could initiate a safe abort, there does not appear to be a requirement for an automatic abort initiation.

5.0 LEM DESCENT LOGIC FLOW

The preceeding sections have described and explained the design of the LM descent strategy and the resulting trajectory design. From the pilot's standpoint there are a number of judgments and decisions that will have to be made in the period from Hi Gate to Lo Gate to touchdown. It is believed that the strategy allows a logical sequence of events and decisions and adequate time for the pilot function. This will be partly confirmed or adjustments made through extensive simulations with the LM Mission Simulators. The final confirmation will, of course, be the results of the first LM landing approach. In order to aid in the understanding of the logic and proposed sequence of decisions, a logic-flow chart has been prepared that is applicable from the Hi Gate position to landing touchdown. These charts are presented in figures 60a) and b) for the information and use of persons interested in detailed examination of the logic and in constructing the crew loading time lines. Details of these logic flow charts will not be discussed further in this paper.

6.0 SUMMARY

A LM descent strategy has been presented which is designed to take advantage of the LM system and the LM crew in order that the LM will continually be in an advantageous position to complete the lunar landing. The three phases trajectory is designed to maintain fuel expenditure efficiency, except in those regions of the trajectory where such factors as pilot assessment of the landing area require a judicious compromise of fuel efficiency.

The lunar landing strategy has considered all identified problems which might adversely affect the lunar landing and the resulting design calls for a fuel expenditure budget of 7050 ft/sec of characteristic velocity. This budget is approximately 282 ft/sec less than the current tank capacity of the LM. This margin is considered ample for dealing with presently unforeseen problems which may be identified prior to the lunar landing.

Questions and Answers

LUNAR EXCURSION MODULE DESCENT

Speaker: Donald C. Cheatham

1. Mr. Kelly - Probability plots of landing velocity show constant vertical velocity for all probabilities when horizontal velocity is zero; is this correct?

   ANSWER - Mr. Kelly and Mr. Cheatham discussed the data after the meeting and resolved their differences on the presentation form.

NASA-S-66-6025 MAY

THEORETICAL LM DESCENT

IMPULSIVE ΔV

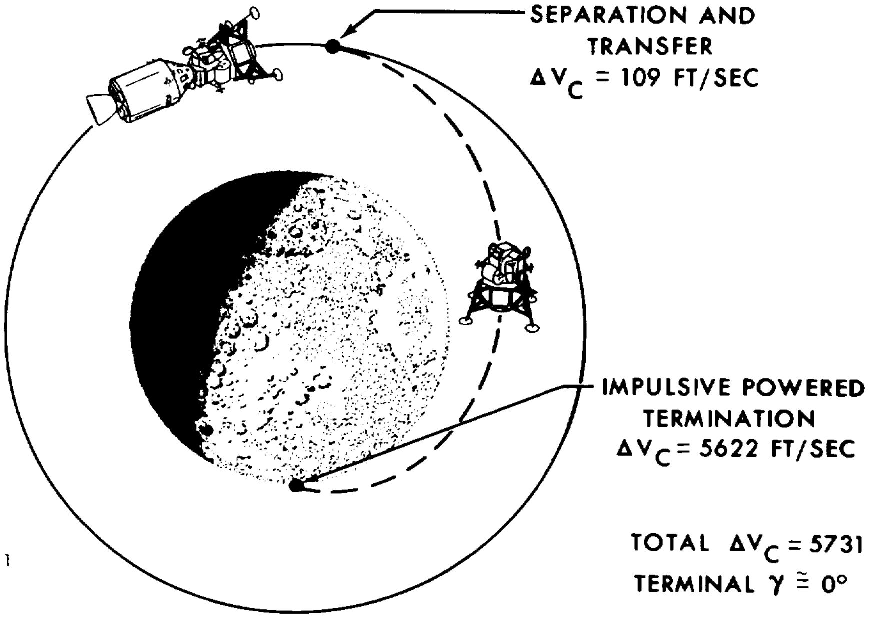

SEPARATION AND TRANSFER
ΔV_C = 109 FT/SEC
IMPULSIVE POWERED TERMINATION
ΔV_C = 5622 FT/SEC
TOTAL ΔV_C = 5731
TERMINAL $\gamma \cong 0°$

FIGURE 1

NASA-S-66-5039 JUN

VARIATION OF POWERED-DESCENT CHARACTERISTIC VELOCITY WITH THRUST-TO-WEIGHT RATIO

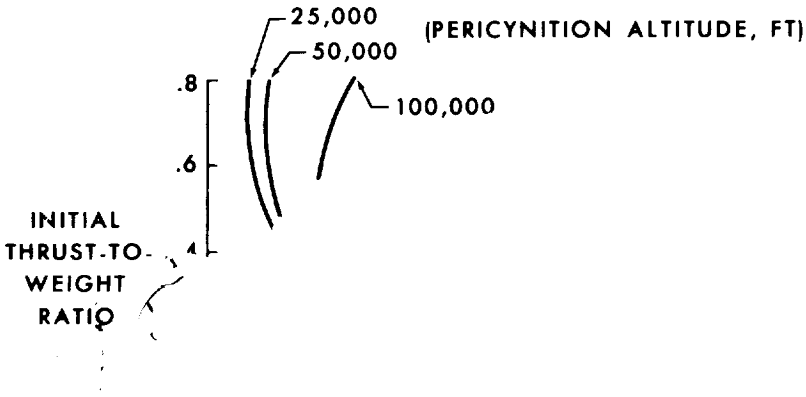
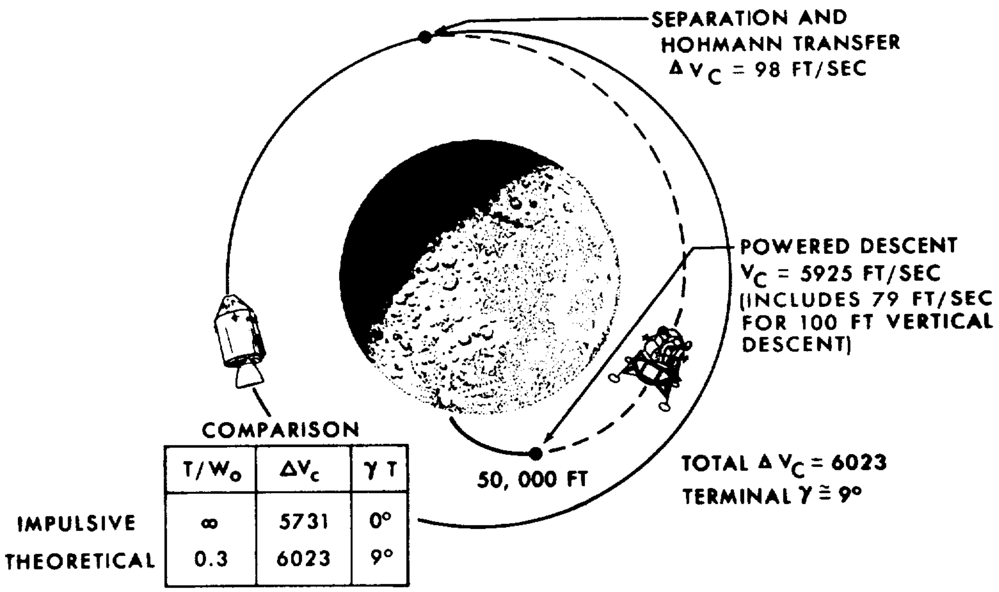


FIGURE 2

NASA-S-66-6026 MAY

THEORETICAL OPTIMUM LM DESCENT

$(T/W_O = .3,\ H_P = 50{,}000\ \text{FT})$

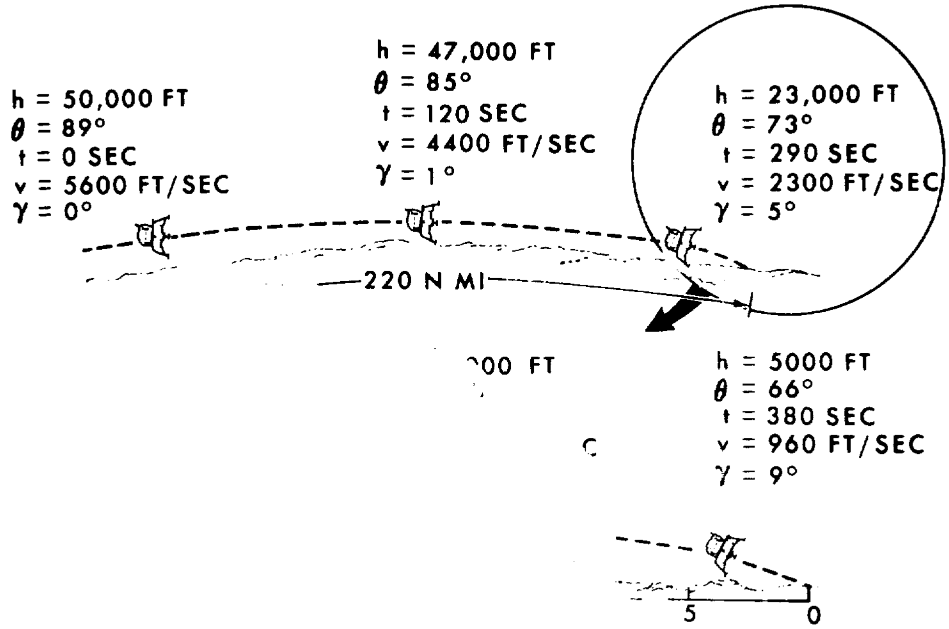

SEPARATION AND
HOHMANN TRANSFER
ΔV_C = 98 FT/SEC
POWERED DESCENT
V_C = 5925 FT/SEC
(INCLUDES 79 FT/SEC
FOR 100 FT VERTICAL
DESCENT)
50, 000 FT
TOTAL ΔV_C = 6023
TERMINAL $\gamma \cong 9°$
COMPARISON

	T/W_o	ΔV_c	γ T
IMPULSIVE	∞	5731	0°
THEORETICAL	0.3	6023	9°

FIGURE 3

NASA-S-66-5048 JUNE 1

OPTIMUM POWERED DESCENT


h = 50,000 FT
θ = 89°
t = 0 SEC
v = 5600 FT/SEC
γ = 0°
h = 47,000 FT
θ = 85°
t = 120 SEC
v = 4400 FT/SEC
γ = 1°
h = 23,000 FT
θ = 73°
t = 290 SEC
v = 2300 FT/SEC
γ = 5°
220 N MI
00 FT
h = 5000 FT
θ = 66°
t = 380 SEC
v = 960 FT/SEC
γ = 9°
5
0

NASA-S-66-6025 MAY

THEORETICAL LM DESCENT

IMPULSIVE ΔV

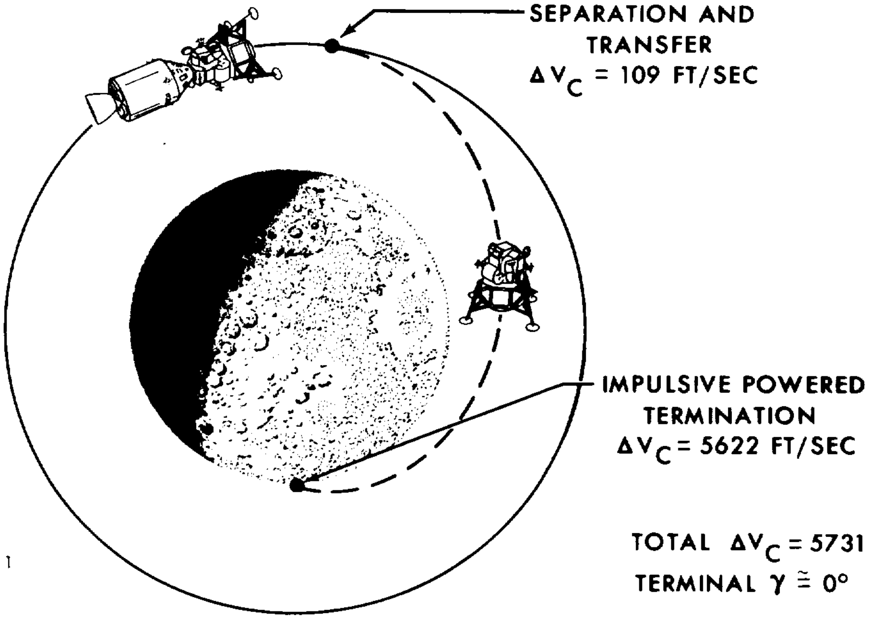

SEPARATION AND
TRANSFER
ΔV_C = 109 FT/SEC
IMPULSIVE POWERED
TERMINATION
ΔV_C = 5622 FT/SEC
TOTAL ΔV_C = 5731
TERMINAL $\gamma \cong 0°$

FIGURE 1

NASA-S-66-5039 JUN

VARIATION OF POWERED-DESCENT CHARACTERISTIC VELOCITY WITH THRUST-TO-WEIGHT RATIO

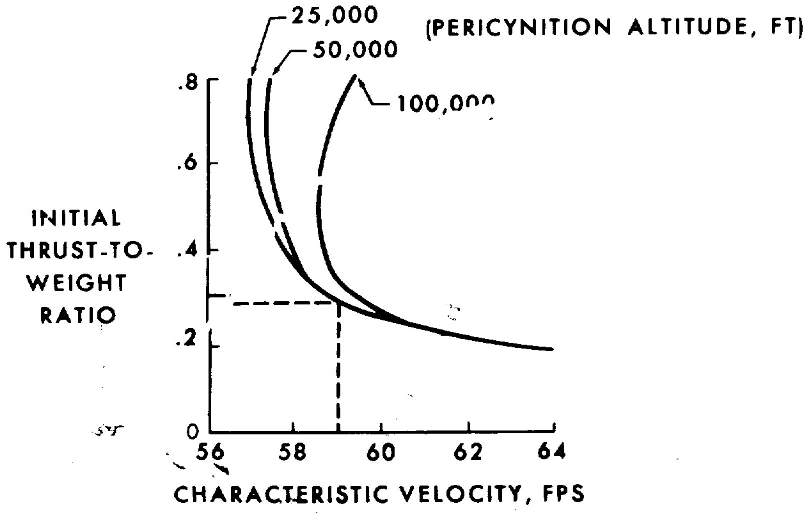
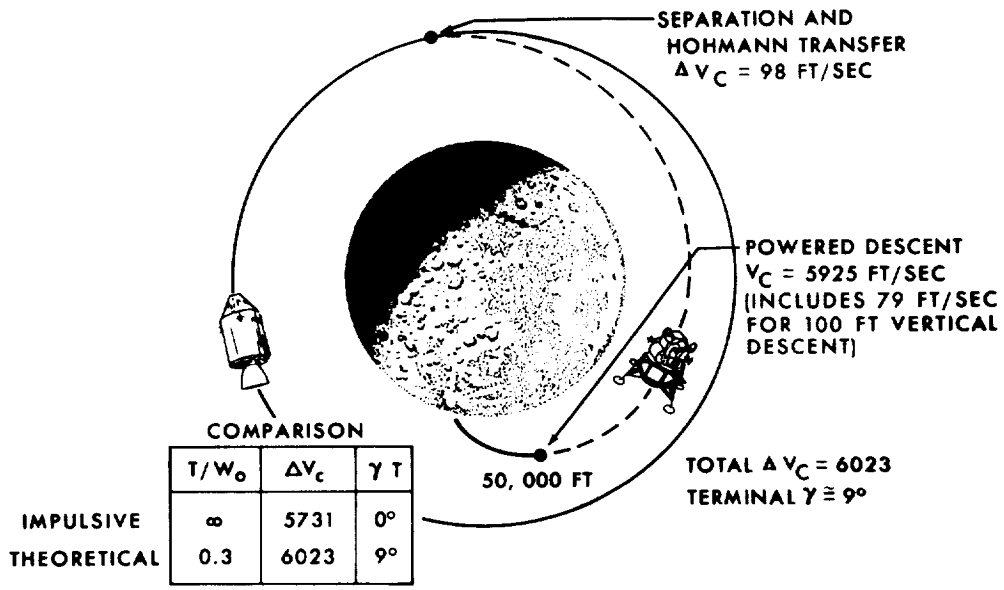


FIGURE 2

NASA-S-66-6026 MAY

THEORETICAL OPTIMUM LM DESCENT

(T/W_O = .3, H_P = 50,000 FT)


SEPARATION AND
HOHMANN TRANSFER
ΔV$_C$ = 98 FT/SEC
POWERED DESCENT
V$_C$ = 5925 FT/SEC
(INCLUDES 79 FT/SEC
FOR 100 FT VERTICAL
DESCENT)
50, 000 FT
TOTAL ΔV$_C$ = 6023
TERMINAL γ ≅ 9°
COMPARISON

	T/W_o	ΔV_c	γ T
IMPULSIVE	∞	5731	0°
THEORETICAL	0.3	6023	9°

FIGURE 3

NASA-S-66-5048 JUNE 1

OPTIMUM POWERED DESCENT

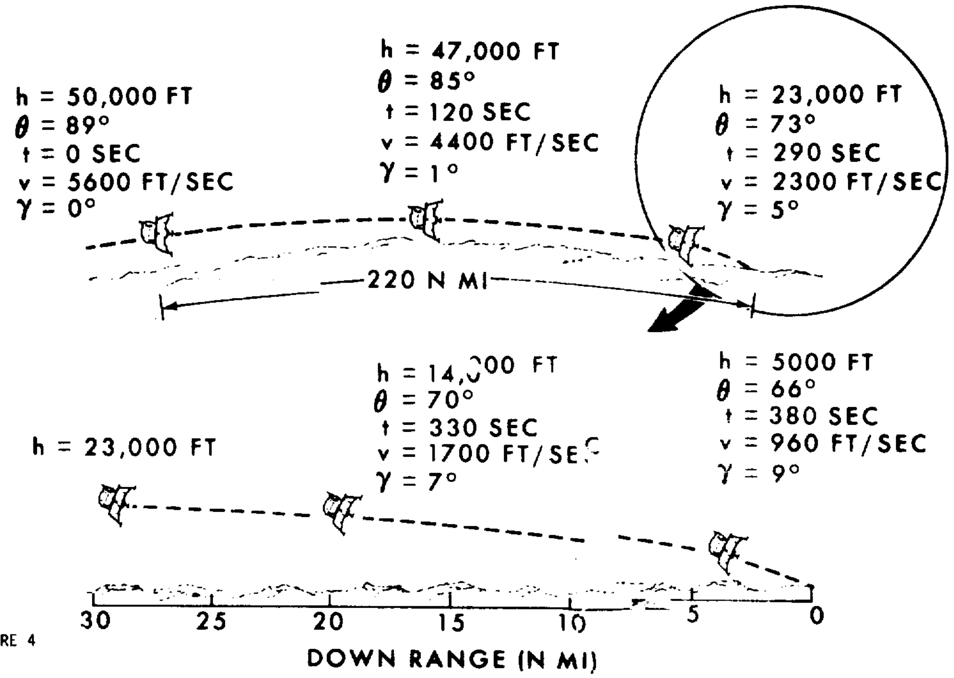

h = 50,000 FT
θ = 89°
t = 0 SEC
v = 5600 FT/SEC
γ = 0°
h = 47,000 FT
θ = 85°
t = 120 SEC
v = 4400 FT/SEC
γ = 1°
h = 23,000 FT
θ = 73°
t = 290 SEC
v = 2300 FT/SEC
γ = 5°
220 N MI
h = 23,000 FT
h = 14,000 FT
θ = 70°
t = 330 SEC
v = 1700 FT/SEC
γ = 7°
h = 5000 FT
θ = 66°
t = 380 SEC
v = 960 FT/SEC
γ = 9°
30
25
20
15
10
5
0
DOWN RANGE (N MI)

FIGURE 4

NASA-S-66-6418 JUN

LM LANDING PLANNING STRATEGY

- OBJECTIVE
  - TO ANTICIPATE THE LUNAR ENVIRONMENT PROBLEMS AND TO PLAN THE LANDING APPROACH SO THAT THE COMBINED SPACECRAFT SYSTEMS INCLUDING THE CREW WILL MOST EFFECTIVELY IMPROVE THE PROBABILITY OF ATTAINING A SAFE LANDING
- MAJOR FACTORS
  - ORBITAL MECHANICS PROBLEMS
  - PERFORMANCE LIMITATIONS OF SPACECRAFT SYSTEMS
  - LUNAR ENVIRONMENT-VISIBILITY, TERRAIN UNCERTAINTIES, AND IRREGULARITIES
- PREDOMINANT SC SYSTEMS
  - GUIDANCE AND CONTROL
  - LANDING RADAR
  - DESCENT PROPULSION
  - SC WINDOW

FIGURE 5

NASA-S-66-6503 JUN

LM LANDING ACCURACY AFTER THREE ORBITS

NAVIGATION PHASE CONTRIBUTION		DOWN-RANGE σ(FT)	CROSS TRACK σ(FT)	CEP (FT)	ALTITUDE σ(FT)
LM SEPARATION AND HOHMANN DESCENT		1070	60	730	540
POWERED DESCENT		260	1410	1000	1490
RSS OF THE ABOVE TWO		1100	1410	1480	1580
LUNAR ORBIT NAVIGATION	MSFN	2320	700	1750	840
	ONBOARD	2840	540	1990	1180
TOTAL ACCURACY	MSFN	2570	1570	2410	1790
	ONBOARD	3040	1510	2630	1970

FIGURE 6A

NASA-S-66-6504 JUN

LM LANDING ACCURACY AFTER THREE ORBITS(CONT)

ASSUMPTIONS & ERROR MODELS (1σ)

- LANDING SITE AT 0° LATITUDE AND 0° LONGITUDE
- MSFN UPDATE PRIOR TO LUNAR ORBIT INSERTION
- TWO LANDMARKS WITH THREE SIGHTINGS PER LANDMARK PER PASS
- LM SEPARATION FROM CSM ON THIRD ORBIT, PLATFORM ALINEMENT AT 15 MINUTES BEFORE A MANEUVER

ACCEL BIAS	.0017 FT/SEC^2	SCANNING TELESCOPE	.06 DEG
ALINEMENT ACCURACY (ACT)	.06 DEG	GYRO DRIFT	.03 DEG/HR
LANDMARK ACCURACY	7500 FT		

FIGURE 6B

NASA-S-66-6522 JUN

LM LANDING 3σ UNCERTAINTY ELLIPSE AFTER THREE ORBITS

LANDING SITE 0° LAT 0° LONG

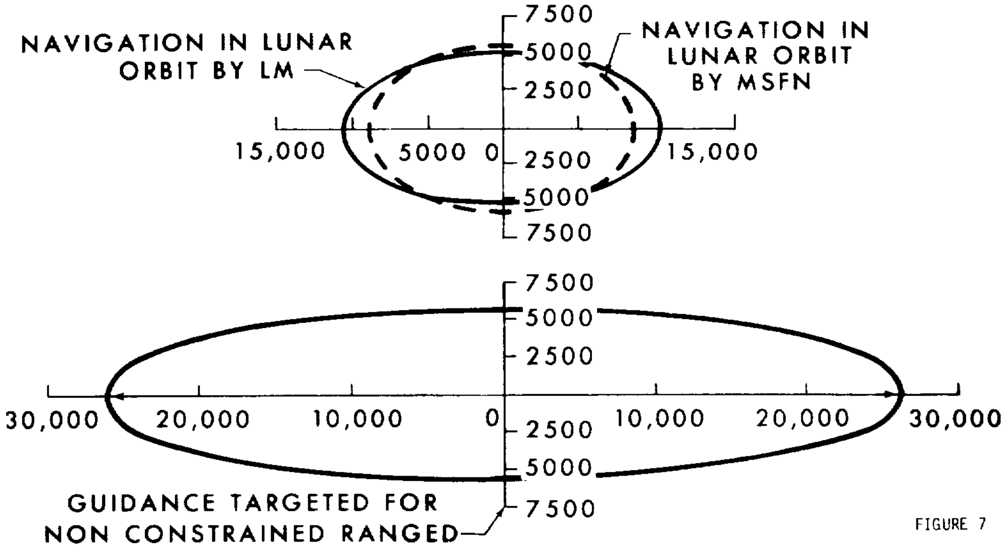
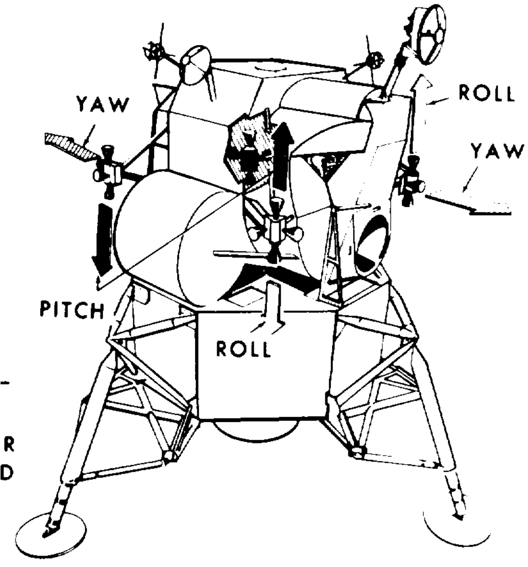


FIGURE 7

NASA-S-65-1684

ATTITUDE CONTROL OF LM


YAW
ROLL
YAW
PITCH
ROLL

NOTE:
IN DECENT THRUST CONFIGURATION MAIN ENGINE GIMBAL IS EMPLOYED FOR TRIMMING THE PITCH AND YAW MOMENT DUE TO CENTER OF GRAVITY SHIFTS

FIGURE 8

NASA-S-66-5050 JUNE 1

LANDING RADAR
BEAM CONFIGURATION AND ANTENNA TILT ANGLES

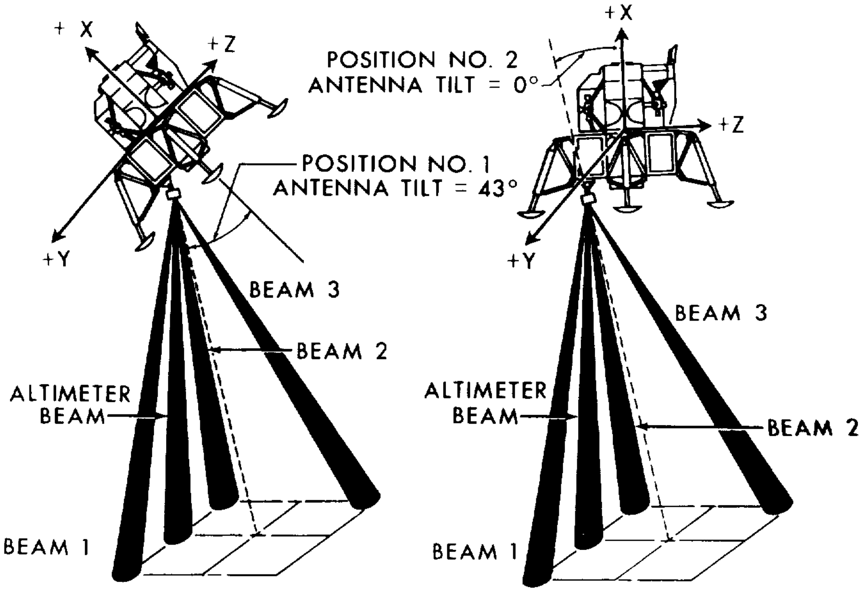

+ X
+ Z
+ Y
POSITION NO. 1
ANTENNA TILT = 43°
BEAM 3
BEAM 2
ALTIMETER BEAM
BEAM 1
+X
POSITION NO. 2
ANTENNA TILT = 0°
+Z
+Y
BEAM 3
ALTIMETER BEAM
BEAM 2
BEAM 1

FIGURE 9

NASA S 66 6479 JUN

LM LANDING RADAR (3σ) SPECIFICATION ACCURACY

ALTITUDE, FT	ACCURACY		
	RANGE TO SURFACE	V_{XA}	V_{YA}, V_{ZA}
5 - 200	1.5% + 5 FT	1.5% OR 1.5 FPS	2.0% OR 1.5 FPS
200 - 2000	1.5% + 5 FT	1.5% OR 1.5 FPS	3.5% OR 3.5 FPS
2000 - 25,000	1.5% + 5 FT	1.5% OR 1.5 FPS	2.0% OR 2.0 FPS
25,000 - 40,000	2%	N/A	N/A

FIGURE 10

NASA-S-66-6140 JUN

LM FLIGHT CONFIGURATION

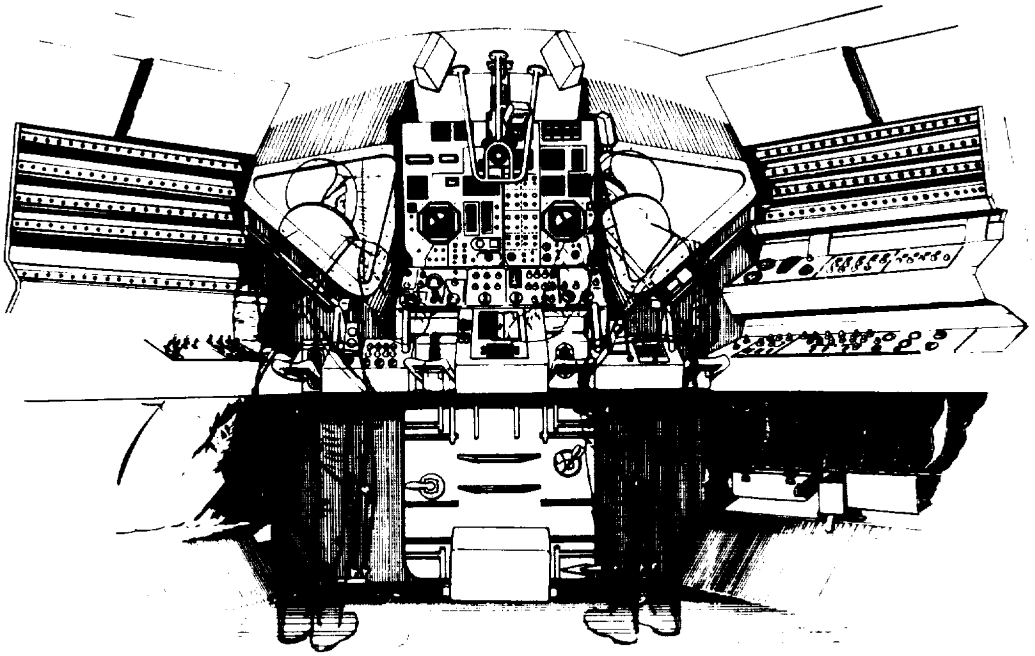


FIGURE 11

NASA-S-66-5045 JUNE 1

LM WINDOW VIEWING LIMITS FROM COMMANDER'S DESIGN EYE POSITION

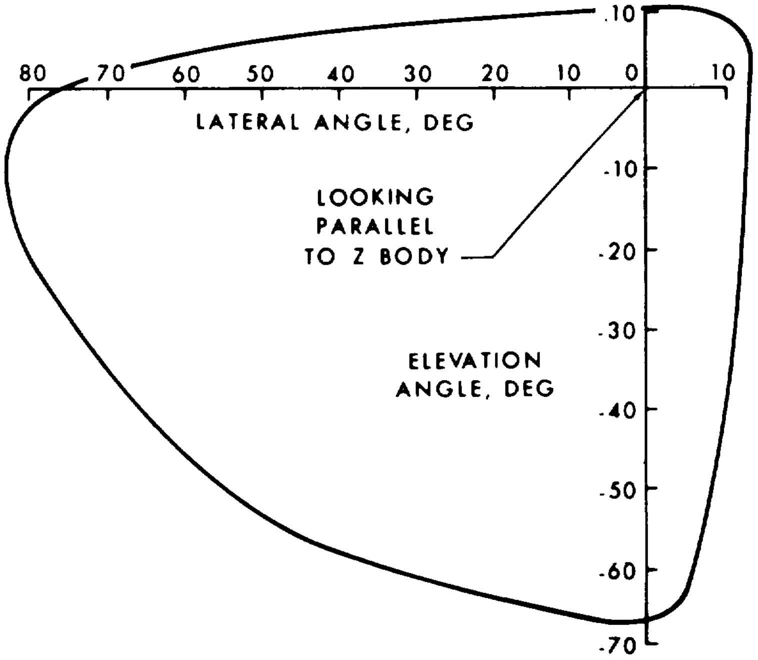


FIGURE 12

NASA-S-66-3576 MAY 12

LM DESCENT ENGINE THRUST CHARACTERISTICS

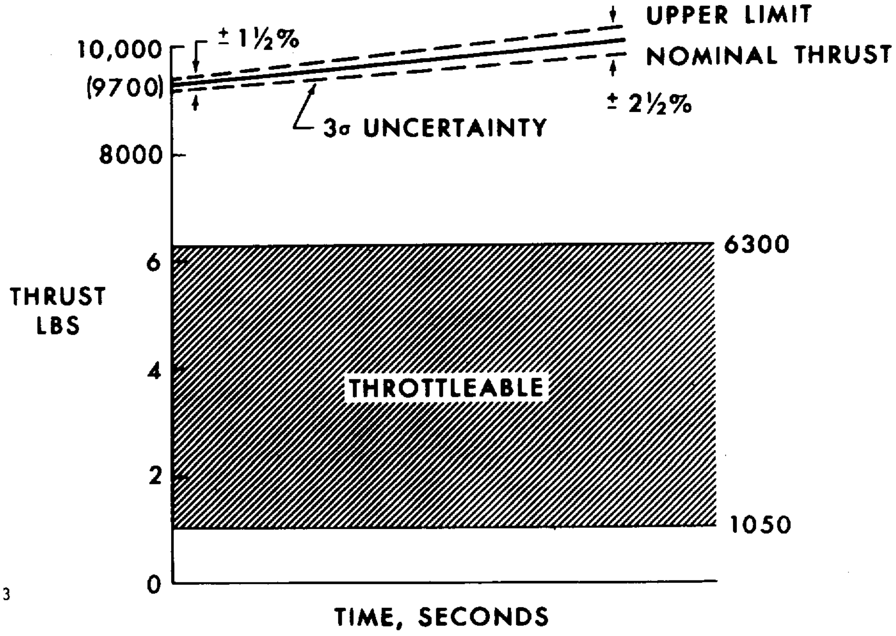


FIGURE 13

NASA-S-66-6470 JUN

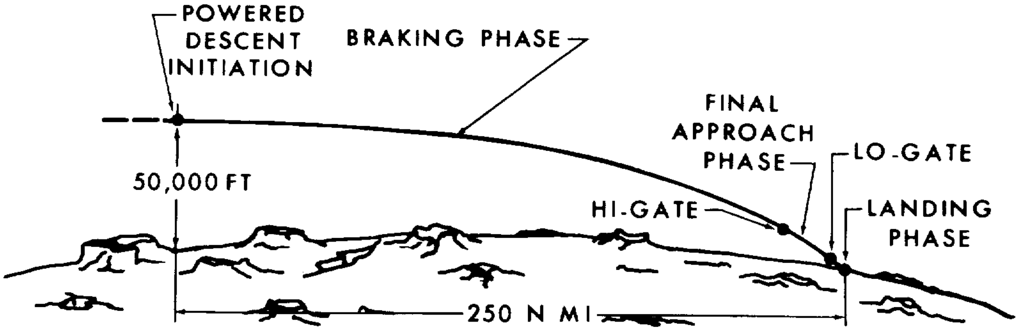
VARIATION OF LM LANDING POSITION REQUIREMENTS THAT HAVE BEEN CONSIDERED

- LANDING AT ANY SUITABLE POINT WITHIN A SPECIFIED AREA
- LANDING AT ANY SUITABLE POINT WITHIN A SMALL AREA CONSTRAINED IN SIZE PRIMARILY BY GUIDANCE DISPERSIONS*
- LANDING AT A PRESPECIFIED POINT (SUCH AS A SURVEYOR)

FIGURE 14 *PRESENT STRATEGY IS BASED UPON THIS REQUIREMENT

NASA-S-66-5044 JUN

LM THREE-PHASED POWERED DESCENT


POWERED DESCENT INITIATION
BRAKING PHASE
FINAL APPROACH PHASE
LO-GATE
50,000 FT
HI-GATE
LANDING PHASE
250 N MI

- BRAKING PHASE - ALLOWS EFFICIENT REDUCTION OF MOST OF VELOCITY
- FINAL APPROACH PHASE - ALLOWS ACQUISITION AND ASSESSMENT OF SITE AND CONFIRMATION OF FLIGHT SAFETY BY PILOT
- LANDING PHASE-ALLOWS VERNIER CONTROL OF POSITION AND VELOCITIES

FIGURE 15

NASA-S-66-5418 MAY 31

LM POWERED DESCENT

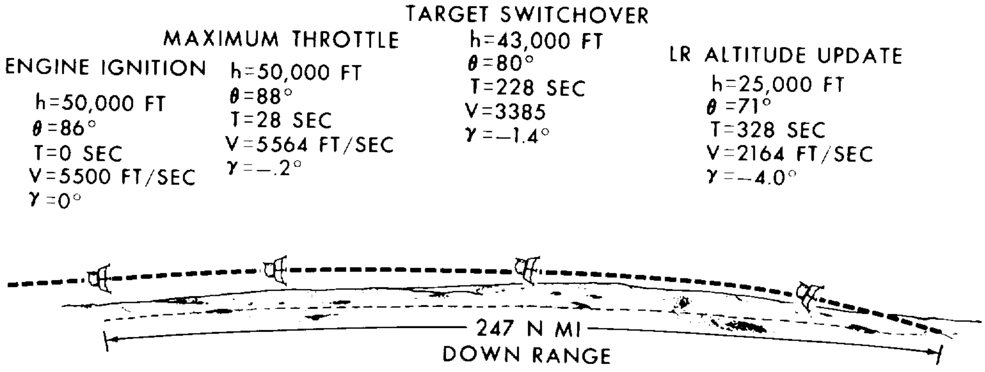

ENGINE IGNITION
h=50,000 FT
θ=86°
T=0 SEC
V=5500 FT/SEC
γ=0°
MAXIMUM THROTTLE
h=50,000 FT
θ=88°
T=28 SEC
V=5564 FT/SEC
γ=−.2°
TARGET SWITCHOVER
h=43,000 FT
θ=80°
T=228 SEC
V=3385
γ=−1.4°
LR ALTITUDE UPDATE
h=25,000 FT
θ=71°
T=328 SEC
V=2164 FT/SEC
γ=−4.0°
247 N MI
DOWN RANGE

FIGURE 16A

NASA-S-66-5414 MAY 31

LM POWERED DESCENT (CONT)

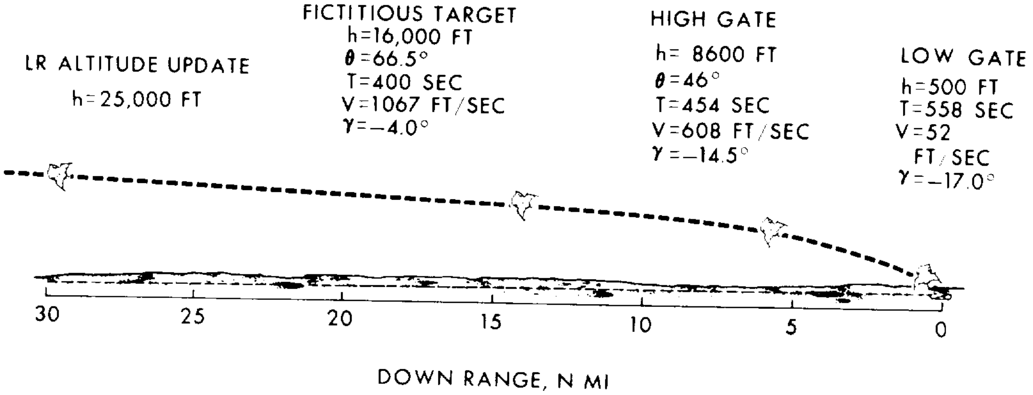

LR ALTITUDE UPDATE
h=25,000 FT
FICTITIOUS TARGET
h=16,000 FT
θ=66.5°
T=400 SEC
V=1067 FT/SEC
γ=−4.0°
HIGH GATE
h= 8600 FT
θ=46°
T=454 SEC
V=608 FT/SEC
γ=−14.5°
LOW GATE
h=500 FT
T=558 SEC
V=52
FT/SEC
γ=−17.0°
30
25
20
15
10
5
0
DOWN RANGE, N MI

FIGURE 16B

NASA-S-66-6476 JUN

BRAKING PHASE DESIGN

- OBJECTIVES
  - REDUCE VELOCITY TO ACCEPTABLE LANDING APPROACH MAGNITUDES
  - MAINTAIN EFFICIENT USE OF PROPELLANT FUEL
  - REACH A PRESPECIFIED STATE VECTOR AT HI-GATE POSITION
- CONSTRAINTS
  - DESCENT ENGINE IS NON-THROTTLEABLE IN MAX THRUST REGION
  - MAXIMUM THRUST OF DESCENT ENGINE IS INITIALLY≈9700 LBS (T/W≅.3)
  - FIXED THRUST UNCERTAINTIES MAY REACH ±2 1/2%

FIGURE 17

NASA-S-66-6440 JUN

POWERED DESCENT IGNITION LOGIC

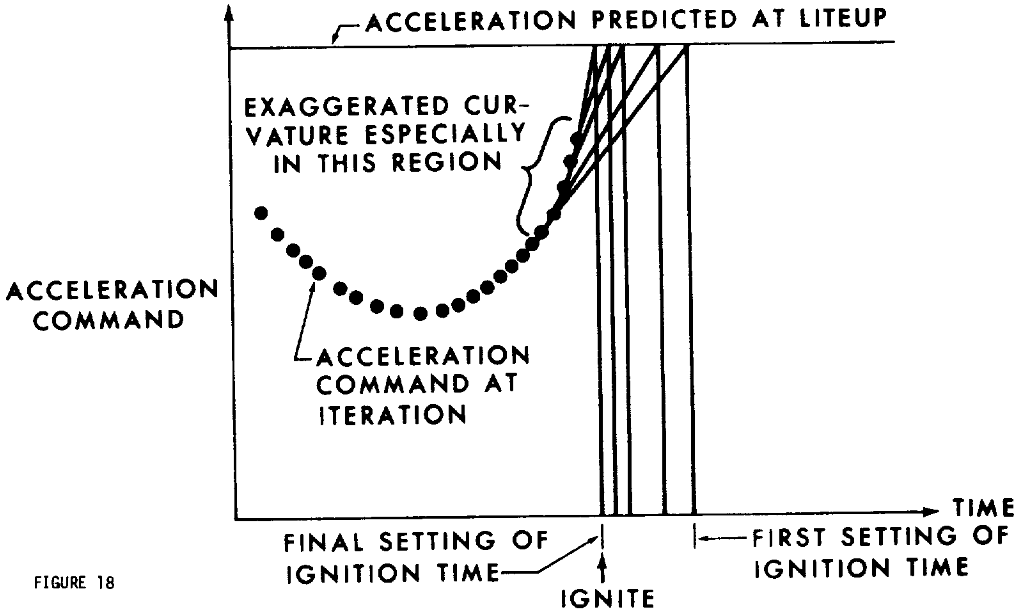

ACCELERATION PREDICTED AT LITEUP
EXAGGERATED CURVATURE ESPECIALLY IN THIS REGION
ACCELERATION COMMAND
ACCELERATION COMMAND AT ITERATION
TIME
FINAL SETTING OF IGNITION TIME
IGNITE
FIRST SETTING OF IGNITION TIME

FIGURE 18

NASA-S-66-6425 JUN

THRUST BEHAVIOR FOR LIMITED THROTTLE GUIDANCE

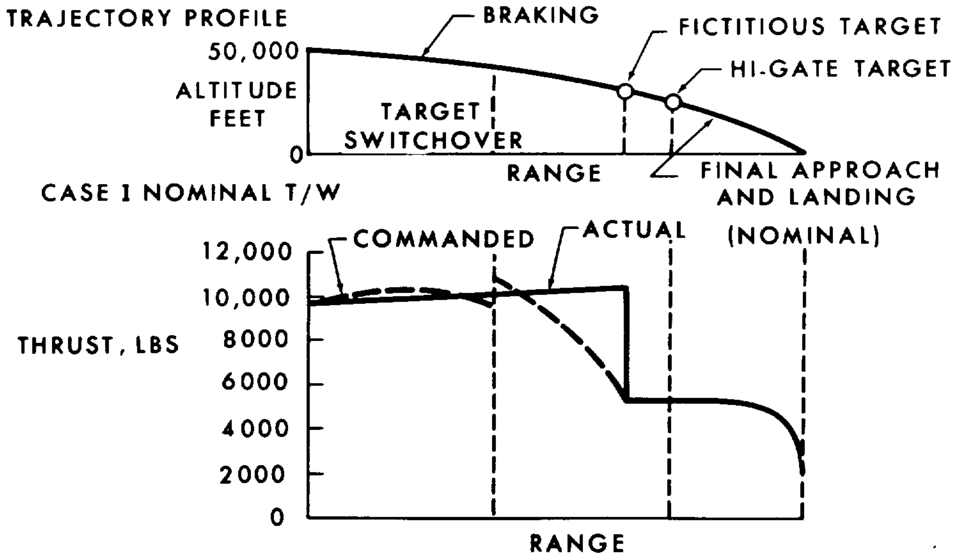


FIGURE 19

NASA-S-66-5042 JUN

THRUST BEHAVIOR FOR LIMITED THROTTLE GUIDANCE

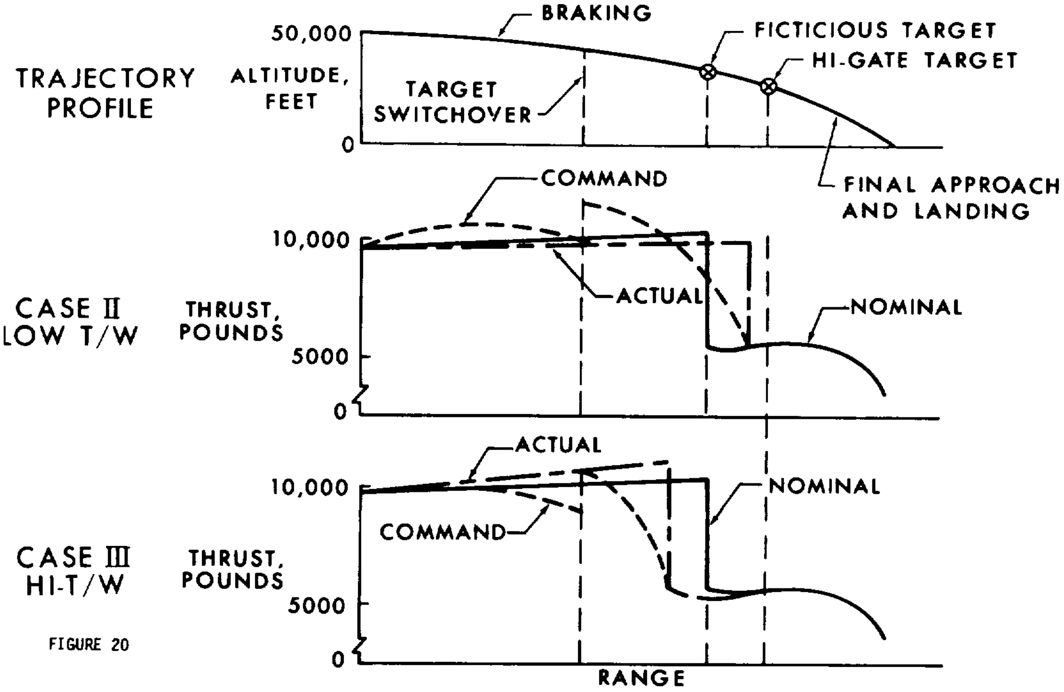


FIGURE 20

NASA-S-66-3043 APR 5

ΔV PENALTY DUE TO FIXED THRUST UNCERTAINTIES

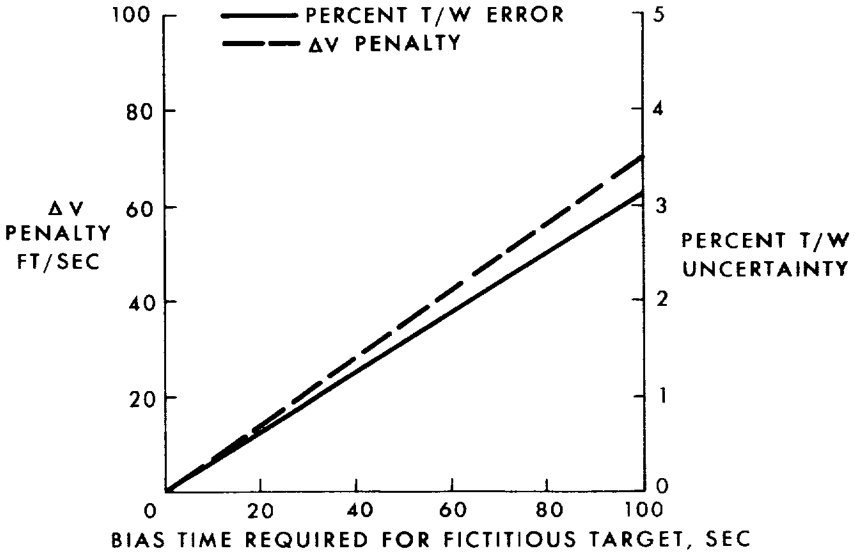


FIGURE 21

NASA-S-66-6426 JUN

LANDING RADAR WEIGHTING FACTORS FOR ALTITUDE AND VELOCITY UPDATES

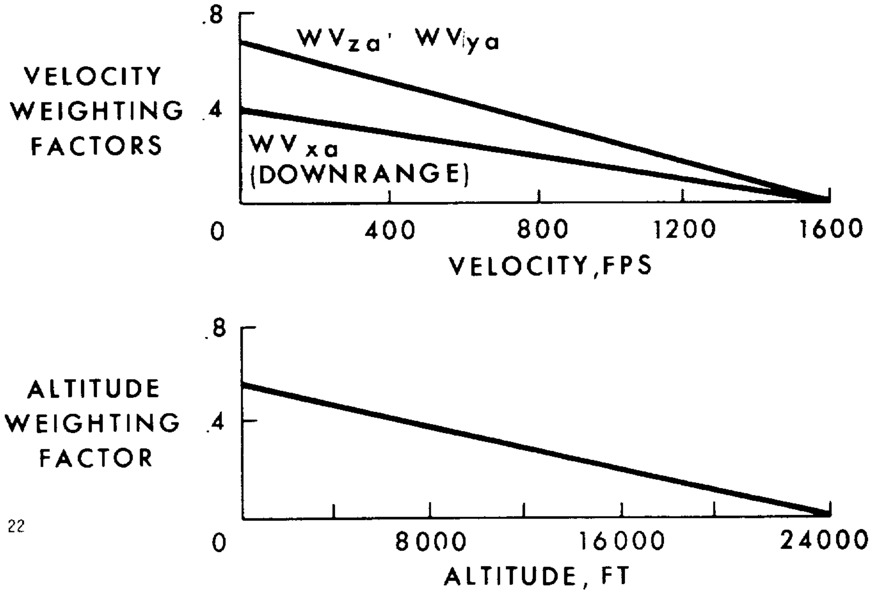
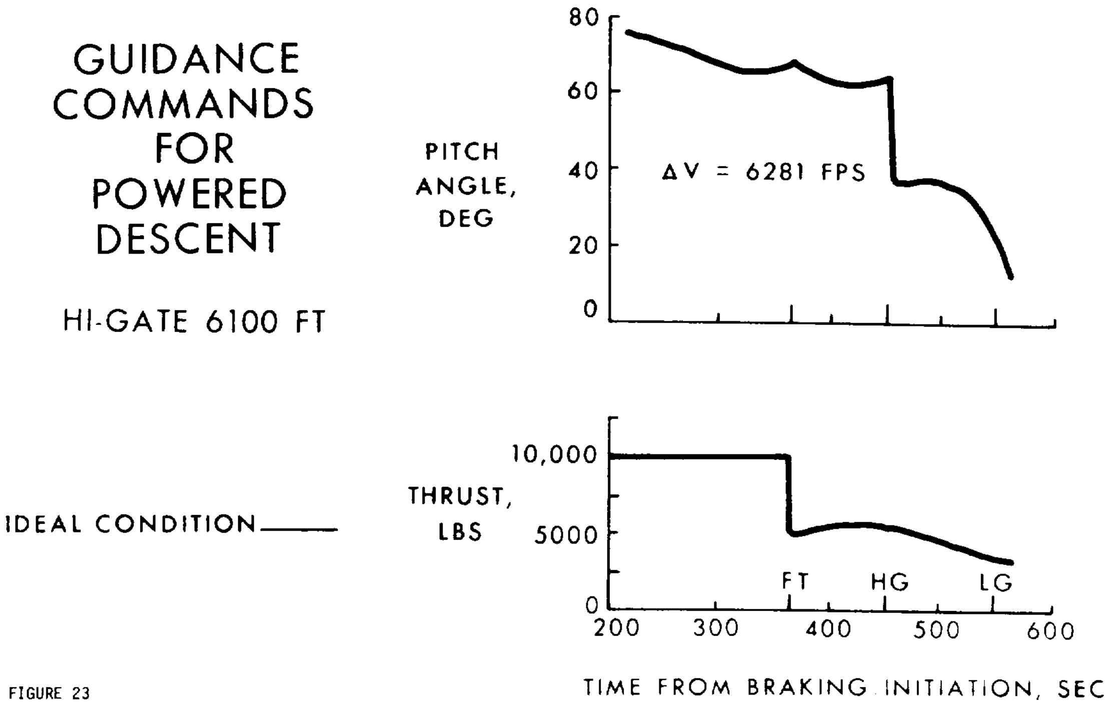


FIGURE 22



GUIDANCE
COMMANDS
FOR
POWERED
DESCENT

HI-GATE 6100 FT

FIGURE 23

NASA-S-66-6513 JUN

TERRAIN PROFILE DURING APPROACH TO LANDING SITE

0 20 N LAT 12 50 E LONG

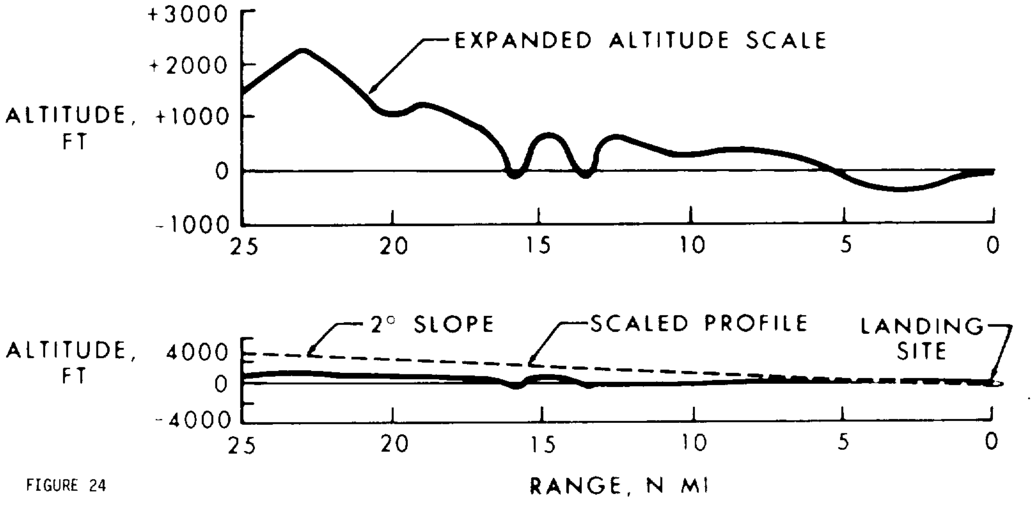
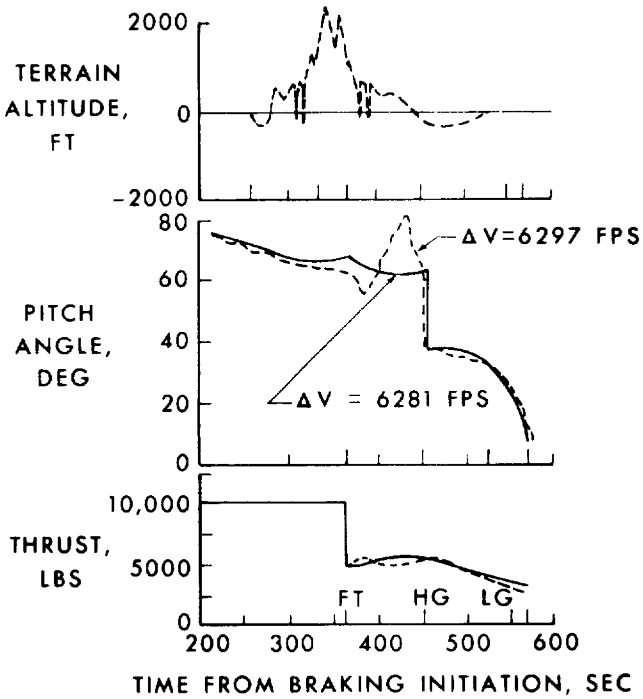


FIGURE 24

NASA-S-66-6515 JUN

GUIDANCE COMMANDS FOR POWERED DESCENT



IDEAL CONDITION ———

TYPICAL ERROR CONDITIONS AND TERRAIN -----

FIGURE 25

NASA S 66 6483 JUN

LM POWERED DESCENT GUIDANCE MONITORING

- PURPOSE OF MONITORING
  - PROVIDE ASSESSMENT OF TRAJECTORY
  - FAILURE DETECTION AND ISOLATION
  - ASSURE SAFE ABORT
- TWO TECHNIQUES
  - MONITORING TRAJECTORY BOUNDS OF PNGS AND AGS
  - PERIODIC DIFFERENCING OF PNGS AND AGS
- ALTITUDE - ALTITUDE RATE MOST SIGNIFICANT FOR ABORT SAFETY
- ALTITUDE RATE DEVIATIONS MOST SENSITIVE TO FAILURE DETECTION
- MSFN MEASUREMENT OF ALTITUDE RATE SHOULD BE SUFFICIENT FOR FAILED SYSTEM ISOLATION
- 3σ GUIDANCE DEVIATIONS WILL NOT ENDANGER FLIGHT PRIOR TO HI-GATE

FIGURE 26

NASA-S-66-6441 JUN

PHASE II - FINAL APPROACH DESIGN

- OBJECTIVES
  - PROVIDE CREW VISIBILITY OF AND ADEQUATE TIME TO ASSESS LANDING AREA
  - PROVIDE CREW OPPORTUNITY TO ASSESS FLIGHT SAFETY
  - PROVIDE A RELATIVELY STABLE VIEWING PLATFORM
- CONSTRAINTS
  - FUEL LIMITATIONS
  - LM WINDOW SIZE
  - T/W OF DESCENT AND ASCENT ENGINE AND REQUIREMENT FOR SAFE STAGED ABORTS
  - TERRAIN LIGHTING/CONTRAST PROPERTIES
- VARIABLES
  - PITCH ATTITUDE
  - TRANSITION ALTITUDE
  - FLIGHT PATH ANGLE
  - LOOK ANGLE TO LANDING AREA REFERENCED TO THRUST AXIS

FIGURE 27

NASA-S-66-6402 JUN

FACTORS AFFECTING CHOICE OF HI-GATE ALTITUDE

- RANGE FROM WHICH LANDING AREA CAN BE ASSESSED
- TIME REQUIRED TO ASSESS LANDING AREA
- FLIGHT SAFETY REQUIREMENTS WITH REGARDS TO TERRAIN ALTITUDE UNCERTAINTIES, LANDING RADAR OPERATING RELIABILITY, AND ASCENT ENGINE ABORT BOUNDARY

FIGURE 28

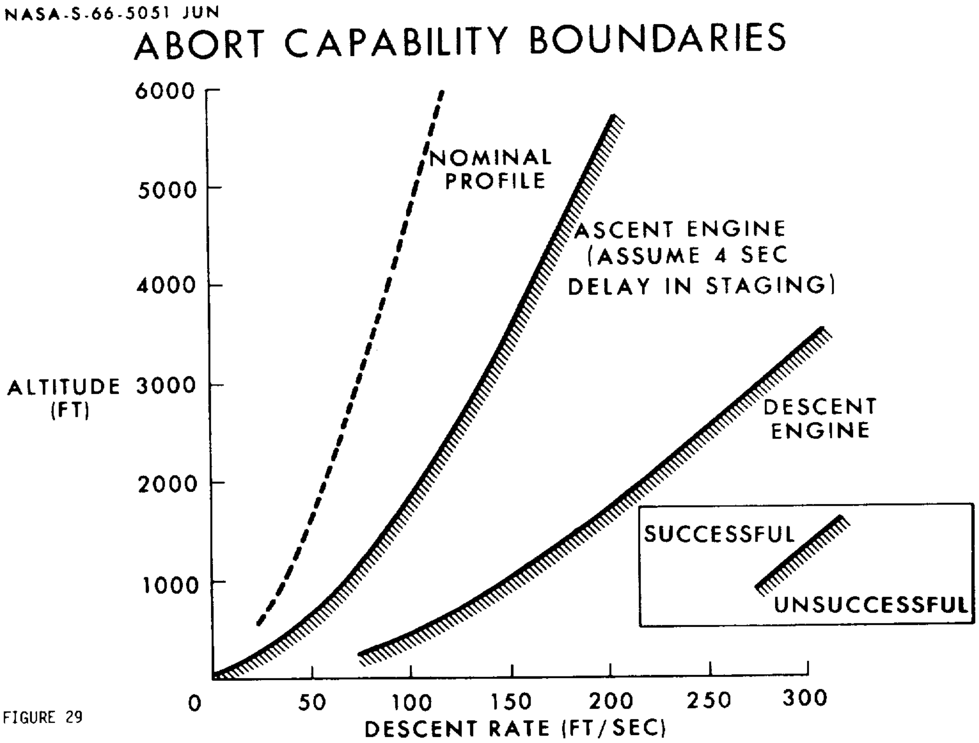


FIGURE 29

NASA-S-66-5041 JUN

FACTORS CONTRIBUTING TO UNCERTAINTIES IN ALTITUDE ABOVE TERRAIN

● GUIDANCE AND NAVIGATION UNCERTAINTIES	(1500 FT ALT 1σ)
● LUNAR RADIUS BIAS MAGNITUDE	(3200 FT ALT)
● LUNAR RADIUS RANDOM MAGNITUDE	(3200 FT ALT 1σ)
● PRESENT ABILITY TO DETERMINE MARIA AREA SLOPES (±3° 3σ)	(FUNCTION OF LANDING DISPERSIONS)
● ALLOWABLE TERRAIN VARIATIONS WITHIN ±2° SLOPE AND ±5% OF NOMINAL ALTITUDE	(FUNCTION OF LANDING DISPERSIONS)

FIGURE 30

NASA-S-66-6471 JUN

DETERMINATION OF MINIMUM HI-GATE ALTITUDE WITHOUT LR UPDATING

ORBIT NAVIGATION	3σ ALTITUDE UNCERTAINTIES,* FT			ALTITUDE BIASES, FT			MINIMUM HI-GATE ALTITUDE FT
	PGNCS	TERRAIN PROFILE	LUNAR RADIUS	LUNAR RADIUS	TERRAIN PROFILE	STAGED ABORT	
MSFN	3700	4700	13,700	9800	4300	3500	32,600
MSFN & LANDING SITE UPDATE	3700	700	1700	—	700	1800	6700
PGNCS & LANDING SITE UPDATE	4500	1000	1700	—	800	1800	7500

* 3σ UNCERTAINTIES ARE ROOT-SUM-SQUARED

FIGURE 31

NASA-S-66-6433 JUN

ΔV PENALTY FOR HI-GATE ALTITUDE VARIATION- TYPICAL FLIGHT PATH ANGLE ≈ 15°

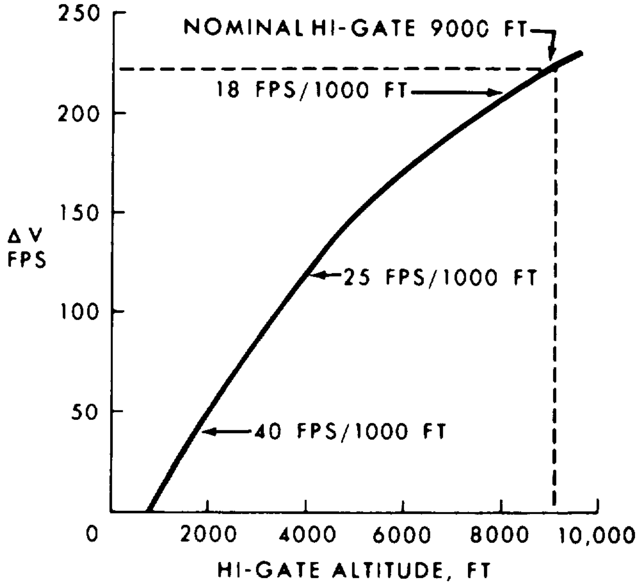


FIGURE 32

NASA S 66 6420 JUN

ΔV PENALTY FOR LOOK ANGLE AND FLIGHT PATH ANGLE (HI-GATE 9000FT)

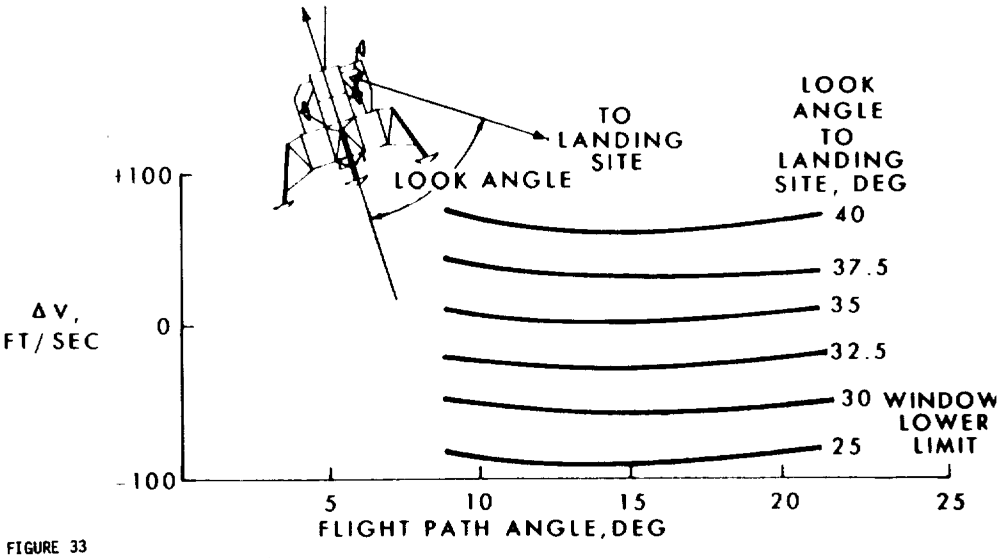


FIGURE 33

NASA-S-66-6495 JUN

COMPARISON OF DESIGN TRAJECTORY AND FUEL OPTIMUM

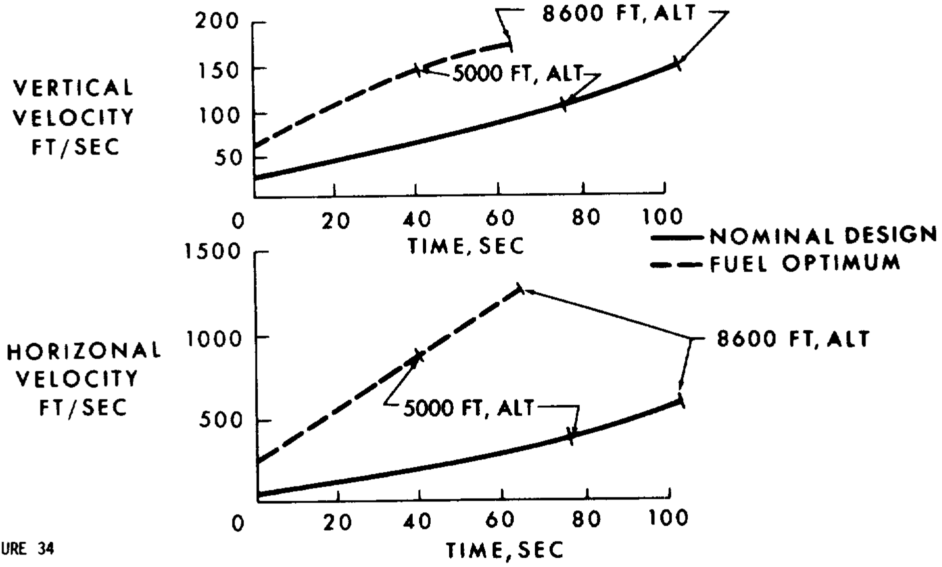


FIGURE 34

NASA-S-66-6579 JUN

OVERHEAD PROFILE OF FOOTPRINT CAPABILITY FROM 8000 FT ALTITUDE

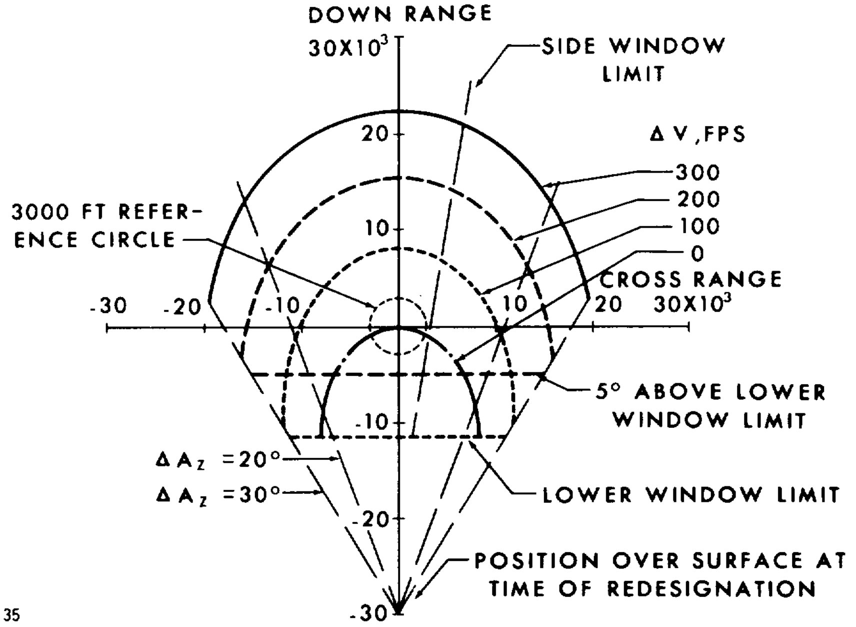

DOWN RANGE
30X10³
20
10
-10
-20
-30
SIDE WINDOW LIMIT
Δ V, FPS
300
200
100
0
CROSS RANGE
-30
-20
-10
10
20
30X10³
3000 FT REFERENCE CIRCLE
5° ABOVE LOWER WINDOW LIMIT
Δ A_Z = 20°
Δ A_Z = 30°
LOWER WINDOW LIMIT
POSITION OVER SURFACE AT TIME OF REDESIGNATION

FIGURE 35

NASA-S-66-3291 APR 16

VARIATION OF FOOTPRINT CAPABILITY WITH ALTITUDE

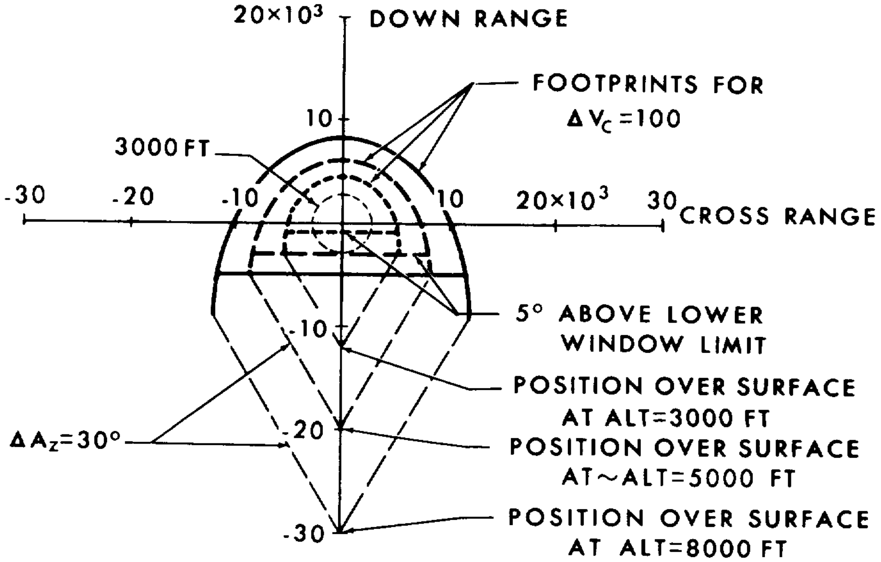
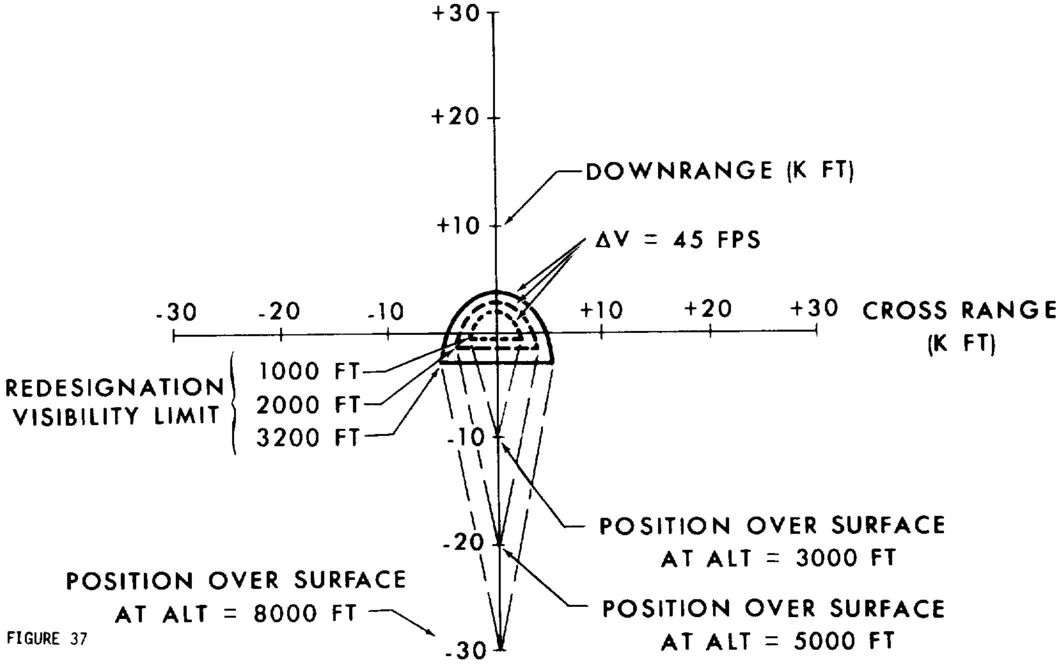

20×10³
DOWN RANGE
10
-10
-20
-30
FOOTPRINTS FOR ΔV_C = 100
3000 FT
-30
-20
-10
10
20×10³
30
CROSS RANGE
5° ABOVE LOWER WINDOW LIMIT
POSITION OVER SURFACE AT ALT=3000 FT
POSITION OVER SURFACE AT ~ALT=5000 FT
POSITION OVER SURFACE AT ALT=8000 FT
ΔA_Z=30°

FIGURE 36

NASA-S-66-5040 JUNE 1

VARIATION OF FOOTPRINT WITH ALTITUDE DURING DESCENT

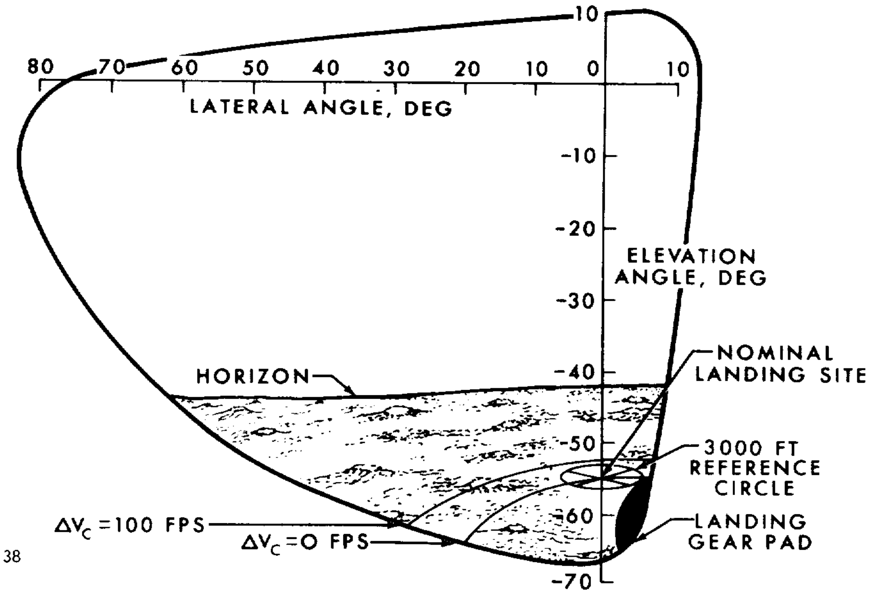


+30
+20
+10
-10
-20
-30
DOWNRANGE (K FT)
ΔV = 45 FPS
-30
-20
-10
+10
+20
+30
CROSS RANGE (K FT)
REDESIGNATION VISIBILITY LIMIT
1000 FT
2000 FT
3200 FT
POSITION OVER SURFACE AT ALT = 3000 FT
POSITION OVER SURFACE AT ALT = 5000 FT
POSITION OVER SURFACE AT ALT = 8000 FT

FIGURE 37

NASA-S-66-6537 JUN

LANDING FOOTPRINT AS SEEN BY PILOT FROM 8000 FT ALTITUDE



10
80
70
60
50
40
30
20
10
0
10
LATERAL ANGLE, DEG
-10
-20
-30
-40
-50
-60
-70
ELEVATION ANGLE, DEG
HORIZON
NOMINAL LANDING SITE
3000 FT REFERENCE CIRCLE
LANDING GEAR PAD
ΔV_C = 100 FPS
ΔV_C = 0 FPS

FIGURE 38

NASA-S-66-5046 JUNE 1

MARKING DETAILS
OF
LANDING POINT
DESIGNATOR GRID

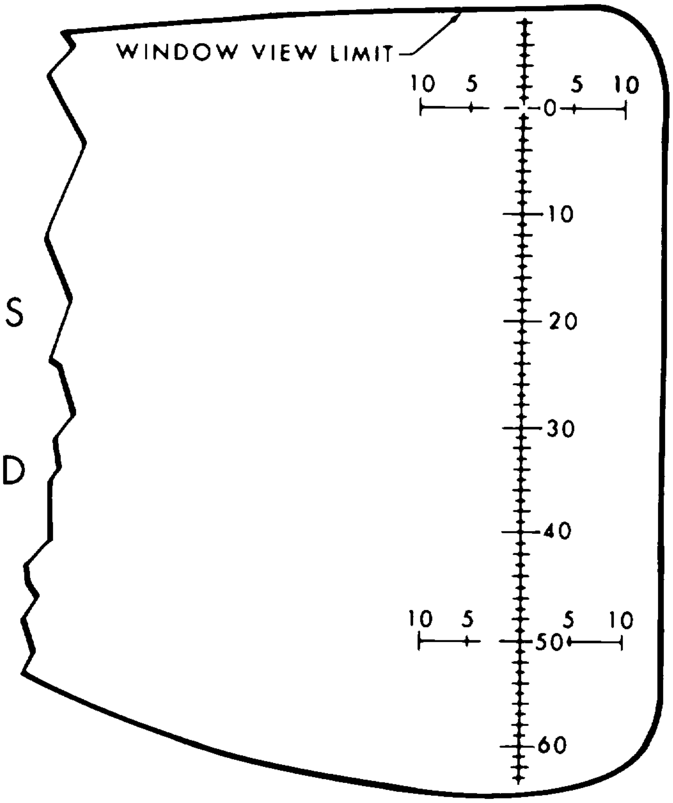

WINDOW VIEW LIMIT
10 5 0 5 10
10
20
30
40
10 5 50 5 10
60

FIGURE 39

NASA-S-66-6432 JUN

LANDING POINT DESIGNATION

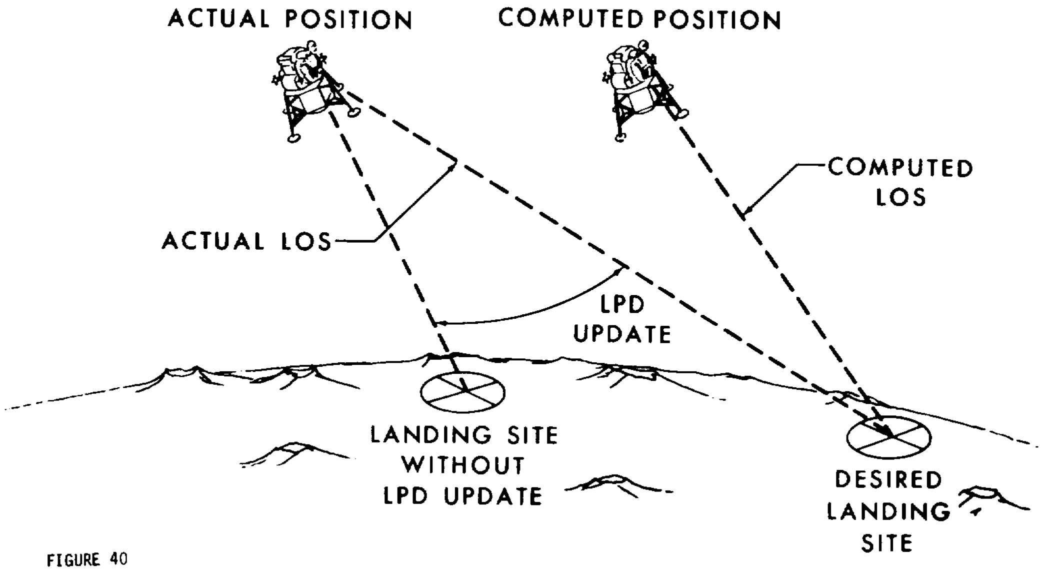

ACTUAL POSITION
COMPUTED POSITION
COMPUTED LOS
ACTUAL LOS
LPD UPDATE
LANDING SITE WITHOUT LPD UPDATE
DESIRED LANDING SITE

FIGURE 40

NASA-S-66-6474 JUN

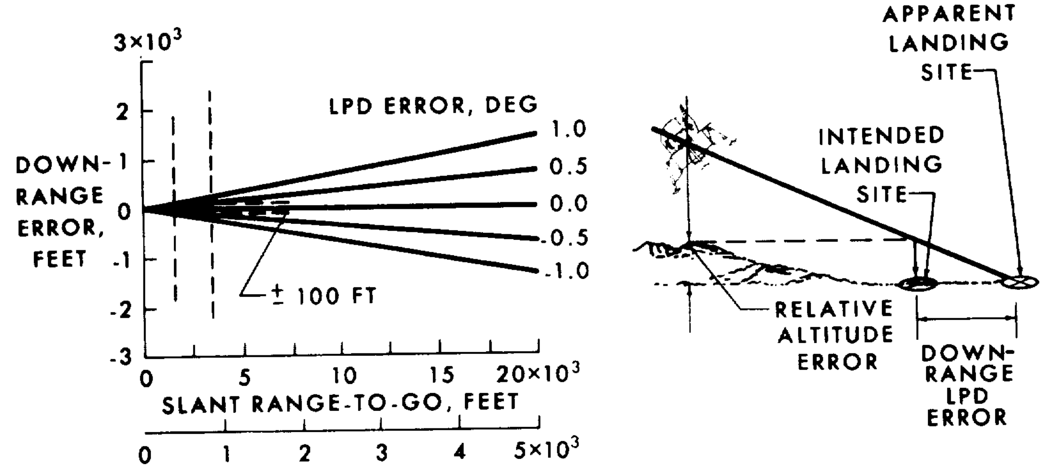
REDESIGNATION ERROR SOURCES

- TERRAIN
- GUIDANCE ALTITUDE DISPERSIONS
  (NON UPDATED)
- BORE SIGHT INSTALLATION
- IMU ALINEMENT
- APPLICATION ERRORS

FIGURE 41

NASA-S-66-6626 JUL 6

LANDING POINT DESIGNATOR ERROR SOURCES
FLIGHT PATH ANGLE = 14°



NOTE: CROSSRANGE ERROR = ¼ DOWNRANGE ERROR FOR EQUIVALENT ANGLE

NOTE: DOWNRANGE ERROR IS APPROXIMATELY 4 TIMES ALTITUDE ERROR

FIGURE 42

NASA-S-66-6630 JUL 6

TIME HISTORIES OF LINE OF SIGHT TO LANDING POINT FOR ALTERNATE SITE SELECTIONS AT 5000FT ALTITUDE

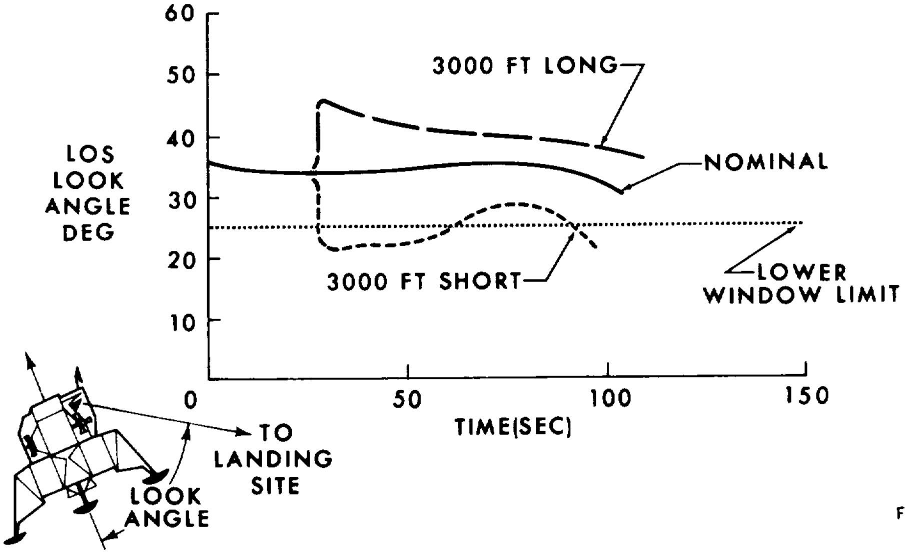
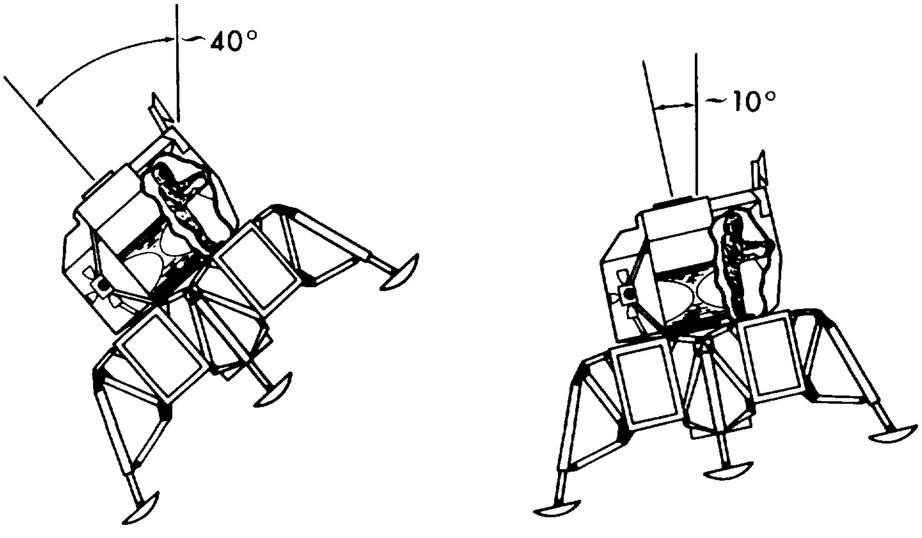


FIGURE 43

NASA-S-66-6448 JUN

LM CREW ATTITUDE RELATIVE TO LUNAR SURFACE



FINAL APPROACH PHASE

LANDING PHASE

FIGURE 44

NASA-S-66-6457 JUN

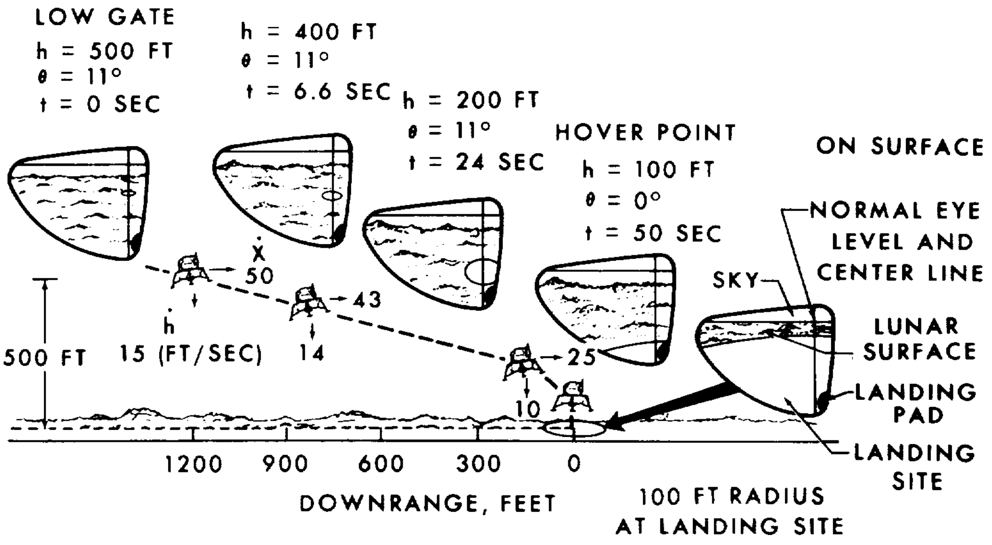
LANDING PHASE DESIGN

- OBJECTIVES
  - ALLOW DETAIL ASSESSMENT AND FINAL SELECTION OF LANDING POINT
  - ALLOW SOME MANEUVERING CAPABILITY AND FOOT-PRINT FOR LANDING POINT ADJUSTMENT
- CONSTRAINTS
  - FUEL UTILIZATION
  - WINDOW AND LIGHTING VISIBILITY
  - TERRAIN AND POSSIBLE DUST
  - LIMITED ATTITUDE FOR MANEUVERING
  - STAGED ABORT BOUNDARIES

FIGURE 45

NASA-S-66-5400 MAY 31

PILOT VIEW DURING LANDING PHASE



LOW GATE
h = 500 FT
θ = 11°
t = 0 SEC
h = 400 FT
θ = 11°
t = 6.6 SEC
h = 200 FT
θ = 11°
t = 24 SEC
HOVER POINT
h = 100 FT
θ = 0°
t = 50 SEC
ON SURFACE
NORMAL EYE LEVEL AND CENTER LINE
SKY
LUNAR SURFACE
LANDING PAD
LANDING SITE
ẋ
50
43
25
10
ḣ
15 (FT/SEC)
14
500 FT
1200
900
600
300
0
DOWNRANGE, FEET
100 FT RADIUS AT LANDING SITE

FIGURE 46

NASA-S-66-3032 APR 5

TRAJECTORY CHARACTERISTICS FOR LANDING PHASE

FLIGHT PATH ANGLE = 17°
THRUST ACCELERATION = 5.46 FT/SEC^2
PITCH ANGLE,θ,= 11°

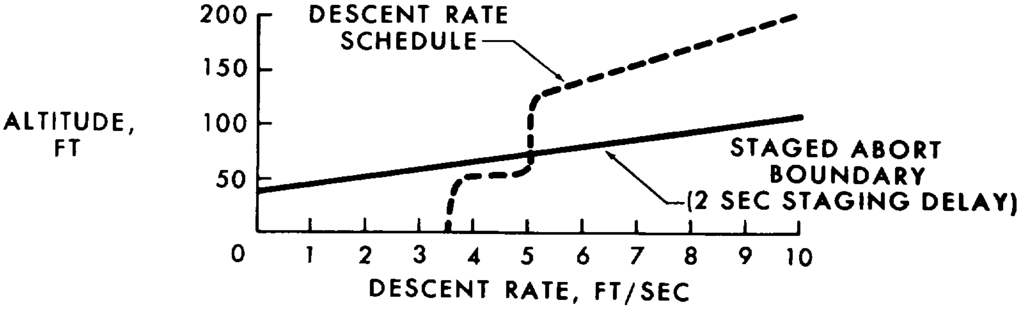


FIGURE 47

NASA-S-66-6467 JUN

LANDING PHASE FUEL BUDGET

BASELINE TRAJECTORY ALLOWANCE 390 FPS

● CONTINGENCY ALLOWANCE, FPS	MEAN	RANDOM (3σ)
• MANUAL CONTROL TECHNIQUE VARIATIONS	80	100
• EFFECT OF LANDING RADAR UNCERTAINTIES	80	65
• LANDING SITE INSPECTION	80	-
• FUEL DEPLETION MARGIN	40	-
TOTAL	280	119 (RSS)

FIGURE 48

NASA-S-66-6505 JUN

SUMMARY OF LM DESCENT BUDGET

BASELINE TRAJECTORY ALLOWANCES

PHASE		ΔV, FPS
DESCENT TRANSFER		97
POWERED DESCENT:	BRAKING	5135
	FINAL APPROACH	932
	LANDING	390
	SUBTOTAL	6554

CONTINGENCY ALLOWANCES

		MEAN	3σ
DESCENT TRANSFER - INCREASE CSM ALTITUDE 10 N MI		13	
BRAKING:	INCREASE CSM ALTITUDE 10 N MI	15	
	THRUST DISPERSIONS OF + 2%		48
	NAVIGATION ALT DISPERSIONS (3000 FT 3σ)		60
FINAL APPROACH - LANDING SITE UPDATE		45	15
LANDING:	MANUAL CONTROL VARIATIONS	80	100
	EFFECT OF LR UNCERTAINTIES	80	65
	LANDING SITE INSPECTION	80	
	FUEL DELETION MARGIN	40	
	SUBTOTAL	353	143 (RSS)
	TOTAL BUDGET	7050	

FIGURE 49

NASA-S-66-6539 JUN

TIME HISTORY OF LM DESCENT FUEL EXPENDITURE

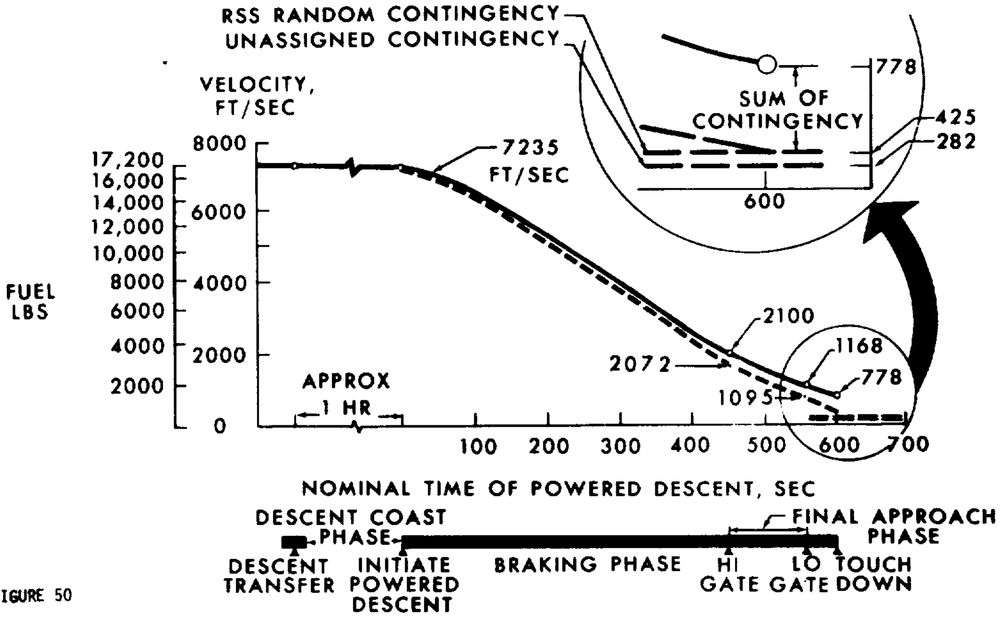


FIGURE 50

NASA-S-66-6403 JUN

LM LANDING TOUCHDOWN CONTROL

MODES

- AUTOMATIC
- MANUAL-AIDED BY AUTOMATIC CONTROL LOOPS
  - VISUAL
  - IFR (BECAUSE OF DUST OR LIGHTING)

MAJOR SOURCES OF SYSTEM ERRORS

- LANDING RADAR VELOCITY MEASUREMENT
- IMU MISALIGNMENT
- DISPLAY SYSTEM AND PILOT (MANUAL ONLY)
- CG POSITION

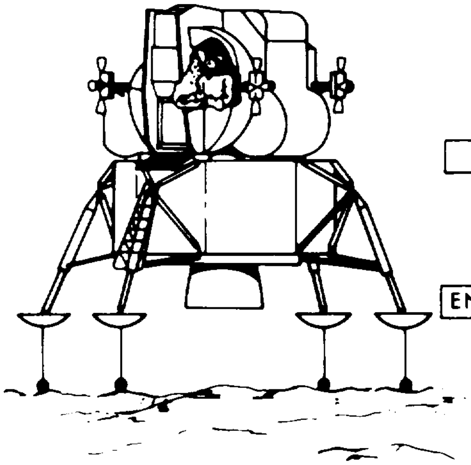
CONSTRAINTS

- LANDING GEAR DESIGN LIMITS
- DESCENT ENGINE REQUIRED TO BE OFF BY TOUCHDOWN

FIGURE 51

NASA-S-66-6466 JUN

DESCENT ENGINE SHUTDOWN SEQUENCE



PROBE CONTACT
↓
LIGHT ON
↓
PILOT ACTIVATES DE CUT-OFF $.2 < t < .66$
↓
DECA RESPONSE 0.10 SECONDS
↓
ENGINE FUEL VALVES CLOSE 0.15 SECONDS
↓
ENGINE TAIL OFF 0.12 SECONDS
↓
ENGINE OFF
↓
TOUCHDOWN

FIGURE 52

NASA-S-65-9321

LANDING DESCENT VELOCITY CONTROL USING PROBE FOR DESCENT ENGINE CUT-OFF SIGNAL

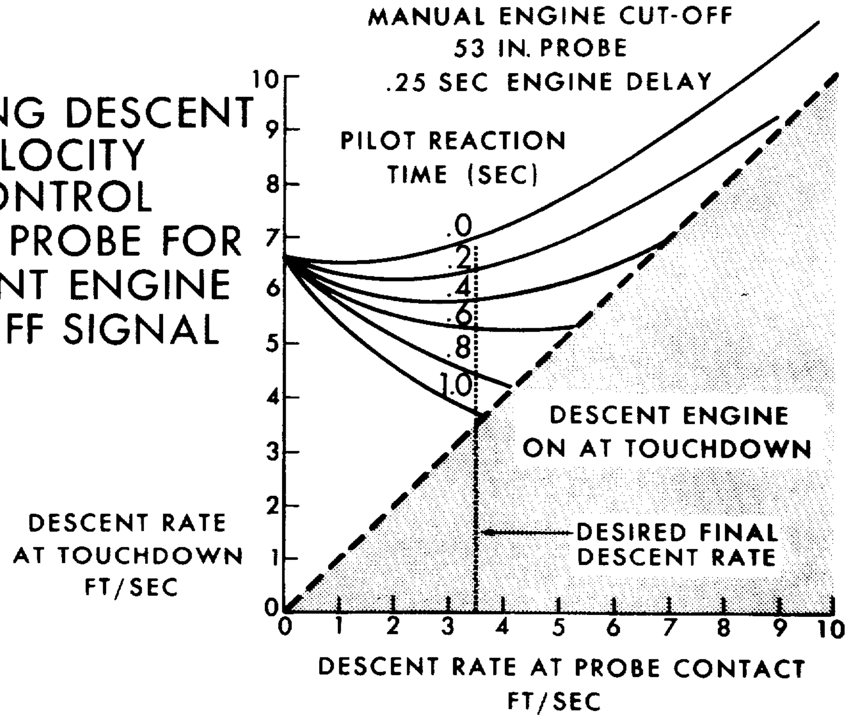


FIGURE 53

NASA-S-65-9324

LM DESCENT ENGINE CUT-OFF PILOT REACTION TIME

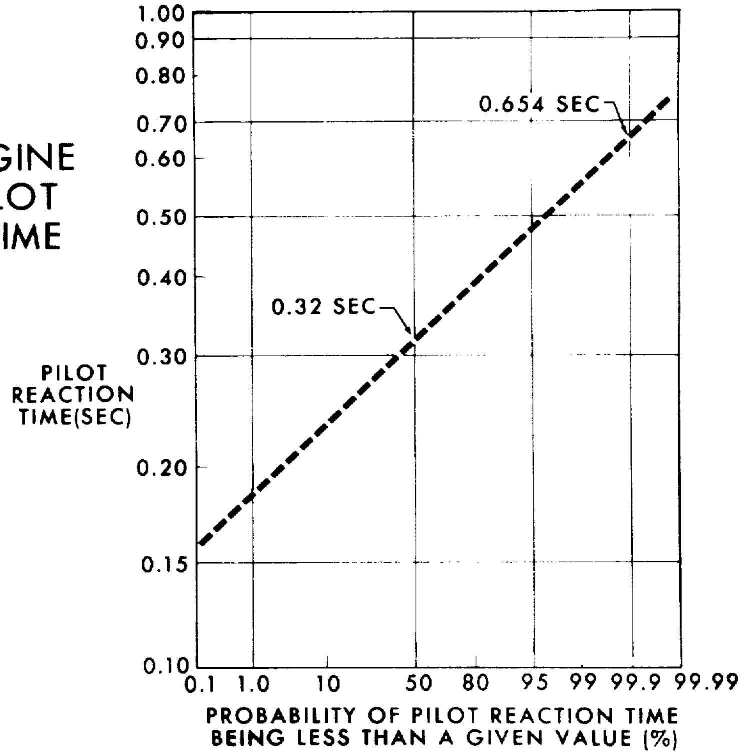


FIGURE 54

NASA-S-66-6468 JUN

ASSUMED LANDING RADAR ERROR MODEL FOR LANDING CONTROL ANALYSIS

	SPECIFICATION	PREDICTED	DEGRADED
VERTICAL	1.5 FT/SEC	.75 FT/SEC	1.5 FT/SEC
LATERAL	1.5 FT/SEC	1.5 FT/SEC	3.0 FT/SEC
FORWARD	1.5 FT/SEC	1.5 FT/SEC	3.0 FT/SEC

FIGURE 55

NASA-S-65-9331

MANUAL CONTROL OF LANDING VELOCITIES WITH SYSTEM ERRORS

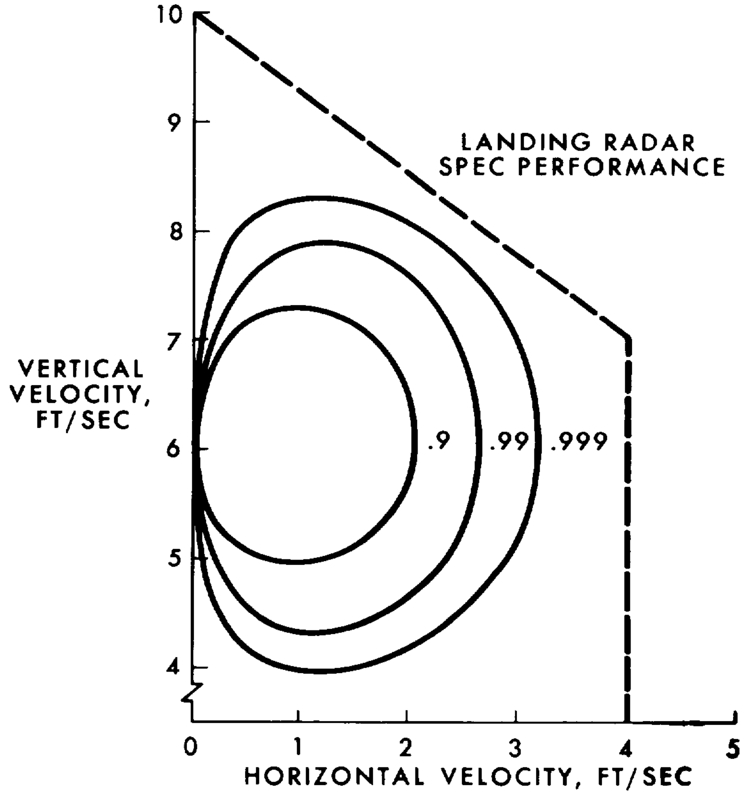


FIGURE 56

NASA-S-65-9330

EFFECT OF LANDING RADAR ERRORS ON VELOCITIES IN MANUAL CONTROL OF LANDING

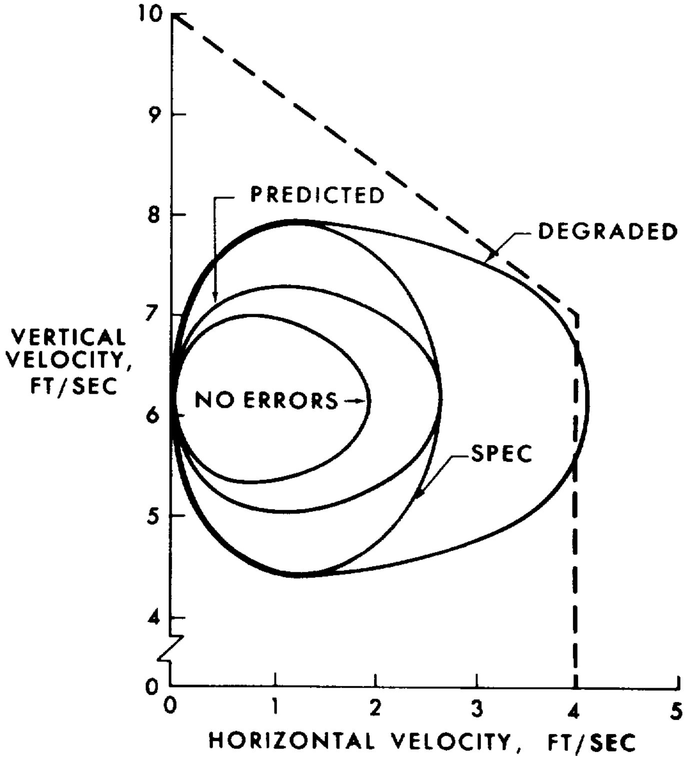


FIGURE 57

NASA-S-65-9329

COMPARISON OF AUTOMATIC AND MANUAL CONTROL OF VELOCITIES LANDING

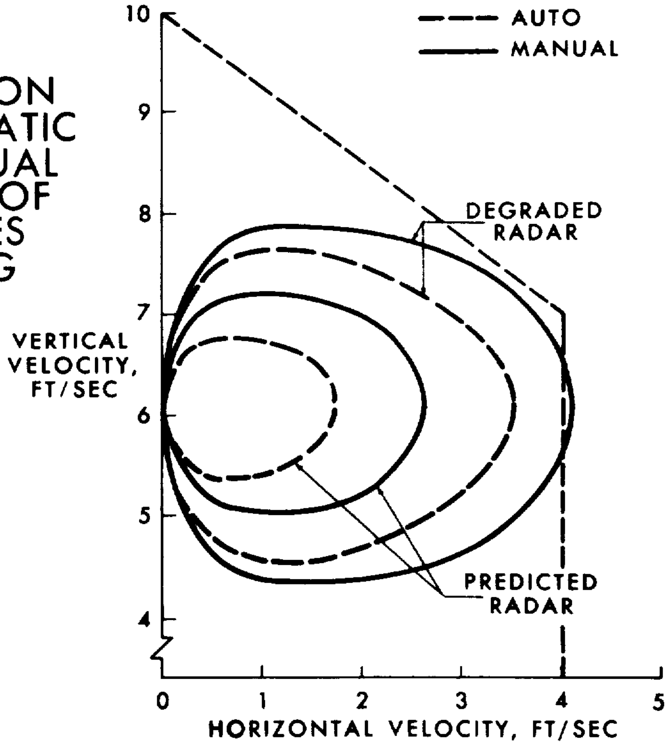
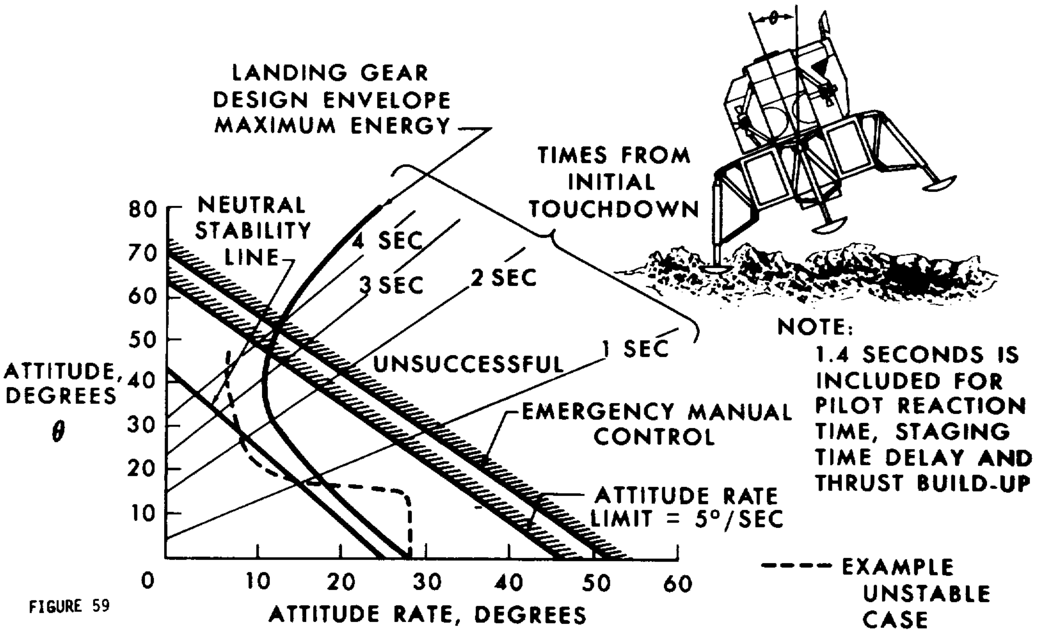


FIGURE 58

NASA-S-66-6462 JUN

BOUNDARY OF ACCEPTABLE ANGLES & ANGULAR RATES FOR TILT-OVER ABORT INITIATE



NOTE:
1.4 SECONDS IS INCLUDED FOR PILOT REACTION TIME, STAGING TIME DELAY AND THRUST BUILD-UP

FIGURE 59

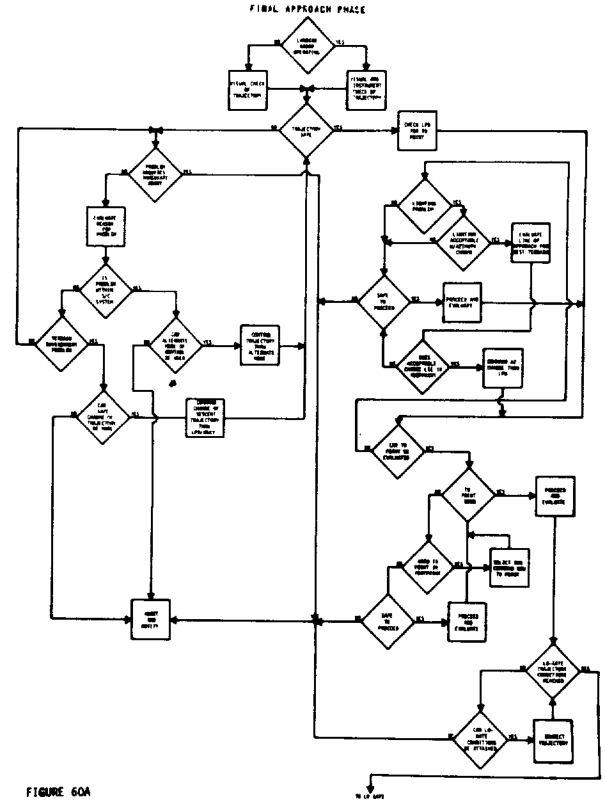

FINAL APPROACH PHASE
VISUAL CHECK OF TRAJECTORY
TRAJECTORY SAFE
YES
NO
CHECK LPD FOR TD POINT
SAFE TO PROCEED
PROCEED AND EVALUATE

FIGURE 60A

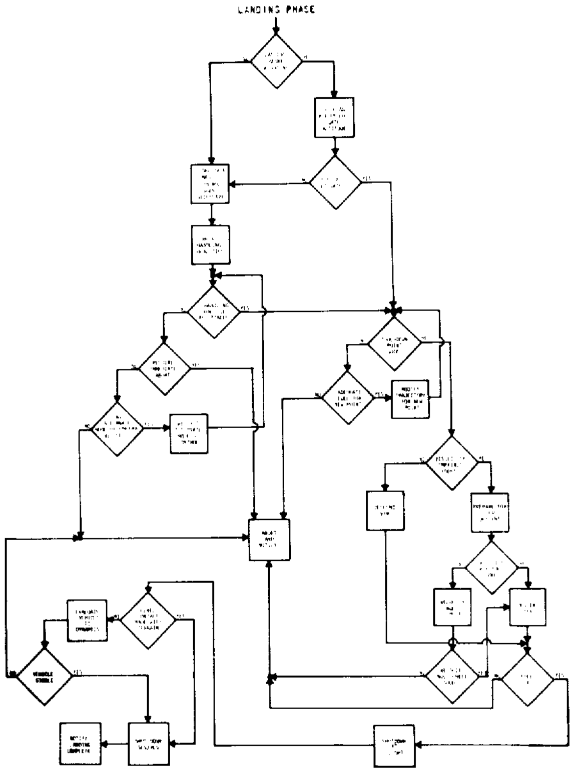

LANDING PHASE
VEHICLE STABLE
YES
NO

FIGURE 60B

APOLLO LUNAR SURFACE SCIENCE PROGRAM

By

Robert E. Vale

APOLLO LUNAR SURFACE SCIENCE PROGRAM

The fundamental objectives and procedures for a long range program of lunar exploration have been extensively discussed and variously stated. A more recent version was derived during a meeting in the summer of 1965 at Woods Hole, Massachusetts and sponsored by the National Academy of Sciences, Space Sciences Board. The gross objectives as expressed on that occasion are grouped in three basic categories as shown by figure 1. Additionally, a series of specific questions were formulated as a more detailed elucidation; i.e. What is the composition of surface materials and how does it vary? What is the tectonic pattern and distribution of tectonic activity? What are the processes of erosion, transport and disposition of surface material? What is the present heat flow at the lunar surface and what is the source of this heat? Is the Moon seismically active and is their active vocanism? Does the Moon have an internally produced magnetic field?

An in-depth understanding of these and the many companion questions will obviously require an extensive program of lunar exploration. However, a very substantial and initial contribution to this understanding will be derived from the early manned lunar landings. It is the intent here to depict a program of lunar surface activities for these early missions which will insure exploiting the full potential of this unprecedented opportunity. These activities are categorized as shown by figure 2 and each will be expanded into considerable detail during the subsequent discussion.

Recognizing that the early missions will be dominated by operational considerations and that astronaut safety is always paramount, there are certain basic criteria that originate in mission definition and spacecraft design that must be honored. Some of the more noteworthy are shown by figure 3. The weight allocation and stowage provisions are more clearly depicted by figure 4. The stowage of equipment is distributed between the ascent and descent stage of the LM. As noted, the maximum weight during the outbound leg of the mission is 250 pounds, which is broken down to 210 pounds in the descent stage and 40 pounds in the ascent stage. This is to minimize the weight of the latter in the event of a mission abort. Clearly the return capability of 80 pounds is totally accommodated by the ascent stage. The scientific equipment bay in the descent stage is in the left rear quadrant of the vehicle. The readily accessible structural fasteners and simple removal procedures insure that the equipment can be retrieved from the vehicle under any condition of slope, gear compression or pad impression that does not cause toppling of the vehicle.

The Extra-vehicular Mobility Unit (EMU) performance is shown here in an essentially parametric manner. This is because the performance variables for the suit and the associated requirements for the equipment do not lend themselves to a quantitative analysis and comparison. It is essential however to recognize qualitatively, the very strong influence which the EMU exerts on the programmed activities and equipment design and to insure this compatibility by a continuing test and simulation program. One exception to the above, which can be semi-quantitatively treated is safe separation distance from the LM. This is a first order consideration and reflects on the feasibility of the entire lunar surface program. The capability in this area is shown by figure 5. It was referred to as semi-quantitative because of the "estimated" values used in its derivation. The EMU is separately discussed in detail, but fundamentally it consists of a pressure garmet, Portable Life Support System (PLSS) and Thermal/Meteoroid Garment (TMG). The primary oxygen supply in the PLSS is augmented by an emergency supply if required. The oxygen supply for breathing and, in an emergency for ventilation by using an open loop blowdown scheme, is the more time sensitive, and is therefore the gauge for defining maximum separation distance. The emergency supply is good for five minutes. The oxygen remaining in the primary supply can be utilized in an emergency mode, therefore, the total available is the emergency bottle plus that remaining in the primary. The latter is continually decreasing during normal operation. The maximum distance that the astronaut can safely separate from the LM is twenty-six minutes at a walking rate conservatively estimated at 150 feet per minute. The plot of figure 5 then is simply a locus of points which at no time are more than the product of the oxygen supply time remaining and 150 fpm walking rate.

Figure 6 is a proposed lunar stay profile for an early mission. The derivation of it will be separately treated and will be shown to be strongly influenced, if not defined by the crew work/rest cycles and the resultant interval between sleep periods. The significance here is the allocation of two separate three-hour extra-vehicular excursions for lunar surface exploration. The excursions are accomplished by both crewmen simultaneously giving a total outside time of 12 manhours.

A first approximation of the scheduled activities for the astronauts during the first excursion is shown in time-line fashion by figure 7. To insure against a contingency which might pre-empt execution of the second excursion, the high priority activities of sample collection and lunar surface experiments package deployment are completed during the first interval. However, beginning with egress of the first crewman, the initial task is to complete a general visual and photographic survey from the forward platform. Time is reserved for the completion of operational tasks such as vehicle walk around inspection, retrieval and erection of the S-Band antenna, etc.

It should be noted here that visual observation is inherent in the situation and direct benefits and data yield are derived as a facet of essentially all of the crew's activities. Inasmuch as there is no mission time expended directly for completion of this activity, the reason for including it here is to recognize it as one of the principal sources of data. The significant implication of effective observation is in pre-flight training of extensive field training and classroom instruction under the auspices of specialists in geology. Mission results will be realized starting with the initial visual contact while still in-flight and extend through the operational and scientific activities of the lunar stay until ascent stage departure. It thus defies discussion as an entity, but this is not to minimize its contribution or emphasis. (figure 8)

Sample collection and return is identified as singularly most important as a scientific objective. In this regard then it is completed first and in the most expeditious manner. The sample return containers which had been stowed in the ascent stage were transferred to the lunar surface during crew egress.

The balance of the geological equipment to aid sample collection is stowed in the descent stage equipment bay as shown earlier and in more detail by figure 9. The right hand package contains the additional tools depicted by figure 10, and can be retrieved independent of the LSEP.

Although each of the scientific activities can be accomplished by a single crewman, crew safety and efficiency are enhanced by partnership and mutual assistance. Figure 11 depicts this arrangement and the carry mode for the equipment. The tool carrier is a three-legged configuration which yields leg relief for ease of carrying and also is collapsible for ease of translunar stowage. The separate tools are individually accommodated on the carrier to improve their utility and logistics of the situation in general during sample collection. The Sample Return Containers (SRC) are carried suitcase fashion. In contrast to what is shown in figure 11, only one SRC is used on the first traverse. The objective is to collect as many samples as possible as expeditiously as possible, without particular emphasis on complete documentation. The purpose is to insure that in the event of an early mission termination an ample supply of samples are available for return. The design of the containers is not firm but it is expected that the individual compartment sizes will be adjustable by movable (or removable) dividers to accommodate a variety of sample sizes. After filling of one container and collection of enough samples in individual bags to fill the second, the samples are returned to the vicinity of the LM where the filled container, after weighing, is heat sealed using LM electrical power. Both sets of samples are left near the ingress ladder for ready retrieval should circumstances dictate.

In the selection of the route for the traverse, zones of contamination resulting from descent engine operation will be avoided. Figure 12 depicts a preliminary estimate of contamination profiles for nitric oxide. There are many other products of combustion with varying profiles (distribution and quantities) in addition to fuel purging effects, LM cabin venting, etc. All of the potential contaiminents will need be assessed and a composite mapping prepared prior to mission conduct. The significance of the figure here is to show qualitatively the influence of contamination zones as a parameter and to identify the need for a more quantitative treatment of the problem.

The second phase of the first excursion is devoted to deployment and emplacement of the LSEP as shown on figure 13. The package is stowed in the same equipment bay of the descent stage as shown by figure 14 and a more detailed configuration on figure 15. The experiments and central station are in the package to the left and the integrated power unit to the rear of the package on the right. Not shown in either view is the fuel cask for the Radioisotope Thermoelectric Generator (RTG). The fuel cask is stowed in a special container external to and thermally shielded from the basic vehicle. This implies then the reason for external stowage; thermal isolation. The fuel cask is at approximately 1300°F. A special mechanism for tilting for ease of extraction and special tool for handling is required for transfer of the fuel cask from stowage to the RTG.

The individual experiments being developed for LSEP application are shown on figure 16. For planning purposes, experiment groupings have been referred to as Array "A" and Array "B". Further, the experiments have been classified as primary or back-up. It is planned that the primaries for a given array will in fact constitute the experiments complement for a given package. The back-ups, however, are candidates for substitution very late in the preparation of flight packages.

The equipment to be deployed can be transported to the selected emplacement site by either of two modes; in suitcase fashion or via barbell, as shown in figure 17. While the method to be employed will be firmly established by testing and simulation, the barbell mode is attractive because of the improvement in walking stability.

The deployed LSEP is shown schematically by figure 18. It is noted the general area selected is approximately 300 feet removed from the LM. This is to avoid the influences of ascent stage engine operation. This distance should avoid the influences of dust and exhaust gas flow patterns. The time required to emplace, erect, activate and align the central station is essentially equivalent to the time required to deploy and emplace the individual experiment sensors. Inasmuch as they are also mutually independent, it results in a very effective partnership activity.

The integrated power unit is shown by figure 19. It should be clarified that while the surface equilibrium temperatures for the radiator fins is approximately 600°F, this steady state situation is not attained immediately. It is a relatively slow process requiring some 30 minutes to occur, which gives time for transport and emplacement without producing an excessive crew hazard. Caution should be exercised but the problem is not unmanageable. Also the separation distance minimizes the electromagnetic interference with the central station electronics, in addition to giving unobstructed radiator "look" angles for thermal efficiency.

The central station is shown by figure 20. The apparent upper level of the base is a solar shield for passive thermal control of the electronics equipment which is in the lower portion. The shield is collapsed during stowage and is erected during emplacement. Electrical heaters for all of the temperature sensitive elements are provided for the lunar night condition. The helical antenna is installed atop a mast to enhance crewman visual access to the alignment optics. Using an ephemeris table, and having a general knowledge of the time and landing position of the LM on the Moon, the astronaut adjusts the optics to obtain a proper offset so that the antenna will be pointed to the center of the Earth's apparent motion. The crewman then centers the Earth in the sight. Since alignment is critical, it was necessary to insure a firm footing for the package initially, to avoid the influences of transient forces disturbing the alignment during the operational life of the system. Also, it is essential that the crewmen not disturb the package after alignment.

In connection with the central station which is the post-launch contact with the package, the Manned Space Flight Network interfaces are shown by figure 21. A data, or down link frequency has been requested for each of three packages to permit their simultaneous operation withoutinterference. The up-link or command frequency is common to all three packages. Appropriate package response is accomplished by signal coding in the transmitter and receivers. An additional point is that the assigned frequencies for the data channels are vacated by a programmed power turn off at the end of the useful life of the package. This turn off can be reprogrammed by Earth command, but in the event of a command link failure the turn off will occur automatically; so turn off must be inhibited if the useful life is to be extended beyond one year.

The magnetometer experiment is shown by figure 22. The sensor heads are on booms in an ortogonal axes arrangement. A bimetallic flipping motor in the base of the instrument permits sensor reorientation and boom rotation which allows a determination to be made of the magnetic field gradients at the magnetometer site. This measurement will reveal any local field anomolies. This survey might be made manually by the astronaut which would simplify the instrument but severely complicate the operational situation, i.e. after each rotation the crewman would have to vacate the area since his equipment is not magnetically clean. It would also define an MSFN and ground readout dependence for checkout assistance prior to completing the installation.

Figure 23 shows the suprathermal ion detector experiment. Installation requirements include leveling to $\pm 5^\circ$ of vertical using a bulls-eye level and aligning the wide entrance aperture to the ecliptic plane. The entrance apertures to the three individual detectors are protected from particle contaimination during deployment and LM launch by a dust cover that is removed by ground command.

The passive seismometer, shown by figure 24, contains two parts in the same package. First a 10-15 second period, three-axis orthogonal seismometer is employed to monitor long-period low-frequency energy. Second, a short period high frequency single axis instrument is included for this energy regime. It is essential during installation to insure a firm couple with the lunar surface since the effectivity of the instrument is directly dependent on this energy transfer. A site survey will thus be required of the astronaut.

The solar wind experiment shown by figure 25 is similar in its installation requirements to that of the suprathermal ion detector. A dust cover removed by Earth command after LM launch is also included.

Subsequent to completion of the LSEP installation of the crewmen return to the vicinity of the LM since the allowable excursion time is essentially used up. Figure 26 shows the relationship between the safe separation envelope and the activities scheduled for inclusion in the first excursion. After return to the vehicle, the first crewman starts the cabin ingress cycle pausing on the forward platform to receive the filled sample containers from the second crewman. This container is returned to the cabin as a safeguard against not being able to execute the second excursion.

It is noted that figure 26 is applicable to the first crewman and included a series of operational tasks preliminary to egress by the second crewman. Since sample collection and LSEP deployment was a partnership activity, the equivalent relationship for the second crewman would in effect be displaced to the left. There will therefore be some 25 minutes remaining that can be used but must be in the vicintiy of the vehicle. It is expected that this will be employed for the completion of additional operational tasks such as measurement of landing gear strut compression, footpad impression depth, pad skid marks, etc. Fulfilling this, the second crewman will ingress the LM cabin and thus complete the first excursion.

A final note should be made about the contingency of only a single crewman descending to the surface. Deployment and emplacement of the LSEP can be accomplished by a single crewman and would require about one hour and 25 minutes to complete. The time devoted to sample collection in the first part of the excursion should then be adjusted accordingly. This would probably mean that he could not get so far from the vehicle and that he could complete little if any documentation. Sample collection and LSEP deployment can however be completed by a single crewman in the first excursion.

The activities associated with the second excursion are shown by figure 27 and it is clearly devoted in its entirety to field geology. Field geology is defined herein according to figure 28. The traverse associated with this excursion is an amplification of the sample collection phase of the first excursion. The second SRC is utilized and the samples collected are very thoroughly documented by photography and verbal description.

Figure 29 is representative of this partnership activity. Extensive attention is directed also toward interests other than sample collection such as local and far field mapping of major geological and topographical features. Figure 30 depicts the safe separation versus time relationship for the second excursion. The significance of it is simply to show that the crewmen should go out the programmed or limiting distance and work their way back to the vehicle. Since a fairly comprehensive procedure of documenting samples and mapping has been indicated to be in effect, the crewmen return will essentially be in phases. They will return well within the safe distance, remain for a prescribed period of time and then move on to another closer-in location. As will be emphasized later, it is not the intent to overplan the mission and thus pre-empt the selective judgement of the astronaut in fulfilling this activity. Thus it should not be inferred that separation or return profile such as this is mandatory; rather it is a limiting case. If in the opinion of the crew, on examination during the outbound leg of the traverse, it would be more effective to spend more time exploring and documenting a feature nearer the vehicle, they should and will be expected to exercise this prerogative. Alternatives withing the boundary condition is at crew discretion.

Inasmuch as this entire area of formulating and validating a program of lunar surface activities defies a hard quantitative analysis, particular emphasis must and will be placed on an exhaustive simulation program. The non-quantifiable facets of the EMU and crew performance and capabilities, coupled with the unknown but obviously hostile envirnnmental conditions on the lunar surface, reduce validation of the design and programmed activities to a "very best" approximation. The principal objectives of such a simulation program are thus as shown on figure 31. To effecively implement such a program the procedure shown schematically on figure 32 will be employed.

The requirements of the program will be collectively established by representatives of the directly effected organizations. In the case of the scientific community a singular representative will be appointed to synthesize the objectives addressing a particular (or grouping) of disciplines. This working group will document the total of the requirements to be implemented jointly by MSC and the U.S.G.S. To insure a minimum duplication of effort and a maximum of data interchange, the implementation will be under the central control of an MSC Working Group. The results will be merged, considered in context, analyzed and distributed for use in design iterations, formulation of crew training requirements and in detailed mission planning. The approach is obviously not unique and therefore its expected effectiveness is not revealed by an examination of this flow diagram. It is more apparent when it is emphasized that this is a continuing process and requirements are formulated, the test conducted and the results distributed in frequent iterations. The necessity for such a flexible and continuing program is further exemplified by figures 33 through 36 which are representative of the different categories of simulations required. No single facility or test type will permit total mission simulation. The 1/6 G counterbalance rig will permit an approximation of the gravity environment for an extended period of time, however, it is not without the obvious constraints of inertial anomolies and encumbrances of the rig. The KC-135 is a reasonable approximation of the actual free body effects of 1/6 G but the time variable is limited to 35 seconds or less; thus continuity is not possible. An integration of these separate results into a singular "net effect" and then attempting an extrapolation into the real situation gives an indication of extremely close co-ordination. An out-of-context data yield for any given test will likely have limited validity and at best should be used with discretion.

The magnitude of the problem can begin to be scoped by compiling a matrix such as that shown by figure 37 which shows tasks by category versus some selected parameters that influence the simulation. Also when assessing which tests have been completed and those that need be initiated, it should be remembered that the tasks by categories have significant second and third tier divisions that are essentially entities. Thus where a check is shown, it perhaps should have been a fractional check since only a portion of that category may have been tested. The final significant point is that virtually all testing to date has utilized conceptual equipment only and an early model pressure suit. A series of tests employing more flight-like equipment in the form of mockups, prototypes, and of course final configuration hardware is required. Relative to the suit configuration, it should be pointed out that while to date only early models have been used, this should inject a degree of conservatism in the completed tests since recent demonstrations have shown a marked improvement in mobility.

The very active and comprehensive simulation program outlined above and already initiated should in fact further define and finally substantiate a lunar surface program of activities such as has been discussed in this entire section.

A final point which should be made relates to program flexibility as shown on figure 38. The LSEP basic configuration will permit a substitution of a back-up experiment for a primary very late in the preparation of flight hardware. Also simulation and training programs will have addressed these experiments, thus minimizing the effects of this late substitution. A most important consideration is in the level of detail that the final mission plan reflects. Crew judgement and selectivity and the capacity for exercising these will undoubtedly be one of the most valuable mission assets in realizing the scientific objectives. The mission profile will therefore not be over planned and crew discretion will prevail.

The successful completion of a science program as outlined herein will make a very substantial contribution to the definition and understanding of the present status of the lunar surface and interior and the evoluionary sequence of events by which the Moon arrived at its present configuration.

Questions and Answers

LUNAR SURFACE EXPERIMENTS

Speaker: Robert E. Vale

1. Dr. Mueller - Would like more detail on sample collection.

   ACTION - Mr. Vale

2. Dr. Mueller - How does the crew sleep in the LM?

   ANSWER - Will be covered by Mr. Loftus during the discussion of crew tasks.

3. Will photos be taken of each sample?

   ANSWER - Yes.

4. Dr. Rees - Why will both crewmen be sleeping at the same time?

   ANSWER - Gemini experience has shown that the activities of the non-sleeping crewman prevent sound sleep and ground monitoring has proven to be adequate. Therefore, it has worked better for both crewmen to sleep at the same time.

5. Why is there no TV shown?

   ANSWER - Block II TV is not considered a part of the scientific experiments.

6. Mr. Holmes - How much real time communication is there with scientists on earth?

   ANSWER - Information will be fed back to the MCC in real time.

7. Mr. Davidson - Was fixed TV looking at earth considered as an experiment?

   ANSWER - No.

8. At the sun angles considered during the lunar stay, what lunar surface temperatures are expected?

   ANSWER - Approximately + 200°F.

9. For the scientific equipment, what is the apportionment of weights between the ascent and descent stages?

   ANSWER - 250 pounds total of which 210 pounds will be in the descent stage and 40 pounds in the ascent stage.

10. Are we bringing the TV camera back?

   ANSWER - No.

11. Is there any scientific equipment in the CSM?

   ANSWER - No, not for the early lunar landing missions.

NASA-S-66-5194 JUN

SCIENTIFIC OBJECTIVES BY CATEGORY

- INVESTIGATE THE STRUCTURE AND PROCESSES OF THE LUNAR INTERIOR
- DETERMINE THE COMPOSITION AND STRUCTURE OF THE SURFACE OF THE MOON AND THE PROCESSES MODIFYING THE SURFACE
- ESTABLISH THE HISTORY OR EVOLUTIONARY SEQUENCE OF EVENTS BY WHICH THE MOON HAS ARRIVED AT ITS PRESENT CONFIGURATION

NOTE: FROM A MEETING SPONSORED BY THE NATIONAL ACADEMY OF SCIENCES, SPACE SCIENCES BOARD, AT WOODS HOLE, MASSACHUSETTS, IN THE SUMMER OF 1965

Figure No. 1

NASA-S-66-5197 JUN

PRIORITY ACTIVITIES

- OBSERVATIONS
  PROVIDE QUALITATIVE DESCRIPTION OF LUNAR SURFACE FEATURES
- SAMPLE COLLECTION
  TO PERMIT POST-MISSION ANALYSIS ADDRESSING BASIC QUESTIONS IN THE FIELDS OF GEOCHEMISTRY, PETROLOGY, GEOLOGY, AND BIOSCIENCE
- DEPLOYMENT OF LUNAR SURFACE EXPERIMENTS PACKAGE
  TO OBTAIN CONTINUED MEASUREMENT OF GEOPHYSICAL PARAMETERS FOR ONE YEAR AFTER LM DEPARTURE
- FIELD GEOLOGY
  TO OBTAIN INFORMATION ON POSSIBLE GEOLOGIC STRUCTURE AS IT MAY BE REVEALED BY SURFACE FEATURES AND FORMATIONS

Figure No. 2

NASA-S-66-5204 JUN

BASIC CRITERIA FROM SPACECRAFT/MISSION

• WEIGHT ALLOCATION	250/80
• STOWAGE PROVISION	LM STRUCTURE
• EMU PERFORMANCE	OPERATING TIME, METABOLIC LOADS, MOBILITY, DEXTERITY, VISIBILITY, THERMAL CONTROL, COMMUNICATIONS
• SITE SELECTION	OPERATIONAL PRIORITY
• LUNAR SURFACE STAY	DISTRIBUTION, DURATION & NUMBER OF CREWMEN FOR EXTRA-VEHICULAR EXCURSIONS

Figure No. 3

NASA-S-66-5198 JUN

SCIENTIFIC EQUIPMENT STOWAGE

LUNAR MODULE

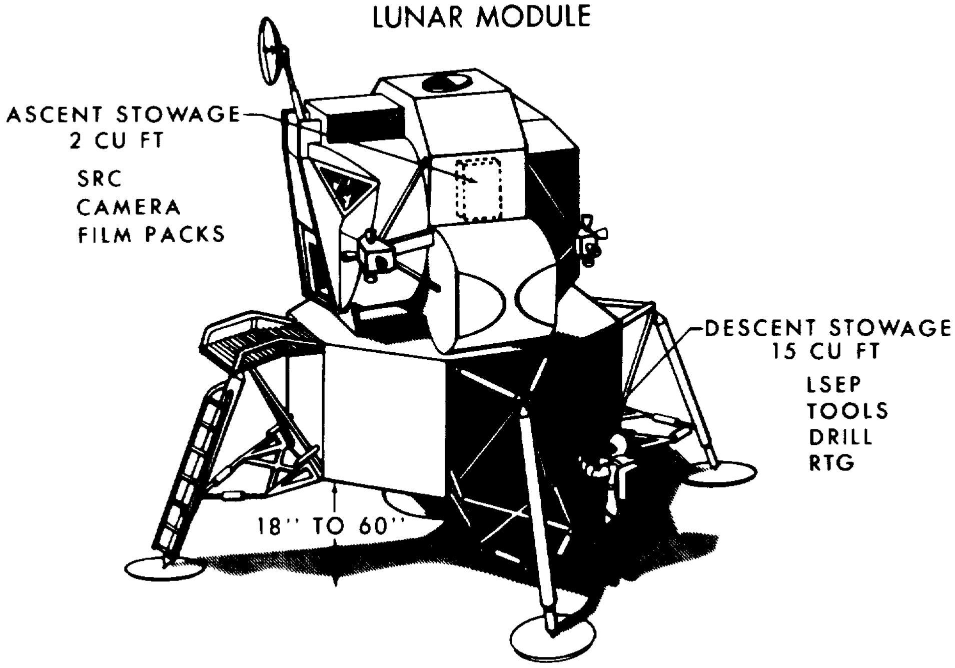

ASCENT STOWAGE
2 CU FT
SRC
CAMERA
FILM PACKS
DESCENT STOWAGE
15 CU FT
LSEP
TOOLS
DRILL
RTG
18" TO 60"

Figure No. 4

NASA-S-66-5187 JAN

SAFE SEPARATION FROM LM

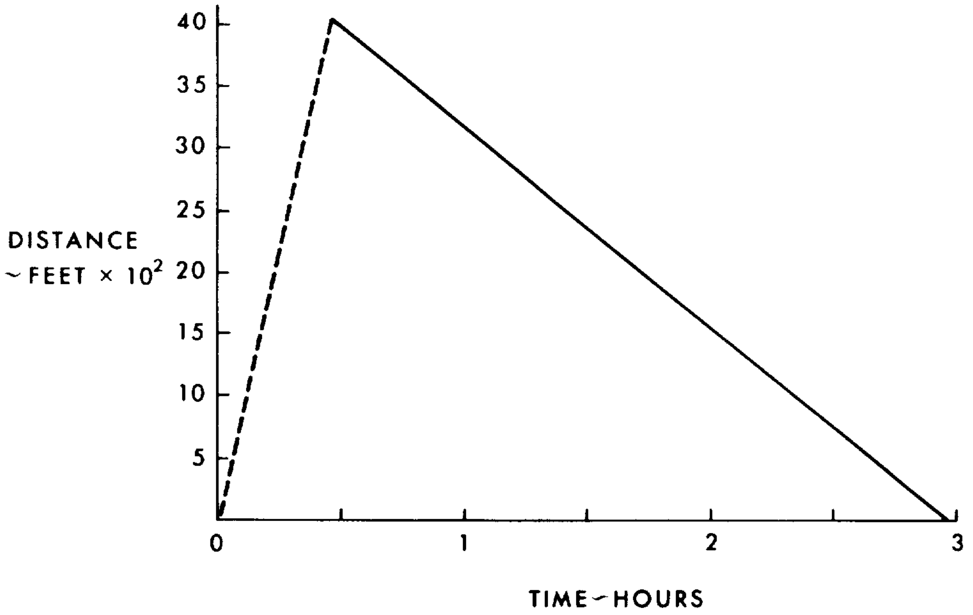


Figure No. 5

NASA-S-66-5166 JUN

PROPOSED LUNAR STAY
(18 HOURS 22 MINUTES)

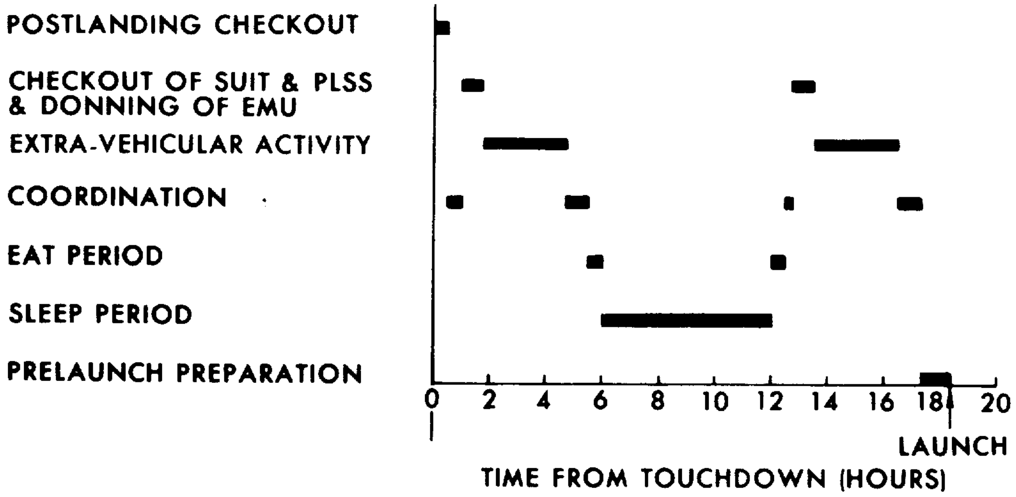


Figure No. 6

NASA-S-66-6519 JUN

FIRST EXCURSION

CREWMAN NO. 1	TIME	CREWMAN NO. 2
EGRESS TO FORWARD PLATFORM GENERAL VISUAL AND PHOTOGRAPHIC SURVEY	0	
DESCEND TO LUNAR SURFACE CONDUCT OPERATIONAL TASKS SUCH AS VEHICLE WALK AROUND INSPECTION	15	
ACCEPT EQUIPMENT TRANSFER	25	TRANSFER EQUIPMENT TO NO. 1
RETRIEVE LGE FROM EQUIPMENT BAY	30	EGRESS TO LUNAR SURFACE AND CONDUCT OPERATIONAL TASKS SUCH AS RETRIEVE AND ERECT S-BAND ANTENNA
SAMPLE COLLECTION TRAVERSE	40	SAMPLE COLLECTION TRAVERSE
PARTNERSHIP ACTIVITY WITH NO. 2 IN AN ESSENTIALLY UNDOCUMENTED SAMPLE COLLECTION		SAME AS NO. 1
RETURN ONE FILLED CONTAINER AND CONTENTS FOR SECOND TO VICINITY OF LM LADDER	95	SAME AS NO. 1

Figure No. 7

NASA-S-66-5195 JUN

OBSERVATIONS

- PRE-FLIGHT TRAINING AND CRITIQUE
  EXTENSIVE FIELD GEOLOGY AND CLASSROOM INSTRUCTION TO INSURE PROFICIENCY IN IDENTIFICATION OF SIGNIFICANT FEATURES AND ACCURATE REPORTING OF RESULTS
- COMMUNICATIONS
  TWO-WAY VOICE BETWEEN BOTH CREW MEN AND MSFN TO RECORD REALTIME IMPRESSIONS
- RECORDS
  MAINTENANCE OF CREW LOGS TO INSURE COMPLETE RECORDING OF DETAIL IMPRESSIONS AND OBSERVATIONS
- AUGMENT LIGHTING
  UTILIZATION OF PORTABLE LIGHT TO IMPROVE VISUAL TASK IN SHADOWED AREAS
- TIME CONSUMED
  OBSERVATION IS INHERENT IN THE SET OF CIRCUMSTANCES. NO TIME CHARGED DIRECTLY

Figure No. 8

NASA-S-66-5172 JUN

ALSEP COMPARTMENTS 1 & 2 STOWED - ARRAY A

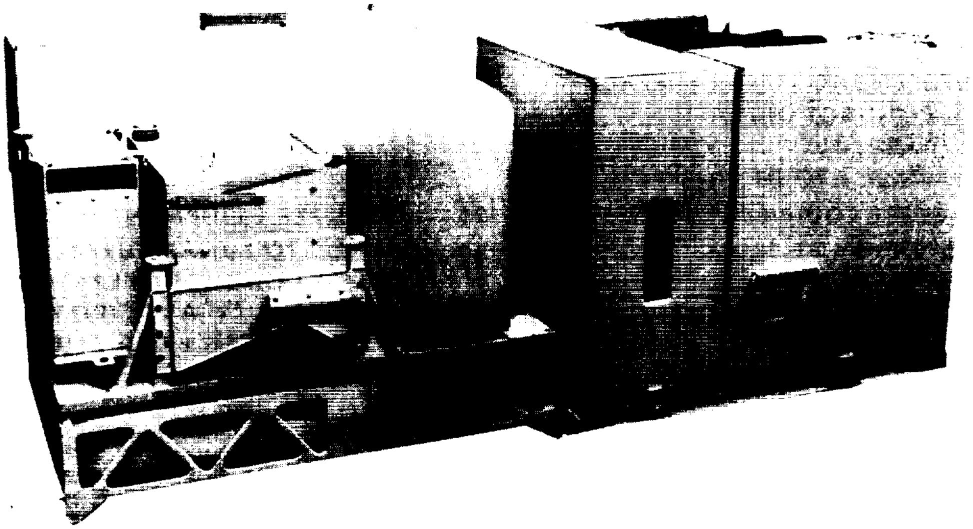


Figure No. 9

NASA-S-66 6804 JUN

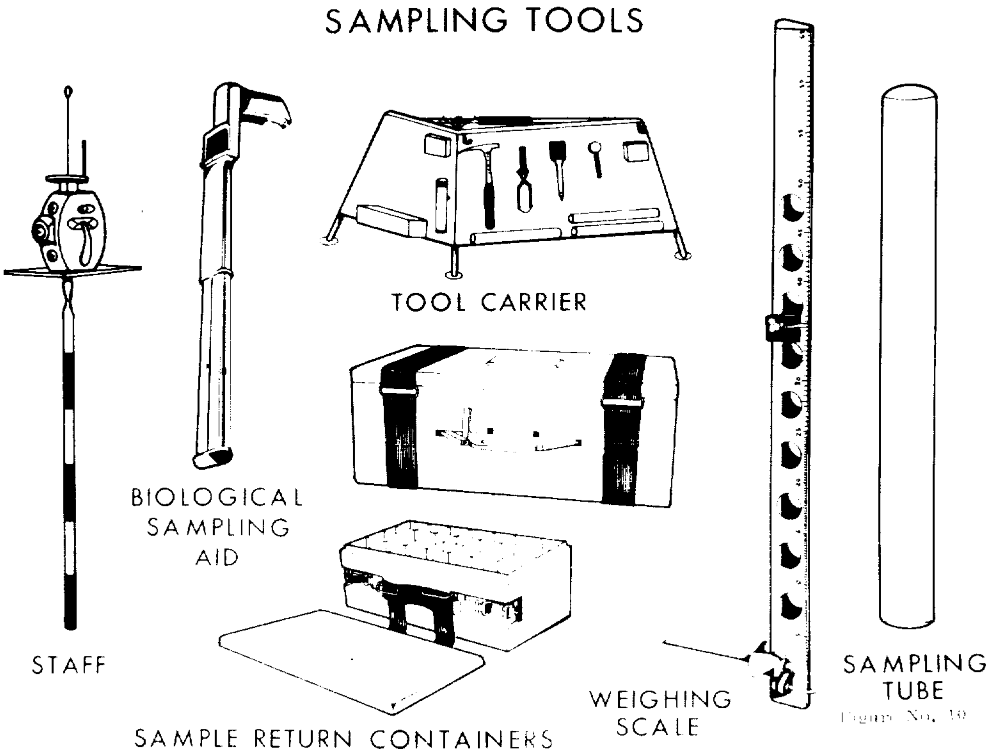

SAMPLING TOOLS
TOOL CARRIER
BIOLOGICAL SAMPLING AID
STAFF
SAMPLE RETURN CONTAINERS
WEIGHING SCALE
SAMPLING TUBE

Figure No. 10

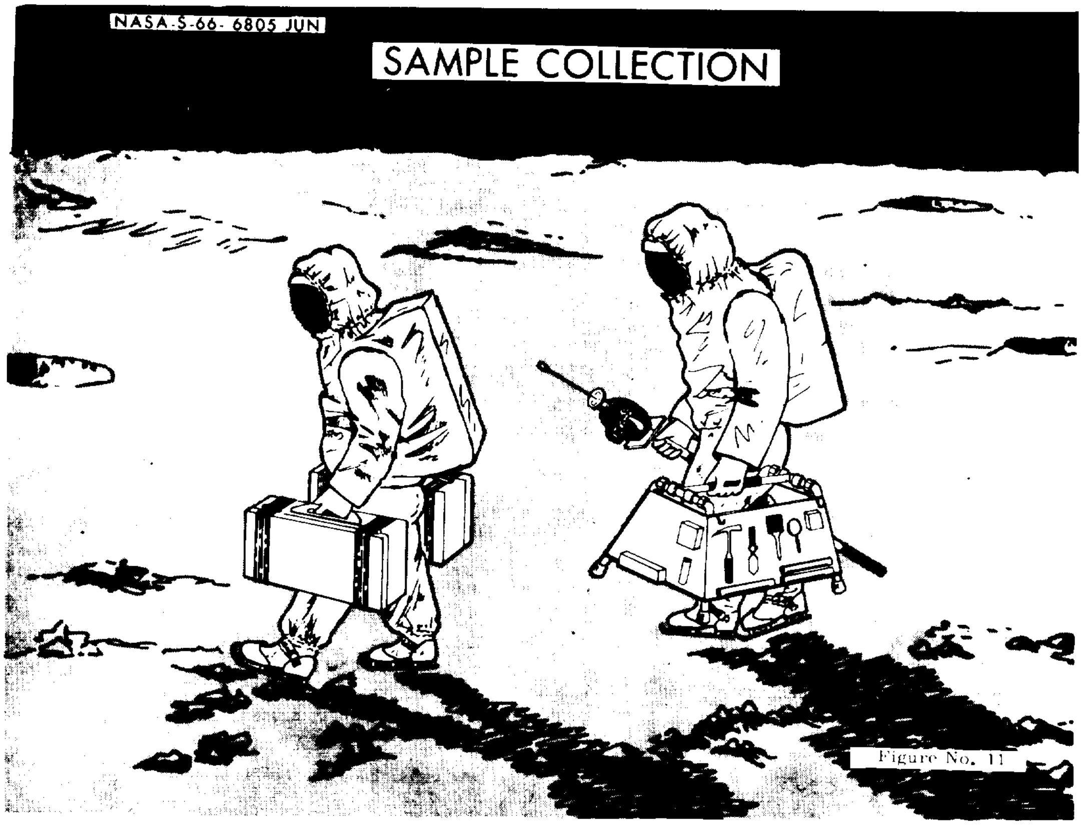

NASA-S-66-6805 JUN
SAMPLE COLLECTION

Figure No. 11

NASA-S-66-6088 JUN

CONTOURS OF EQUAL ADSORPTION OF NO IN UNITS OF $\mu g/cm^2$

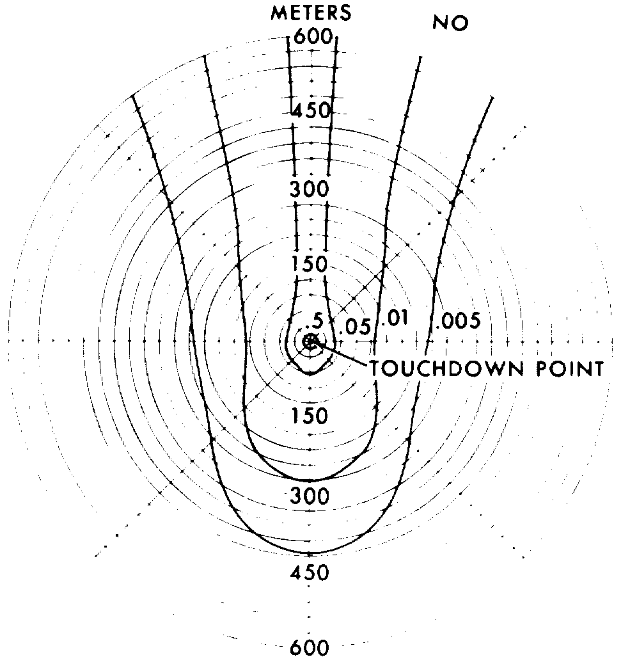


Figure No. 12

NASA-S-66-6000 JUL 5

FIRST EXCURSION (CONT)

CREWMAN NO. 1	TIME	CREWMAN NO. 2
UNLOAD LSEP AND ASSEMBLE FOR CARRYING	105	WEIGH AND SEAL ONE CONTAINER AND PREPARE FOR LM LOADING
TRANSPORT TOTAL PACKAGE TO SITE	115	ACCOMPANY NO. 1 TO LSEP SITE
DEPLOY RTG, EMPLACE CENTRAL STATION, ERECT AND ALIGN ANTENNA	125	DEPLOY EXPERIMENTS
	(125)	(MAGNETOMETER)
	(135)	(SUPRO THERMAL ION DETECTOR)
	(145)	(PASSIVE SEISMOMETER)
	(150)	(SOLAR WIND)
COMPLETE LSEP INSTALLATION	155	COMPLETE LSEP INSTALLATION
RETURN TO LM AND ASCEND TO FORWARD PLATFORM TO RECEIVE CONTAINER	155	RETURN TO LM
TRANSFER CONTAINER TO LM CABIN	170	TRANSFER CONTAINER TO NO. 1
INGRESS LM CABIN	175	CONDUCT OPERATIONAL TASKS IN LM VICINITY SUCH AS MEASURE GEAR STROKE, GEAR PAD IMPRESSION, SKID DISTANCE, ETC
COMPLETE EXCURSION	180	
	200	ASCEND LADDER AND INGRESS LM CABIN
	205	COMPLETE EXCURSION

Figure No. 13

NASA-S-66-5198 JUN

SCIENTIFIC EQUIPMENT STOWAGE

LUNAR MODULE

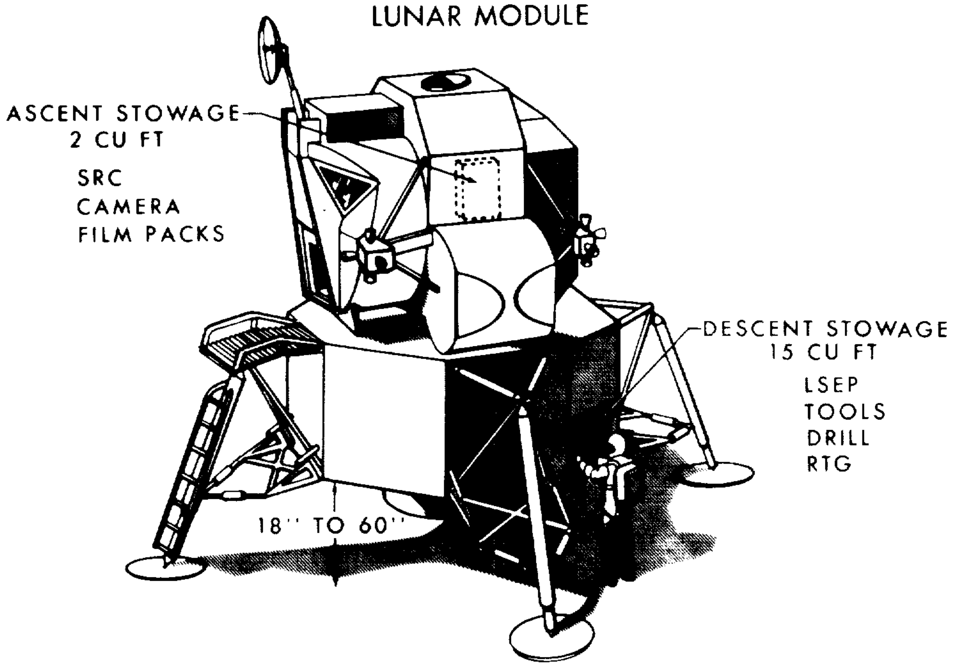


Figure No. 14

NASA-S-66-5172 JUN

ALSEP COMPARTMENTS 1 & 2 STOWED - ARRAY A

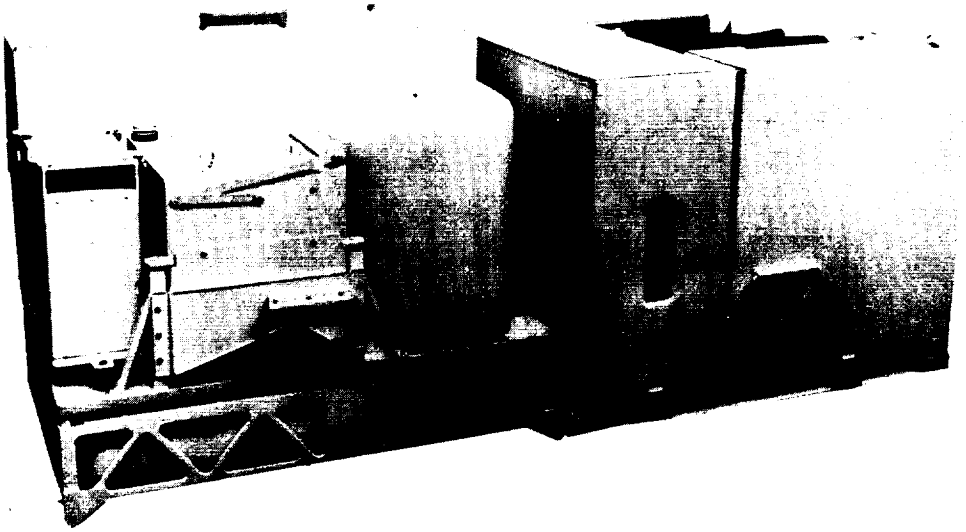


Figure No. 15

NASA-S-66-6587 JUN

ALSEP EXPERIMENTS

NUMBER	EXPERIMENT	INVESTIGATOR	ARRAY A	ARRAY B
S-1001	PASSIVE SEISMIC	(SUTTON)	P	P
S-1004	MAGNETOMETER	(SONNET)	P	B-1
S-1005	SOLAR WIND	(SNYDER)	P	B-2
S-1006	ION DETECTOR	(FREEMAN)	P	P
S-1007	HEAT FLOW	(LANGSETH)	B-1	P
S-1008	ELECTRON/PROTON	(O'BRIEN)	B-2	B-3
S-1003	ACTIVE SEISMIC	(KOVACH)	B-3	P

NOTES:

ARRAY A LSEP 1 AND 2

ARRAY B LSEP 3 AND 4

P PRIMARY

B BACKUP AND PRIORITY

Figure No. 16

NASA-S-66-5203 JUN

BARBELL CARRY MODE

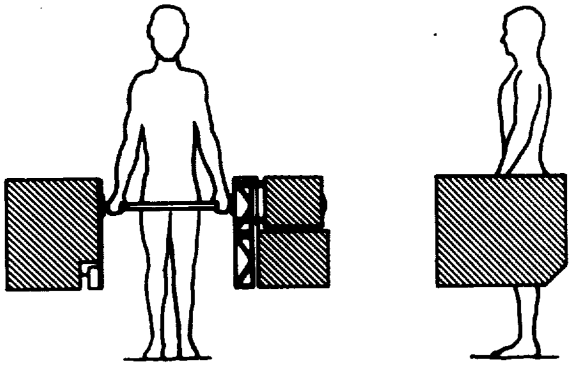
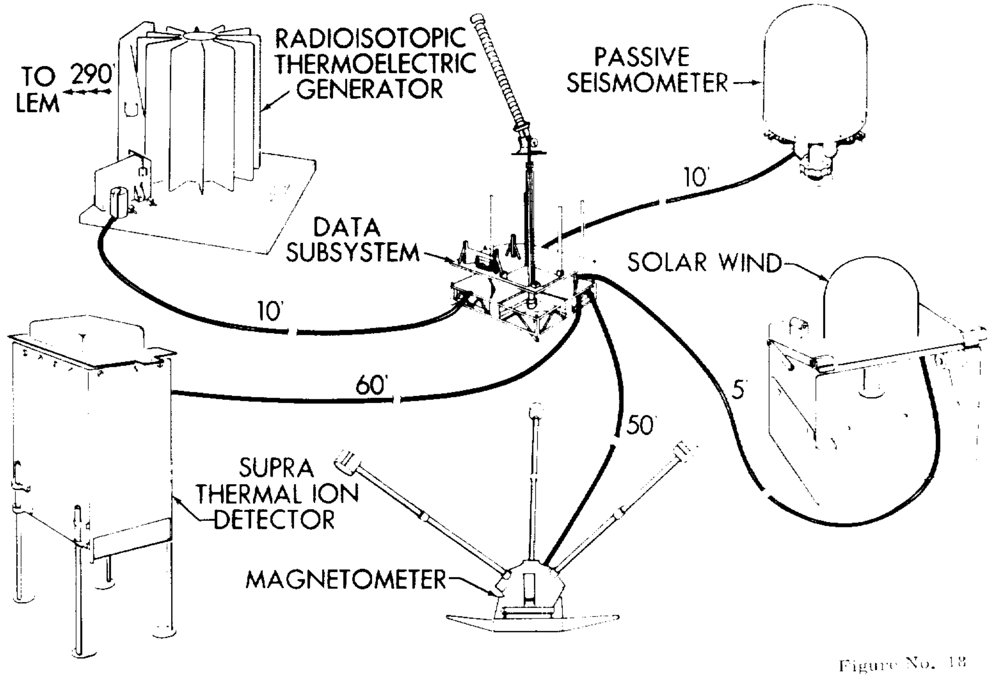


Figure No. 17

NASA-S-66-5213 JUN

EXPERIMENT ARRAY 'A' DEPLOYED

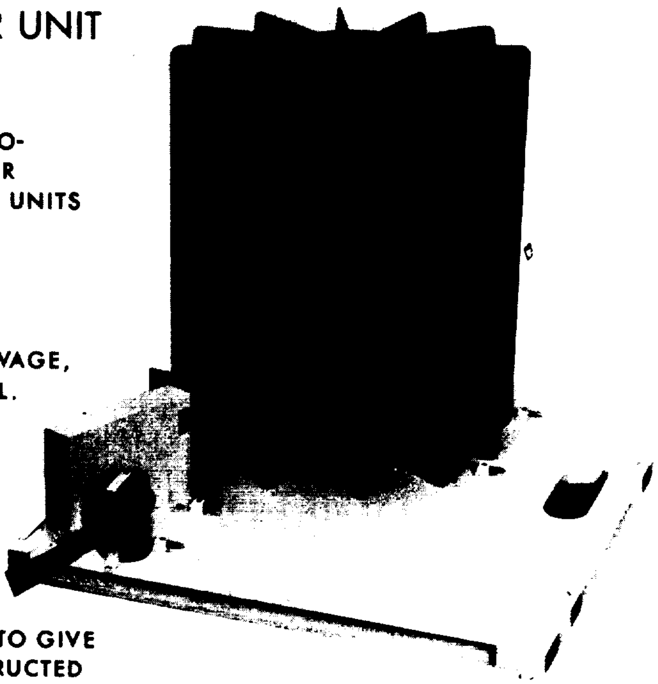


TO LEM
290'
RADIOISOTOPIC THERMOELECTRIC GENERATOR
PASSIVE SEISMOMETER
10'
DATA SUBSYSTEM
SOLAR WIND
10'
60'
5'
50'
SUPRA THERMAL ION DETECTOR
MAGNETOMETER

Figure No. 18

NASA-S-66-5173 JUN 13

INTEGRATED POWER UNIT

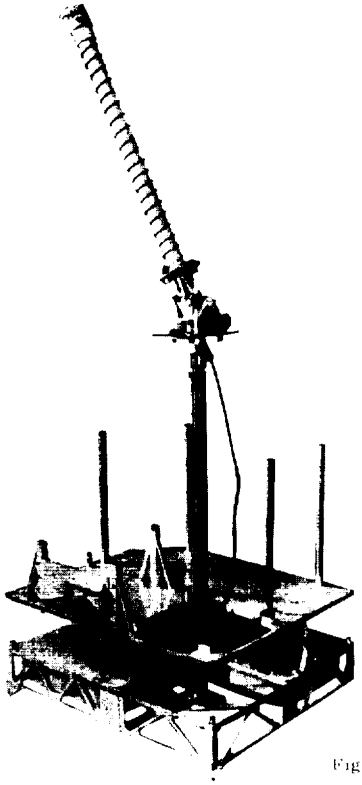


- CONSISTS OF
  - RADIOISOTOPIC THERMOELECTRIC GENERATOR
  - POWER CONDITIONING UNITS
  - POWER CABLE
  - 50 WATTS AT 28 VDC
- UNIQUE TASKS
  - RETRIEVE FUEL CASK FROM SPECIAL STOWAGE, USING SPECIAL TOOL. INSERT IN RTG
- HANDLING CAUTION:
  - SURFACE TEMPERATURE OF RTG IS APPROX 600° F
  - AT LEAST 10 FEET FROM CENTRAL PACKAGE TO GIVE RADIATORS UNOBSTRUCTED LOOK ANGLE

Figure No. 19

NASA-S-66-5170 JUN

CENTRAL PACKAGE



- CONSISTS OF
  - DATA SUBSYSTEM
  - STRUCTURAL/THERMAL SUBSYSTEM
  - ELECTRICAL POWER SUBSYSTEM
- UNIQUE TASKS
  - AZIMUTH ALIGNMENT TO PREVENT DIRECT SUN ON THERMAL RADIATOR
  - ANTENNA ERECTION AND ALIGNMENT FOR MSFN ACQUISITION

Figure No. 20

NASA-S-66-5196 JUN

MSFN INTERFACE OPERATIONS

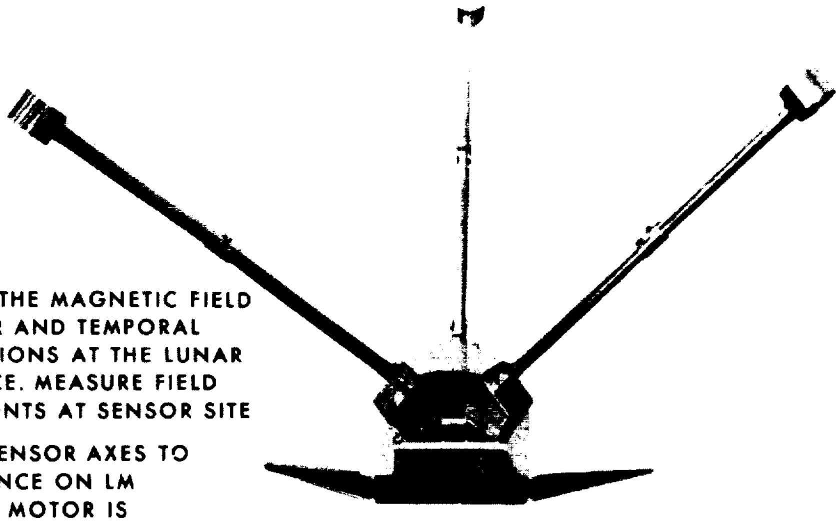
● CHANNEL FREQUENCY REQUESTS	2275.5 MC 2276.5 MC 2278.5 MC 2119.0 MC

- SYSTEM ACTIVATED BY EARTH COMMAND AND MONITORED BY MCC AND MSFN. ALL DATA TRANSMISSIONS EMPLOY 30-FOOT MSFN ANTENNA EXCEPT ACTIVE SEISMIC EXPERIMENT (DATA RATE 10,600 BPS) WHICH NEEDS 85-FOOT ANTENNA
- NORMAL MODE IS FOR MSFN STATIONS TO TAPE RECORD LSEP DATA AND SHIP TO MSC. IF/AS REQUIRED DIRECT DATA LINK TO MSC PERMITS REAL-TIME DISPLAY OF ENGINEERING DATA TO ENHANCE SYSTEM MONITORING AND CORRECTIVE ACTION COMMANDS
- NO REQUIREMENT FOR REAL-TIME DISPLAY OF EXPERIMENTS DATA. TAPE PLAYBACK ADEQUATE

Figure No. 21

NASA-S-66-5175 JUN

MAGNETOMETER



- MEASURE THE MAGNETIC FIELD VECTOR AND TEMPORAL VARIATIONS AT THE LUNAR SURFACE. MEASURE FIELD GRADIENTS AT SENSOR SITE
- ALIGNS SENSOR AXES TO REFERENCE ON LM
- IF FLIPPER MOTOR IS DELETED, CREW WILL HAVE TO CYCLE THE HEADS FOR LOCAL SURVEY
- 50 FEET FROM CENTRAL PACKAGE

Figure No. 22

NASA-S-66-5169 JUN

SUPRA THERMAL ION-DETECTOR

- TO MEASURE THE FLUX, ENERGY, AND VELOCITY OF POSITIVE IONS IN LUNAR IONOSPHERE
- LEVEL ENTRANCE APERATURE TO ±5° OF VERTICAL
- 60 FEET FROM CENTRAL PACKAGE

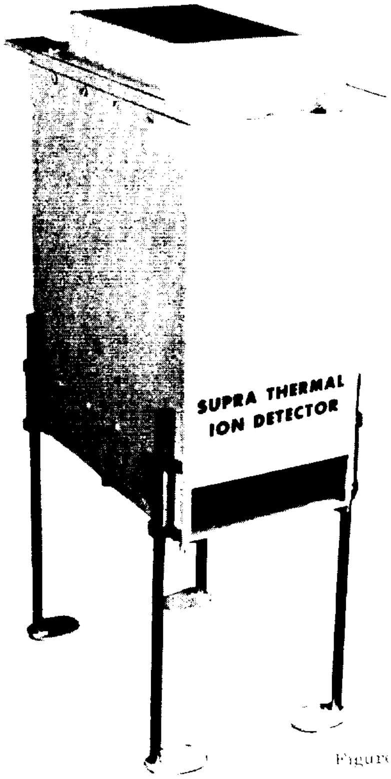


Figure No. 23

NASA-S-66-5174 JUN

PASSIVE SEISMOMETER

- TO DETERMINE EXISTENCE OF SEISMIC ACTIVITY AND TO INTERPRET WAVE FORMS FOR DATA ON INTERNAL PROPERTIES
- ESTABLISH A FIRM FOOTING FOR LEGS AND ADJUST TO ± 10° OF VERTICAL
- 10 FEET FROM CENTRAL PACKAGE

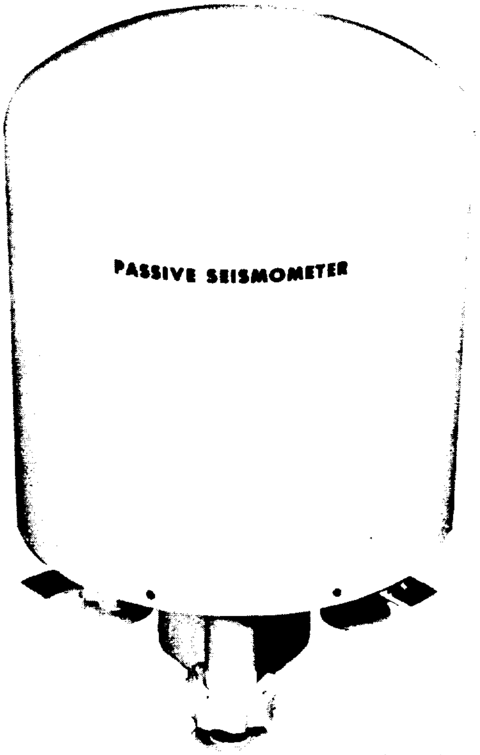


Figure No. 24

NASA S-66-5171 JUN

SOLAR WIND

- TO MEASURE THE SPECTRAL AND DIRECTIONAL CHARACTER OF SOLAR PLASMA AT THE LUNAR SURFACE
- ROUGH LEVEL AND ORIENT RELATIVE TO ECLIPTIC PLANE
- OPPOSITE CENTRAL PACKAGE FROM RTG

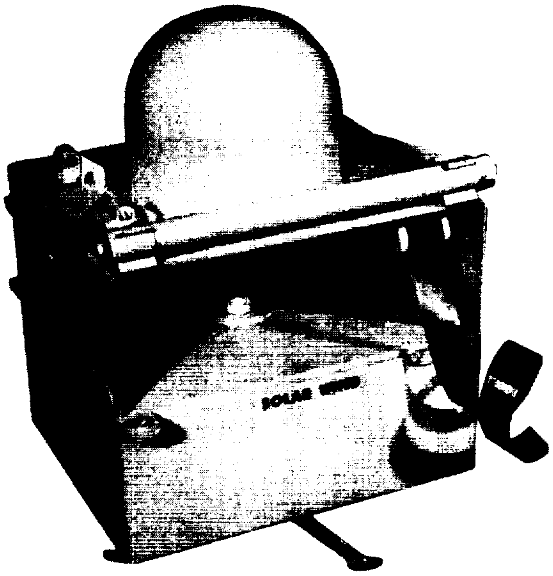


Figure No. 25

SAFE SEPARATION FROM LM
FIRST EXCURSION

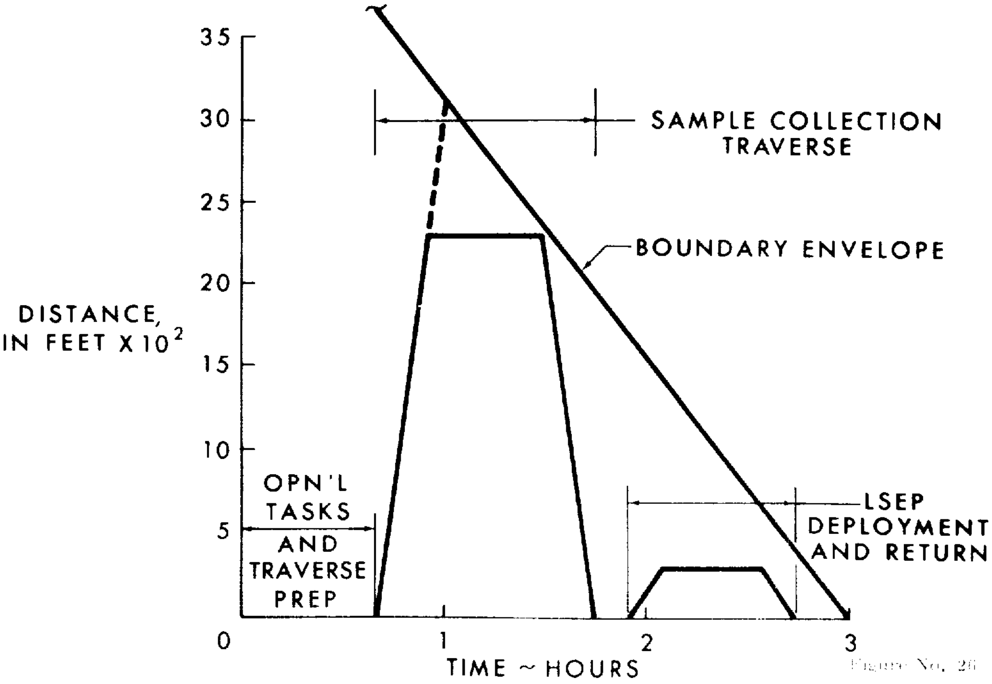


Figure No. 26

NASA-S-66-6508 JUN

SECOND EXCURSION

CREWMAN NO. 1	TIME	CREWMAN NO. 2
EGRESS TO LUNAR SURFACE	0	
RETRIEVE LGE AND PREPARE FOR TRAVERSE	5	EGRESS TO LUNAR SURFACE
CONDUCT FIELD GEOLOGY EXPERIMENT IN PERIMETER NOT TO EXCEED SAFE SEPARATION DISTANCE FROM LM. COMPLETE DOCUMENTATION OF SAMPLE COLLECTION FOR ONE CONTAINER. NEAR AND FAR FIELD PHOTOGRAPHY. MAPPING OF MAJOR FEATURES AND LANDMARKS	10	SAME AS NO. 1
RETURN TO LM, ASCEND TO FORWARD PLATFORM TO RECEIVE CONTAINER	160	RETURN TO LM, WEIGH AND SEAL CONTAINER
TRANSFER CONTAINER TO LM CABIN	170	TRANSFER CONTAINER TO FORWARD PLATFORM
INGRESS LM CABIN	175	ASCEND TO PLATFORM
COMPLETE EXCURSION	180	INGRESS LM CABIN
	185	COMPLETE EXCURSION

Figure No. 27

NASA-S-66-5191 JUN

FIELD GEOLOGY

- SYSTEMATIC EXAMINATION, PHOTOGRAPHING, DESCRIPTION & COLLECTION OF GEOLOGIC UNITS & SAMPLES ALONG A TRANSVERSE
- DETERMINATION OF FIELD RELATIONS, SUCH AS SHAPE, SIZE, RANGE, PATTERNS OF ALIGNMENT OR DISTRIBUTION OF ALL TYPES OF TOPOGRAPHIC FEATURES
- GEOLOGIC SIGNIFICANCE OF SAMPLES DEPENDS IN PART ON FIELD RELATION, THEREFORE FIELD GEOLOGY WILL INCLUDE DOCUMENTATION OF SAMPLE COLLECTION
- DATA YIELD COMES FROM VERBAL COMMUNICATIONS, PHOTOGRAPHIC DOCUMENTATION & SAMPLE RETURN. POSITION INFORMATION PROVIDED BY INCLUDING GNOMEN AND WHEN POSSIBLE, VEHICLE IN FIELD OF VIEW
- HARDWARE REQUIREMENTS ARE THE GEOLOGICAL TOOLS, CAMERA AND GNOMEN

Figure No. 28

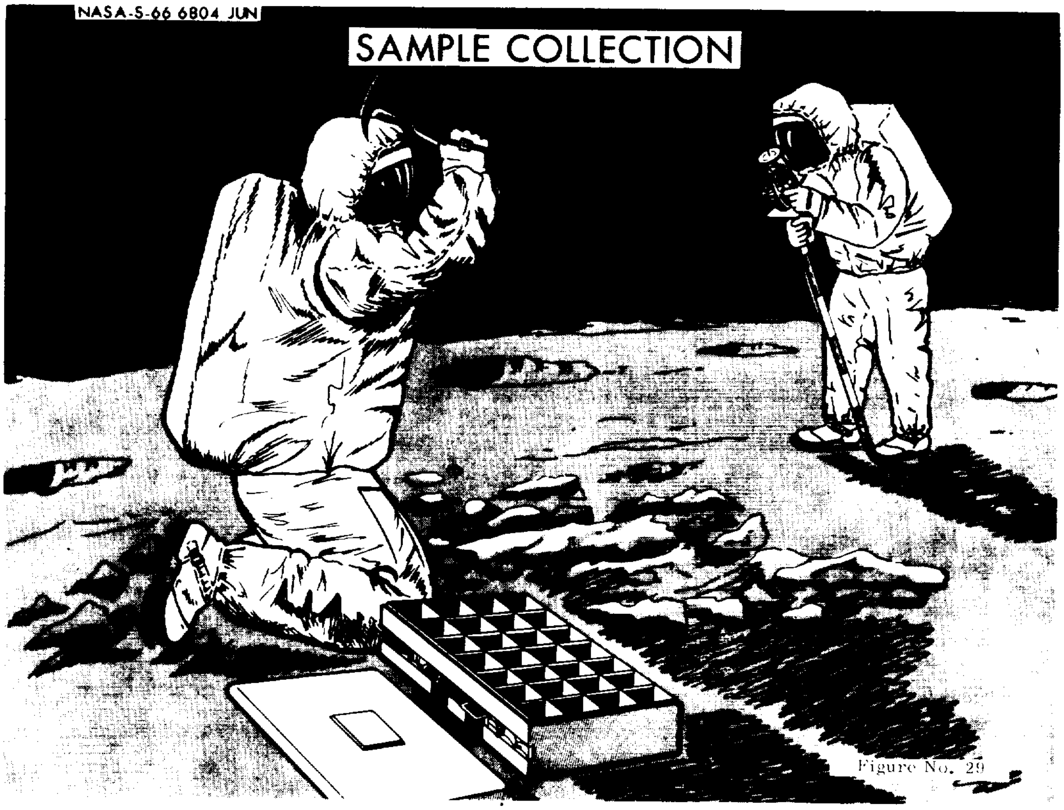
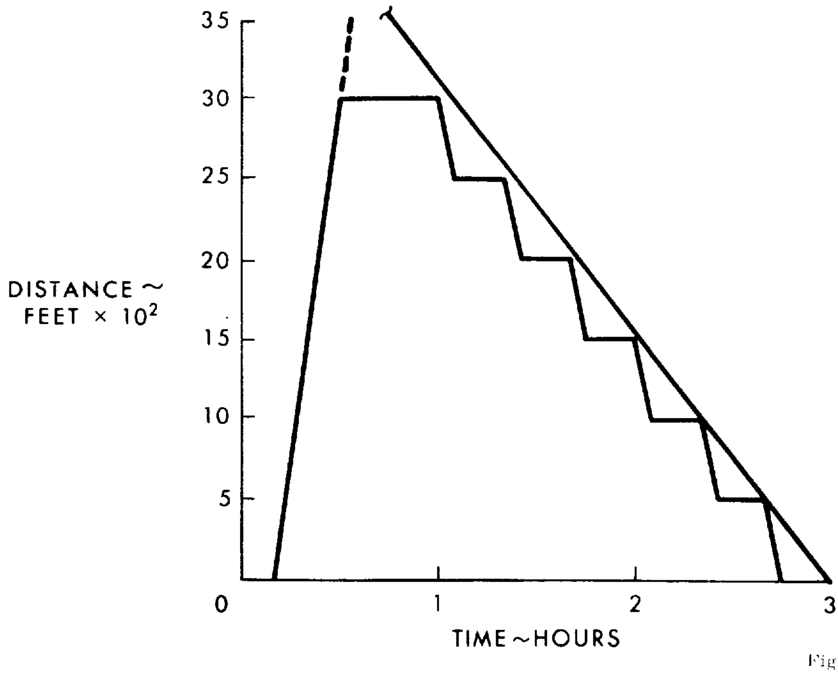

NASA-S-66 6804 JUN
SAMPLE COLLECTION

Figure No. 29



NASA-S-66-6509 JUN

SAFE SEPARATION FROM LM SECOND EXCURSION

Figure No. 30

NASA-S-66-5201 JUN

LUNAR SURFACE SIMULATION PROGRAM

- OBJECTIVES
  - DEVELOPMENT TESTING FOR DERIVATION OF DESIGN INFORMATION
  - GENERATION OF DETAIL MISSION PLANNING INFORMATION
  - ESTABLISHING OF OPERATIONAL PROCEDURES AND CREW TRAINING CRITERIA
- METHODS
  - WORKING GROUP FOR DEFINITION OF PROGRAM REQUIREMENTS
  - PROGRAM IMPLEMENTATION COMBINING UTILIZATION OF MSC AND U.S.G.S. FACILITIES

Figure No. 31

NASA-S-66-6384 JUNE

LUNAR SURFACE SIMULATION PROGRAM

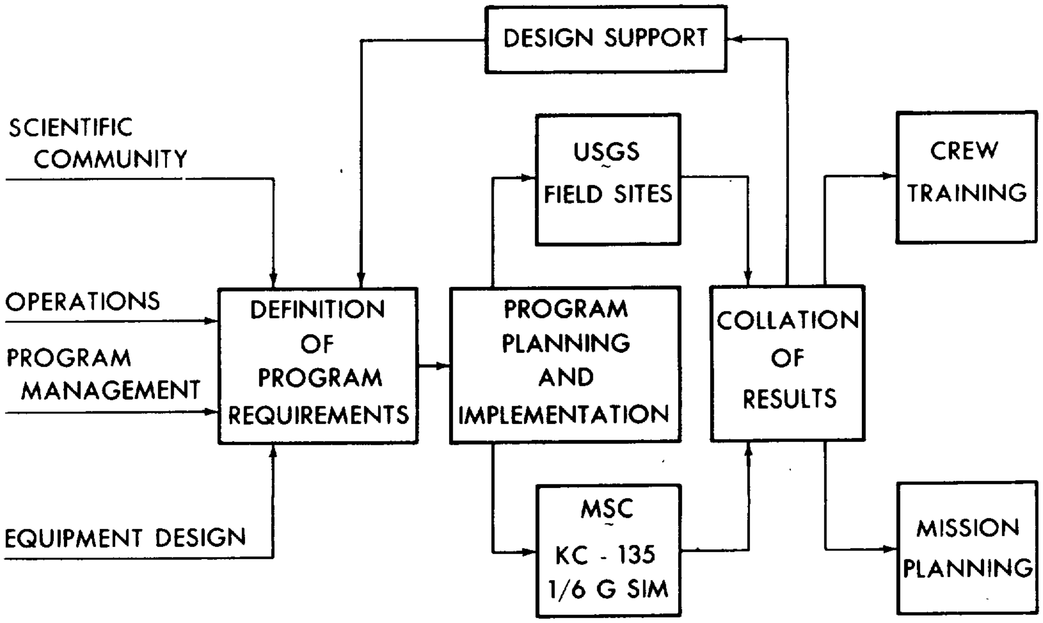

DESIGN SUPPORT
SCIENTIFIC COMMUNITY
USGS FIELD SITES
CREW TRAINING
OPERATIONS
PROGRAM MANAGEMENT
DEFINITION OF PROGRAM REQUIREMENTS
PROGRAM PLANNING AND IMPLEMENTATION
COLLATION OF RESULTS
EQUIPMENT DESIGN
MSC KC - 135 1/6 G SIM
MISSION PLANNING

Figure No. 32

NASA-S-66-5250 JUN

1/6 G SIMULATION

LUNAR SURFACE TASKS

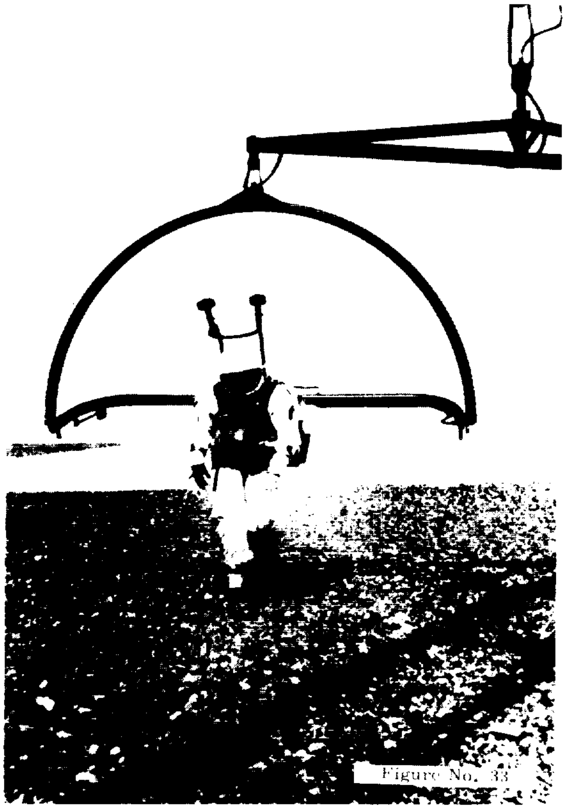


Figure No. 33

NASA-S-66-5249 JUN

KC-135 SIMULATION

WALKING ON LUNAR SURFACE

Figure No. 34

NASA-S-66-5245 JUN

LABORATORY SIMULATION

SAMPLE BAGGING

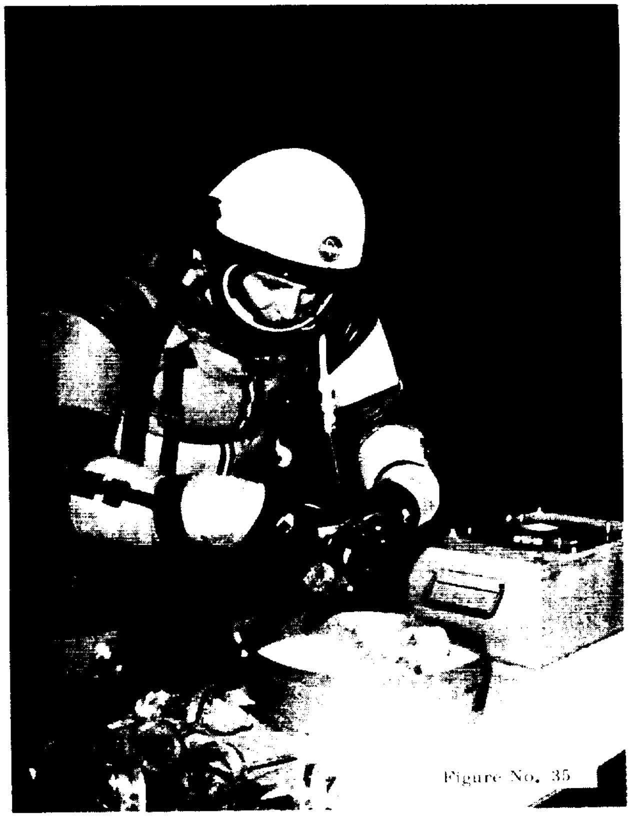


Figure No. 35

NASA-S-66-5248 JUN

SPECIAL TASK SIMULATION

EQUIPMENT DEPLOYMENT

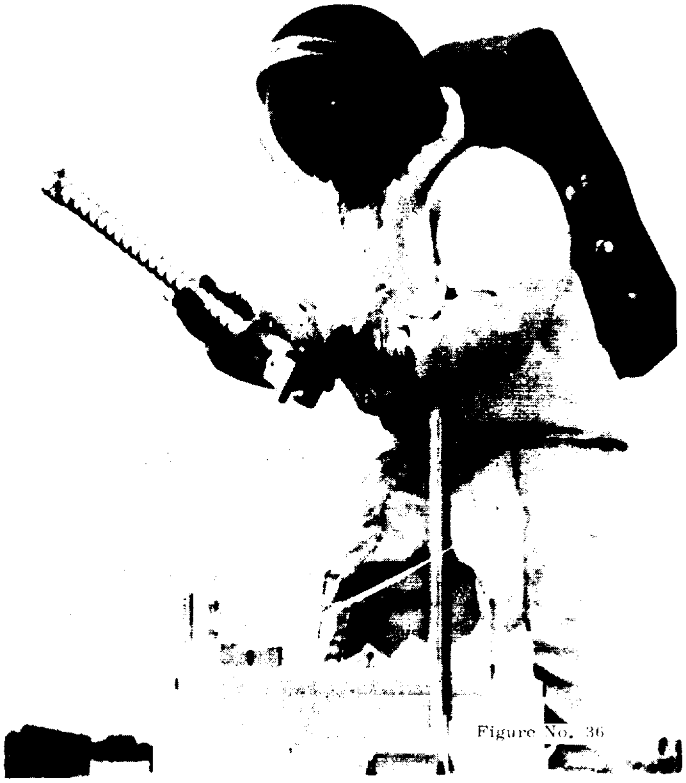


Figure No. 36

NASA-S-66-5849 JUL 5

SIMULATION SUMMARY

ACTIVITY	CONDITIONS				LOCATION				EQUIPMENT				
	SUITED	UNSUITED	1 G	1/6 G	USGS	MSC	LAB	WPBFB	CONCEPT	M U	PROTO	FINAL	EARLY SUIT
● SAMPLING													
• OUTCROP OR LARGE BLOCK	✓	✓	✓	✓	✓			✓	✓				✓
• COURSE FRAGMENTS	✓	✓	✓		✓				✓				✓
• FINE MATERIAL	✓	✓	✓		✓		✓		✓				✓
● LOCOMOTION													
• LEVEL TERRAIN	✓	✓	✓	✓	✓	✓	✓	✓					✓
• GENTLE SLOPES	✓	✓	✓	✓	✓	✓							✓
• STEEP SLOPES	✓	✓	✓		✓				✓				✓
● EQUIPMENT UTILIZATION													
• GEOLOGICAL TOOLS	✓	✓	✓	✓	✓	✓			✓				✓
• SAMPLE CONTAINERS	✓	✓	✓				✓		✓				✓
• LSEP	✓	✓	✓			✓		✓	✓	✓			✓
● PHOTOGRAPHY/ DESCRIPTION													
• OF SAMPLES	✓	✓	✓		✓				✓				✓
• NEAR FIELD	✓	✓	✓		✓				✓				✓
• FAR FIELD	✓	✓	✓		✓				✓				✓

Figure No. 37

NASA-S-66-5181 JUN

PROGRAM FLEXIBILITY

- LSEP ACCOMMODATES ANY OF THE CANDIDATE EXPERIMENTS. INTERCHANGEABILITY POSSIBLE UP TO CONFIGURATION FREEZE FOR FLIGHT ARTICLE
- LSEP IS DESIGNED TO PERMIT OPERATION OF MULTIPLE INSTALLATIONS WITHOUT INTERFERENCE
- THE EQUIPMENT AND PROGRAMMED ACTIVITIES ARE CONFIGURED TO PERMIT A MAXIMUM BENEFIT FROM THE EXERCISE OF CREW JUDGEMENT AND SELECTIVITY

Figure No. 38

DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

by

Morris V. Jenkins

DETAILED MISSION PLANNING

CONSIDERATIONS AND CONSTRAINTS

Figure 1 - Apollo Rendezvous

The current status of Apollo rendezvous is such that nominal and contingency plans are at an advanced stage. The current activity may be summarized as an evaluation of detailed operational trade-offs. This is somewhat emphasized by the fact that computer program implementation is proceeding in the Lunar Module Guidance Computer (LGC), the Command Service Module Computer (CMC), the Abort Electronics Assembly (AEA), and in the ground Real Time Computer Complex (RTCC). Crew guidance and navigation procedures are being established for AS-278 and the basic ground rule is that whatever is established for the development flights (AS-278 and AS-503) shall be used on the lunar landing mission. These procedures are being established for AS-278 now.

During the discussion, it will be seen that cross checking navigational sources are available to aid the rendezvous flight plans. It will be seen that there is a preference for Lunar Module active rendezvous wherever it is possible and that Lunar Module-Command Service Module (LM - CSM) combination plans are the next preference. In addition, it will be seen that there is a CSM potential rendezvous in all cases. However, an entirely CSM active rendezvous is the last preference and steps are being taken to lessen the probability of resorting to CSM total active rendezvous.

During the last year, there has been a change of thinking as regards the modes of rendezvous to be employed, and this change is briefly discussed in the next section.

Figure 2 - The Change

In the past, the nominal mode of rendezvous was the "direct ascent", in which the launch puts you on a direct intercept trajectory. The direct ascent was also used in contingencies whenever possible; for example, at the commencement of powered descent. However, when an abort was required in which large catch up rates were involved, for example, an abort from hover, then an intermediate parking orbit was required before the final transfer. So in the past, it should be recognized that two modes of rendezvous were required.

The present plans are that the "concentric flight plan" shall be used for all nominal and all normal non time critical contingencies. For example, it can be used for all aborts from all phases of descent.

For time critical rendezvous from the lunar surface, an "equivalent direct ascent" will be employed.

The time critical situation, for example, could be brought about by failure in a life support system.

Both the "concentric flight plan" and the "equivalent direct ascent" plans will be discussed in subsequent sections. The objectives of the change are discussed in the next section.

Figure 3 - The Objectives

The objectives of the employment of the Concentric Rendezvous Flight Plan (CRFP) are:

1. To facilitate a time line in which crew and ground specialists can participate in operational decisions without a time press. This has been made possible because the final critical transfer trajectory is arranged to occur on the front side of the moon.

2. To facilitate effective crew monitoring techniques. It will be seen that the concentric coast prior to the Terminal Phase Initiation allows some effective checking by the crew before the critical transfer phase.

3. To facilitate a policy of conservative step-by-step commitment. At no time is there a "one shot" critical expenditure of fuel.

4. To allow full utilization of MSFN as an independent source of navigation to confirm onboard determination of corrective maneuvers. It is in the coasting phases prior to the maneuvers that the MSFN is particularly useful.

5. In the event of a failure of the onboard prime guidance system, MSFN updates of the Abort Guidance System (AGS) and provision of targeting for transfer and midcourse corrections may be found to be particularly useful. This is because of the errors incurred by the drift of the AGS attitude gyros.

6. To provide a standard plan of action for nominal and all normal abort cases.

7. In the event that a substantial launch window is required for operational reasons, then a significant delta V payload saving may be achieved.

In the next section, we begin the description of the CRFP. The beginning of the CRFP really starts in the prelaunch phase.

Figure 4 - Prelaunch

In the mission planning description by M. P. Frank, it was established that it is intended to achieve a CSM plane change prior to nominal LM ascent. The object of the plane change is to bring the CSM orbit over the launch site at nominal lift-off time.

An essential activity that must occur prelaunch is an update of CSM orbital elements. These orbital elements will be entered into the LGC through the DSKY. The source of the update could be the MSFN or from an onboard CSM determination. However, the latter way may be undesirable because it may incur undesirable activity by the one crewman and/or undesirable use of the CSM Reaction Control System (RCS). It is important that the update occurs in that the LM Inertial Measuring Unit (IMU) alignment is a function of the CSM orbit, but it is the AGS which is directly dependent. The AGS uses the AEA, a small capacity computer, which has some simplifications in its programming based on the IMU alignment relative to the CSM orbit.

The alignment is also a function of lift-off time. The lift-off time is calculated such that with perfect guidance and navigation, the LM will have a concentric coast 15 n.mi. below the CSM prior to Terminal Phase Initiation (TPI).

Through a permanently stored transformation in the AEA, the AGS attitude measurements are referenced to the new IMU alignment.

Also, the LM state is updated prior to launch.

Finally, the AGS time data must be reset because of its limited scale.

The discussion of launch is split into two main topics: monitoring and targeting.

Figure 5 - Launch (Monitoring)

The strategy for monitoring the launch is currently being developed.

Normally, the prime guidance will be used during launch. However, the AGS has stored in it equivalent launch targeting. This means that reference may be made to the 8 Ball display slaved by selection, for monitoring purposes only, to the AGS attitude reference. Therefore, it can be seen on the 8 Ball in terms of attitude errors, what the AGS assessment of the performance of the prime guidance system is.

To aid launch monitoring, it is intended to have cockpit displays giving altitude rate, and crossrange velocity.

Additionally, there will be keyboard displays (DSKY and DEDA) of vacuum pericynthion, absolute velocity, and accumulated velocity along the X axis.

Further, it is thought that some out of the window, track monitoring is possible by observations of previously noted special terrain features.

Rendezvous radar in the latter stages of the launch will detect down-range velocity and it may be possible to detect gross crossrange positional errors.

If the range and backup onboard guidance systems telemeter their respective state vectors to the ground, and this information is resolved along the line of sight between the ground radar dish and the LM, and if either onboard guidance system has a velocity component error in this direction, it will be detected by MSFN. The MSFN uses Doppler rate measurement which provides an extremely accurate form of measurement in the direction specified. Gross crossrange rate errors will also be detected by MSFN.

Figure 6 - Launch (Concluded) (Targeting)

Both the LM prime and the backup guidance systems have equivalent launch targeting. The launch targeting essentially specifies the following:

1. That by the end of the launch, the LM shall have translated into the CSM orbit plane.
2. The absolute cutoff velocity, which is approximately 30 ft/sec in excess of circular orbit speed, and in free coast conditions, would take the LM to an apocynthion of 30 n.mi.
3. The cutoff flight-path angle is constrained to zero.
4. The cutoff altitude is specified by current planning that it shall be 60,000 ft.

The concentric sequence follows the launch.

Figure 7 - Concentric Flight Plan

The concentric sequence is initiated 30 min elapsed time after launch insertion. The objectives of the concentric sequence are:

1. To provide a period of constant differential altitude coast prior to TPI. This is to facilitate crew monitoring and checking of the guidance system before the critical transfer.
2. To insure that TPI will occur compatible with maximum operational convenience. This involves considerations of lighting conditions for both the CSM and LM and monitoring by the ground network prior to TPI and during the transfer up to the time of first braking.

The concentric sequence consists of the Concentric Sequence Initiation (CSI) maneuver, the Constant Delta Height (CDH) maneuver, TPI as previously mentioned, and Intercept. The concentric sequence is determined as a whole prior to the CSI maneuver. The manner in which the targeting is determined is described in "Software Compatibility with Lunar Mission Objectives". The subsequent discussion will temporarily restrict itself to the nominal lift-off time case.

The CSI maneuver is a horizontal burn parallel to the CSM orbit plane, using a ΔV of approximately 60 ft/sec. The fact that it is a posigrade horizontal addition to the existing safe orbit implies that the pericynthion altitude will be raised higher than that of the existing safe orbit. Therefore, it is a safe burn, not predicated on the quality of the navigational update subsequent to launch. The burn is made parallel to the CSM orbit plane and therefore, any out of plane condition incurred by launch errors is constrained. It should be noted, however,, that the magnitude of the CSI maneuver is a function of the best estimate of the CSM and LM orbits prior to the maneuver.

The CDH maneuver is made at a time determined as a by product of the determination of the entire concentric sequence which occurs prior to the CSI maneuver. The time of the initiation of the CDH maneuver is held constant in order to keep the operational time line intact. The objective of the maneuver is to produce a minimum variation in differential height subsequent to it and prior to TPI. For the nominal case, the CDH maneuver ΔV is approximately 70 ft/sec. The burn is constrained to be parallel to the CSM orbit and, in general, this is a positive reduction to the out of plane characteristics of the LM trajectory. If navigation and guidance conditions were perfect and the CSM is in a circular orbit, then this maneuver would achieve a concentric trajectory at a constant distance below the CSM orbit. This is really the derivation of the name of the flight plan. Furthermore, it should be realized that the pericynthion altitude has again been raised. Although this maneuver is made behind the moon, it is an extremely safe maneuver. It also should be noted that although the time of initiation has been maintained, this maneuver is also a function of the current knowledge of the LM and CSM orbital characteristics. Again the manner of the targeting is described in "Software Compatibility with Lunar Mission Objectives".

In general, there will have been execution errors associated with both the CSI and CDH maneuvers, in addition to the navigational errors subsequent to launch, and therefore, insistence on initiation of the TPI maneuver at a previously planned time would incur ΔV penalties. This is because at the same time a standard transfer of 140° center angle travel of the transfer trajectory is intended. It is, of course, more desirable to initiate the transfer when there is a near optimum phase-height relationship. This means that all previous dispersion errors can be absorbed for near zero ΔV penalty. The manner in which this is accomplished in the concentric flight plan and it is an inherent property of it, is to initiate the TPI maneuver at a set elevation angle. This elevation angle is the elevation angle of the line of sight of the LM to the CSM referred to the current LM horizontal plane. Therefore, the onboard computers predict when this line of sight angle will occur and this becomes the newly selected time of TPI. In the framing of the original flight plan, allowance will have been made for dispersions and adjustment of TPI time does not incur serious operational consequences.

Another property of the Concentric Flight Plan (CFP) is that if the crewmen know the differential height prior to TPI, then they know the direction and magnitude of the TPI burn. The magnitude of the direction of the burn, and hence ΔV, is proportional to the differential height and the direction of the burn will be along the line of sight towards the target vehicle. These two properties have an empirical derivation and an exact theoretical one should not be sought. Independently crewmen with the aid of the rendezvous radar by noting differentials in range and range rate referred to the nominal flight plan, can detect the actual phase and height relationship such that in emergency backup circumstances they can manually perform the TPI maneuver making use of the properties of the TPI maneuver previously mentioned, i.e., they would also make use of the set elevation angle of the line of sight and a knowledge of the height differential. Normally the LM prime guidance or backup guidance will control the maneuver.

With the sun direction and line of sight direction as indicated in Figure 7, it is a matter of conjecture whether the crewman in the right hand seat sitting in front of the "shaded" window will be able to see the sun illuminated CSM at time of TPI. The distance is 34 mi. but the possibility hinges on the amount that the right hand window is shaded.

An amplification of the determination of the targeting for TPI occurs in the above reference. It will suffice here to say that the TPI maneuver is intended to achieve an intercept at a position corresponding to 140° of target vehicle travel or an equivalent elapsed time of target vehicle travel. It will incur a ΔV of approximately 25 ft/sec.

At a fixed elapsed time after TPI, a midcourse correction will be applied. Again, the objective will be to bring about intercept at the same position intended at TPI. It will be noted later that there are cross checking navigational guidance systems being brought into play during the concentric sequence flight plan.

At 3 mi. relative range, the first braking will occur. This will be a maneuver again aimed at an intercept, but at the same time instantaneously reducing the closing rate to approximately 20 ft/sec. Subsequent similar maneuvers at reduced relative ranges will accomplish orbit matching characteristics with the CSM. The summation of the ΔV's of these terminal phase maneuvers will be somewhat more than 25 ft/sec.

The docking will be performed manually which will take some time and sometime later behind the moon, crew transfer will be achieved. However, it will be noted that the MSFN was able to participate in the critical navigational guidance phases. The discussion is centered around a nominal launch lift-off time which may be defined as that which would result in a 15 n. mi. differential if navigation and guidance conditions were perfect. The late lift-off is 5-1/2 min. later and corresponds to a zero CSI maneuver, i.e., the late window is essentially defined by the safe orbit at insertion, referred to AGS launch insertion errors. In the case of the early lift-off, the CDH maneuver will occur approximately 150 from the CSI. In the case of the late lift-off, the CDH maneuver occurs 90 later after the nominally zero CSI maneuver. In general, the CSI maneuver for the late lift-off will not be zero because of launch errors. It should be noted however, that only posigrade CSI maneuvers are tolerated. If the dispersed conditions at CSI for the case of the late lift-off potentially demand a retro-impulse, then in this case, no maneuver will occur and the consequence will be that the finally selected TPI time after CDH will be displaced from the original nominal TPI time. In this case, perhaps a little more than desirable, but the probability of the case is low.

Figure 8 - Concentric Flight Plan in Relative Coordinates

The flight plan indicated on Figure 7 is shown in this coordinate system so that it can be seen to scale. The coordinate system is set in the CSM and height differential is shown in the direction of the "current vertical" through the CSM and phase difference is shown in the direction of the current horizontal through the CSM plane. Essentially, the moon has been opened up and made flat. Hence, the 80 n. mi. circular orbit becomes a horizontal line. The concentric coast is shown parallel to it.

The notes of background refer to the conditions with which the sextant has to view the LM. The nautical mile figures refer to the relative range between the LM and the CSM. The orbital rate of the CSM is approximately 3 per minute, hence there is a large loss of phase during the 6.9 min. launch period.

Study of Figure 8 gives a fresh perspective to the CFP.

In the next section a brief summary of the propulsion systems used is given.

Figure 9 - Utilization of Propulsion Systems

The main engine, with a fixed nozzle, is used to provide the power for launch. In addition, a forward thrusting RCS engine is used to null out the effects of cg thrust offset. Both engines are using Ascent Propulsion System (APS) fuel. It should be noted that there is sufficient APS fuel to launch into a safe orbit even if there is an RCS failure where resort has to be made to an RCS engine thrusting backwards to null out the cg-thrust offset. This is true when referred to all probable adverse cg offset conditions.

It is considered wise to use the residual useable APS fuel for the next maneuver after launch, i.e., the CSI maneuver. However, the pressurized RCS tanks must first provide fuel for ullage action by the X axis RCS thrusters. Ullage having been provided, then following a switching sequence, the same RCS thrusters are able to use the residual APS fuel.

For the remainder of the maneuvers for rendezvous, the RCS engines are used and the fuel is taken from the pressurized RCS tanks. It is planned to use Z axis thrusters for nearly all cases from TPI on.

At the bottom of Figure 9, there is a reminder that the CSM may be involved in active rendezvous maneuvers. When the ΔV is more than 12 ft/sec, CSM RCS will be used for ullage and then main engine thrust is applied. If the ΔV required is less than 12 ft/sec, then the CSM RCS alone will be used. An example of this is the final terminal thrustings to complete CSM active rendezvous.

The next section reminds us that a time critical rendezvous may be required.

Figure 10 - Time Emergency Rendezvous

This figure indicates the time critical flight plan which will be employed. It consists of a standard insertion from launch exactly as in the nominal mode. This is followed by approximately a 5 min. review of the launch cutoff conditions. Subsequent to this, a direct transfer of approximately 100° is selected. Subsequent to the transfer initiation, midcourse corrections will occur. At a relative range which will call for braking maneuvers, the closing rate prior to first braking will be higher than that in the nominal mode. As indicated in the figure, the time line is a busy one and this mode of rendezvous will only be employed in time emergency circumstances.

Necessary studies have taken place to examine vehicle to vehicle tracking possibilities for the CFP.

Figure 11 - Optical Tracking Assumptions

Figure 11 summarizes the sextant tracking assumptions for the studies and the conditions have two major categories:

1. Where the line of sight has a radial component outwards.

2. Where the line of sight has a radial component inwards.

Associated with the first category, sextant viewing conditions are acceptable if the angle between the sun direction and the line of sight is greater than 20°.

Associated with the second category, is the probability that there will be a lunar background to the LM. This corresponds with the LM being below the horizon.

The figure indicates the maximum viewing range for various background conditions and it should be noted that it is the sun illuminated LM rather than the flashing light of the LM, that can be detected against a sunlit lunar background.

The kind of analysis that has been made will be shown in the next series of figures, but first, attention must be paid, to the notation used.

Figure 12 - Explanation of Line of Sight Convention

Figure 12 gives an explanation of the line of sight convention. The zero datum for the local horizontal through the CSM in the direction of motion and a 360° convention is used.

Figure 13 - CSM Sextant Tracking LM (Nominal Flight Plan)

This is an example of conditions you have when a landing has been made with the sun elevation of 18° and when the lift-off corresponds to the nominal and the concentric altitude of 65 n. mi.

The status of the mission is given by the trace with the events of the CFP indicated with appropriate symbols. The status of the mission is given in terms of the line of sight, using the convention shown in the previous figure, and the longitude of the LM.

At insertion, the CSM looks back to see the LM on the horizon and as the LM at a lower altitude begins to catch up, the line of sight steepens and the LM then becomes viewed against a lunar background. First the background is sunlit and then it is that of earth reflected light, but the range is greater than 250 n. mi. and the flashing light can be acquired. Hence, the trace shows a series of dots to indicate trajectory status. However, the tracking summary at the bottom of the diagram quite correctly indicates a gap in tracking.

After CSI, the LM altitude increases and the LM is seen against a space background which corresponds to a dark background, and since the range is less than 500 n.mi., the LM flashing light can be seen and the tracking summary shows a definite line of "footballs" indicating that acquisition is possible. Following the circularization or CDH maneuver, the line of sight steepens again and the LM is viewed against a sunlit background. However, the range at TPI is 34 n.mi. and hence, just prior to it, the range is 40 n.mi. or less and hence, the sun illuminated LM can be seen against the sunlit background. After TPI, the relative range becomes less and hence, sextant acquisition continues to be possible. Toward main braking, the lunar background is that of earth reflected light and again, by reference to the Optical Tracking Assumptions in Figure 11, it will be seen that acquisition is possible and this is recorded in the tracking summary. In fact, favorable lighting conditions prevail right through to docking.

The assumption was that the rendezvous radar could acquire when the relative range was 400 mi. or less and in the example given, the range is always less than 400 mi. and, therefore, rendezvous radar acquisition is continuous.

MSFN acquisition can occur from approximately 100° east to 100° west and this is indicated in the tracking summary.

In the case of the late window, the concentric altitude is 30 n.mi. and the concentric coast between circularization and transfer would be done with the LM below the horizon and at a greater relative range with the LM against a sunlit background. Hence, when a spectrum of flight plans is reviewed later, a gap in sextant viewing corresponding to this phase for the late window should be anticipated.

The late window resembles very closely, the conditions for abort from hover. Therefore, it should be borne in mind when considering the late window that as regards lighting conditions, a similar pattern prevails for the abort from hover.

Figure 14 - CSM Sextant Tracking LM (Abort from Initiation of Powered Descent)

This is an example of an abort from the initiation of powered descent. The landing site is at zero longitude and the sun elevation was 30° upon landing. The CSM is 10° behind the LM initially at the abort warning and the LM commences its powered descent at 50,000 ft. It will be seen on Figure 14 that the sextant is viewing the LM below the horizon at insertion and the LM is viewed against a sunlit background.

At CSI, the LM is inserted into an orbit the apocynthion of which, is 130 n. mi. and a concentric coast above the CSM commences at this altitude. Therefore, before circularization, the sextant will have to look above the local horizontal to see the LM, and hence, the trace giving the status of the mission indicates an elevation angle of above zero for that mission phase.

After circularization, the line of sight comes within 20° of the sun direction and when it does, it will be noted that there is a gap in the tracking summary at the bottom of the figure. Thereafter, there is a dark space background for the sextant to see the LM flashing light while the line of sight steepens and sextant tracking conditions will remain favorable right through to main braking, which is followed by subsequent terminal phase maneuvers. However, it is not suggested that the sextant is of signifcant operational use during the terminal maneuvers.

The rendezvous radar is within its specification acquisition range of 400 n. mi. and therefore, continuous rendezvous radar range tracking is indicated.

Again, it will be noted that the MSFN can track approximately between 100° east and 100° west.

This is an example of sextant tracking when the LM has to go above and let the CSM catch it up and it will be seen that there are no particular sextant tracking problems in this case. The next figure shows several cases involving nominal flight plans for early and late window, and different sun elevation angles at landing. Similarly, abort cases from the initiation of powered descent are shown.

Figure 15 - Tracking Summary

The cases are identified on the left of the figure, the sun directions during rendezvous are shown on the right. The key to this figure has been explained on the two previous figures. It will be noted that from CSI onwards, there are always two sources of navigation. It will be noted in general, that from TPI onwards, there are three sources of navigation. The exceptions to this are the late window cases and similarly, aborts from hover would suffer loss of CSM sextant tracking. The reason for the gap around TPI was mentioned in the discussion of the late window nominals.

However, the potential of the MSFN should not be forgotten. A reminder of this potential is given in the immediately following sections.

Figure 16 - MSFN Error Analysis Assumptions

A summary of the error assumptions is given on this figure. It will be seen that standard 2-way and 3-way error source assumptions have been adopted. MSC and GSFC are in agreement on the quantitative values.

Station location biases were inserted into the study. Station positions were of the order of 100 ft. in error. However, it must be realized that this is not particularly important as the only observation used in the trajectory determination processes is Doppler rate. The systematic error on the moon's gravitational constant is indicated and the JPL derivation is indicated. The analysis used a simulation of the proposed RTCC orbit determination scheme. The three components of velocity are solved for, the three components of position are solved for, and the individual systematic errors on both of the two slaved situations are solved for. This is what is meant by bias on 3-way Doppler as shown at the bottom of the figure.

The next two figures summarize some of the results of the analyses.

Figure 17 - MSFN Tracking Uncertainties (3σ)

This is an analysis of the MSFN tracking during a CFP where the launch was from 45° west. The short tracking pass subsequent to launch is, in itself, of marginal use. This is, in some measure, due to the length of time for update by voice and telemetry verification by the ground. The complete update allowance is of the order of 7 min. However, tracking continues during this time and it is the basis for good a prior information for when the MSFN reacquires when the LM reappears on the eastern limb. Whereupon, convergence is rapid and uncertainties become low. It should be noted that the out of plane uncertainties are reduced because of the two tracking passes, the initial one on the western limb, and the subsequent one on the eastern limb. The analysis did assume what are currently considered to be 3σ execution errors.

The next figure indicates the potential of the MSFN in assisting the AGS.

Figure 18 - (3σ) MSFN Errors at Intercept

In this figure, it must be recognized that a midcourse correction is applied always at 30 min. elapsed time after TPI. The abscissa indicates that TPI is delayed by varying increments of time after MSFN reacquisition on the eastern limb. The left and right ordinates indicate the errors in position and velocity respectively for intended zero miss conditions at intercept. The execution errors at TPI and midcourse are assumed to be insignificant and it is only the navigation performance of the MSFN to guide the LM to an intended zero miss that is illustrated.

Some emphasis has been given to the fact that it is the potential MSFN performance that has been illustrated. This is done because those with operational experience at MSC consider that the network, coupled with the trajectory determination organization of the ground control, must have exercised this capability prior to the lunar landing mission in order to achieve the potential. It must be recognized that none of the development flights orbit the moon; therefore, there is great dependence on the opportunities that could be provided by the Lunar Orbiter program. The extent of the opportunities remain undefined at this time.

The next diagram gives a brief reminder of the main programs involved in the rendezvous capability.

Figure 19 - Main Components of Onboard Rendezvous Programs

This diagram reminds us that three major programs previously mentioned in the paper are formulated as onboard programs in the prime and backup guidance computers of the LM.

The programs referred to are as follows:

1. The "standard insertion" which is used for launch and is the major component of the powered phase of the abort from powered descent. This applies to both the LM prime and backup guidance systems.

2. The second one is direct transfer. This routine is contained in the LM prime and backup guidance systems and is also formulated for the CMC. The final major transfer is always made using this routine and it is essentially used with subsequent midcourse action.

3. The concentric sequence is the third program. This program has, of course, been mentioned and additional information on it is contained in the above reference.

There is a fourth program which has not been mentioned and this is the "external ΔV" program, and is used when a required ΔV input derived externally, either from pilot's notes, charts, etc., or from ground control. •The input is referenced to local polar coordinates such that the crewmen have an understanding of what they are entering into the computer.

All of these programs are being formulated in the RTCC. This is in order that advantage may be taken of the MSFN tracking capability, which in many cases, would provide the source for the determination of targeting to be transmitted by voice.

In the following section, a reminder is given of the total G&N information during rendezvous.

Figure 20 - Provision of G&N Information for the Operation

It will be observed on Figure 20 that there are three sources of observation, i.e., rendezvous radar, MSFN, and the CSM sextant. The diagram indicates the way in which the observations find their way into the onboard computers. If data has to be entered manually through the keyboards, then this is also shown. If this is subsequent to voice transmission, this is shown. Hence, the diagram gives reasonable clarity and an immediate picture of how navigation data finds its way into the computers. The CSM orbital elements are inserted into the LM computer. The LM inertial orbit is subsequently determined by rendezvous radar observations of the CSM. At the same time, the CSM sextant can take observations of the LM and derive the LM inertial orbit and, therefore, potentially this provides an onboard navigational check.

It is most important to note the resultant capabilities indicated on the right hand side of this diagram. Both LM computers have a concentric sequence capability and a contingency time critical "equivalent direct ascent" capability.

The CMC has a direct transfer capability only. However, if the CSM state vector is entered into the LGC, and the LGC calculates a concentric sequence as though the LM had that state. In this way, the LM can prepare targeting information for the CSM which is transmitted by voice and is entered through the DSKY of the CMC.

In the next section, a number of contingency flight plans are being introduced in order to give an indication of how the CFP can be used in contingency cases.

Figure 21 - LM Active Contingency Rendezvous (Phasing at Insertion 60°)

The coordinates of this diagram and the succeeding three are exactly the same as used on Figure 8 (Concentric Flight Plan in Relative Coordinates). An explanation of the coordinates is contained in that section. Briefly, the moon is transformed into a flat moon and measurement in the local vertical direction is that of differential altitude and measurement along the local horizontal is that of phase lead.

In this first diagram, the phase lead of the CSM at insertion is 60°. The familiar CFP maneuvers are employed. It should be realized that when the differential altitude is small, the catch up rate is small. Therefore, though the contingency flight plan has been framed such that the differential altitude subsequent to CDH is of a differential altitude not too dissimilar from that of a nominal lift-off case, more than one orbit of concentric coast is incurred.

Figure 22 - LM Active Contingency Rendezvous (Phasing at Insertion 140°)

In this case, the CSM is further ahead, 140° at insertion. In this case, the greater differential altitude is selected and multiple orbits of concentric coast are incurred. However, the familiar concentric sequence is easily recognized.

Figure 23 - LM Active Contingency Rendezvous (Phasing at Insertion -50°)

In this instance, the CSM is 50° behind the LM at insertion. Consequently, the CSI maneuver drives the LM to an apocynthion above the CSM whereupon the CDH maneuver will be executed to achieve a concentric coast above the CSM prior to TPI. The transfer following CSI is the equivalent of a Hohmann transfer initiated of equal period to the previous orbit.

Figure 24 - LM Active Contingency Rendezvous (Phasing at Insertion -160°)

In this instance, the LM is very much in front of the CSM, in fact, 160° at insertion. It also is in a lower orbit following insertion. This would call for the CSI maneuver to be initiated immediately after insertion in order to end the increasing adverse phase situation. The CSI maneuver transfers the LM high above the CSM and at apocynthion, the CDH maneuver achieves a concentric coast at a considerable differential altitude above the CSM in order to retain the required negative catch up rate. Finally, the TPI maneuver initiates a rendezvous from above.

One of the main reasons for drawing attention to these contingency flight plans shown in these particular coordinates was in order that a particular feature could be recognized. This feature is that the concentric sequence pattern is just as applicable in these cases as in the nominal case, i.e., the concentric sequence program onboard has the capability to cope with these contingency cases.

The next section will give a summary of the overall Apollo rendezvous situation.

Figure 25 - Rendezvous Categories

The main correlating parameter by which a rendezvous situation can be immediately judged is phase lead. Particularly is this the case if a nominal CSM circular orbit of 80 n.mi. altitude is a constant assumption. Therefore, in these circumstances, all rendezvous situations can be categorized according to phase and a summary is graphically indicated on this diagram.

All phase leads of the CSM are calibrated relative to the LM at insertion and zero phase lead is found at the bottom of the circle. All phase leads referred to on the left hand side of the circle refer to the central angle that the CSM is in front at LM insertion. The number of degrees on the right hand side of the circle refer to the central angle lag that the CSM has at LM insertion.

First of all, the nominal launch window is identified, i.e., for a nominal lift-off, the CSM will be 18° ahead at launch cutoff. If the lift-off occurred $5\text{-}\frac{1}{2}$ minutes later, corresponding to the end of the launch window, then the CSM would be 32° ahead of the LM at insertion.

In the late case, the concentric differential altitude is 50 n.mi. In the early window case, the concentric differential altitude is 15 n.mi. Extrapolating this information, it can be understood that when the phase lead is 10° at LM insertion, then the corresponding concentric differential altitude would be zero. In this way, we have identified a physical boundary at which the CFP neither acquires a positive or negative catch up rate during the concentric coast. It should be noted that in the instance of this particular phase lead being incurred, then the CFP can be framed to have a concentric differential altitude coast by advancing or retarding the time of CSI. It is now convenient just to use the 10° boundary as a phase delineation where, if the CSM phase lead at insertion is greater than 10°, then the concentric altitude coast will be below that of the CSM. Alternatively, if the phase lead of the CSM is less than 10°, and this includes phase lags, then the LM concentric coast will be above that of the CSM for completely LM active rendezvous.

Circular arror A_1 indicated all the CSM phase leads at insertion from which the LM can catch up from below within the ascent module lifetime of 11.3 hours by rendezvousing from below.

If it is desired that the concentric coast shall be specifically 15 n.mi. below the CSM, then circular arrow A_2 indicates the reduction of range of lead angles corresponding to which it is possible for LM active rendezvous within 11.3 hours.

Circular arrow A_3 indicates, in the main, the phase lags of the CSM at insertion corresponding to which LM active rendezvous may be achieved within the ascent module lifetime following a concentric coast above the CSM. Because this entails the LM going higher than the CSM, it must be noted that A_3 is bounded by ΔV capability allowing sufficient negative catch up for rendezvous within required lifetime.

It should be noted so far that we have only dealt with in plane rendezvous cases. This is not unnatural in that nominally the CSM has made a desirable plane change prior to LM ascent, or in the case of multiple orbit rendezvous, if for any reason the CSM has not previously made a plane change, then it has the opportunity to make the change at a desired node. It can do this without using any of the ΔV allocated to CSM rescue since there is a 2° allowance in the nominal mission. However, it is interesting to note that an entirely LM active rendezvous with a .5° wedge angle incurred by dispersions that circular arrow A_4 indicates the reduced range of phase lags, in which we include a small range of phase leads 0° to 10°.

Circular arrow B indicates that by maneuvering the CSM that the "black shaded" region between 130° and 160° phase lag can easily be covered. In fact, when the CSM is called upon to maneuver not only can it negate any out of plane situation by a Hohmann transfer down to a lower circular orbit, it can improve the phase situation such that the total time to rendezvous will be reduced.

In the event of an any time lift-off where the LM is forced into undesirable phase situations, together with the circumstances that the LM propulsion systems just manage to insert the LM into a safe orbit, then it should be noted that the nominal CSM plane change capability, plus the ΔV allowance for CSM rescue, makes it possible in the last resort for the CSM to rescue the LM regardless of the initial phase situation and out of plane displacements of less than 2°. Again, the lifetime assumption for the ascent module is assumed to be 11.3 hours and nominal allowances were made for docking and crew transfer.

In the next section, rendezvous modes in order of preference, are discussed.

Figure 26 - Rendezvous Logic

Before considering the order of preferences, it should be remembered that the main constraints are: (1) ascent stage lifetime, (2) LM ΔV available, and (3) clear pericynthion transfers for both LM and CSM.

A desirable condition is that of satisfactory lighting during the terminal phase. Another very desirable condition is that the final terminal maneuvers are accomplished by the LM RCS thrusters.

The first preference is a LM active rendezvous. Furthermore, in non time critical situations, the concentric sequence will be performed and rendezvous shall be achieved in approximately one orbit. In time critical situations, the "equivalent direct ascent" plan shall be employed. If entirely LM active rendezvous can be accomplished from below by employing multiple orbits and the rendezvous can be accomplished well within the ascent module lifetime, then this is considered preferable to combined maneuvers.

Combined maneuvers involving CSM maneuvers prior to the employment of the CFP for the LM may be utilized for either of the following reasons:

1. To accomplish the rendezvous well within the LM ascent stage lifetime.

2. To avoid the situation where the LM has to rendezvous subsequent to a substantial differential concentric altitude above the CSM prior to rendezvous from above. This latter case may also be bounded by the LM V budget. In addition, it may be contrary to satisfying operational requirements.

The third preference is an entirely CSM active rendezvous and would only be employed in contingency cases where the LM propulsion systems were just capable of inserting the LM into a minimum safe orbit.

At this juncture, an overall summary is given.

Figure 27 - Summary

There are rendezvous plans for every phase situation and for all practical out of plane conditions, specifically, less than 2°.

In all normal cases, both nominal and non time critical contingency cases, the concentric sequence standard pattern of procedures will be employed.

An equivalent direct ascent plan is available for time critical rendezvous situations.

Steps have been taken to utilize more efficiently the LM backup guidance system capability. The concentric sequence can be controlled by it and in addition, it also has the equivalent direct ascent capability. In this way, the probability of having to resort to an entirely CSM active rendezvous has been reduced.

Finally, it should be noted that in all non time critical rendezvous situations, joint ground and onboard decisions may be made prior to and subsequent to time critical intercept trajectory.

Questions and Answers

LUNAR EXCURSION MODULE ASCENT AND RENDEZVOUS

Speaker: Morris V. Jenkins

1. Mr. Richter - Are DEDA inputs required for inputs to the AGS using normal flight plan? Subsequent to launch?

   ANSWER - Radar information must be keyed into the AGS.

2. Dr. Rees - Is such a short lunar prelaunch checkout adequate?

   ANSWER - There is an extensive checkout of the LM prior to earth launch. The LM is also checked out as thoroughly as possible following LM landing and again prior to launch.

3. Dr. Rees - What is the maximum LM lunar surface stay time?

   ANSWER - Approximately 35 hours for the descent stage. The ascent stage has a contingency capability of about 9 hours, plus an ascent time of about 2.5 hours.

4. Mr. Stern - What data exists on coverage by 2 MSFN stations, as compared to 3 station coverage shown in presentation?

   ANSWER - This data is available and will be sent to Mr. Stern.

NASA-S-66-6848 JUN

APOLLO RENDEZVOUS

- NOMINAL AND CONTINGENCY PLANS ARE AT AN ADVANCED STAGE
- EVALUATING OPERATIONAL TRADE-OFFS
- IMPLEMENTATION IN LGC, CMC, AEA, RTCC
- CREW G AND N PROCEDURES AS-278 ⟶ AS-503
- CROSS CHECKING NAVIGATIONAL SOURCES
- LEM ACTIVE RENDEZVOUS POSSIBLE IN ALL CASES
- LM-CSM COMBINATION PLANS NEXT PREFERENCE
- CSM POTENTIAL RENDEZVOUS IN ALL CASES
- STEPS TO LESSEN PROBABILITY OF RESORTING TO CSM TOTAL ACTIVE RENDEZVOUS

Fig. 1

NASA-S-66-6572 JUN

THE CHANGE

PAST	PRESENT
● DIRECT ASCENT – NOMINAL – CONTINGENCIES WHENEVER POSSIBLE • E G COMMENCEMENT OF POWERED DESCENT	● CONCENTRIC FLIGHT PLAN – NOMINAL – ALL NORMAL NON-TIME CRITICAL CONTINGENCIES • E G ABORTS FROM ALL PHASES OF DESCENT, COAST RIGHT THROUGH TO HOVER
● INTERMEDIATE PARKING ORBIT • CONTINGENCIES REQUIRING LARGE CATCH UP RATES • E G ABORT FROM HOVER	● EQUIVALENT DIRECT ASCENT • FOR ABNORMAL TIME CRITICAL SITUATIONS • E G FAILURE IN LIFE SUPPORT SYSTEM

Fig. 2

NASA-S-66-6548 JUN

THE OBJECTIVES

- TO FACILITATE A TIME LINE IN WHICH CREW AND GROUND SPECIALISTS CAN PARTICIPATE IN OPERATIONAL DECISIONS WITHOUT A TIME PRESS
- TO FACILITATE EFFECTIVE CREW MONITORING TECHNIQUES
- CONSERVATIVE STEP-BY-STEP COMMITMENT
- FULL UTILIZATION OF MSFN AS AN INDEPENDENT SOURCE OF NAVIGATION TO CONFIRM ONBOARD DETERMINATION OF CORRECTIVE MANEUVERS
- ONBOARD GUIDANCE FAILURE: MSFN UPDATE OF AGS OBJECTIVES FOR TRANSFER AND MIDCOURSES
- STANDARD PLAN OF ACTION FOR NOMINAL AND ALL NORMAL ABORT CASES
- ΔV PAYLOAD SAVING FOR A SUBSTANTIAL LAUNCH WINDOW

Fig. 3

NASA-S-66-6574 JUN

PRELAUNCH

- CSM PLANE CHANGE
- UPDATE OF CSM ORBITAL ELEMENTS INTO LGC
- CALCULATE LIFT-OFF TIME
- ALIGN IMU
- REFERENCE AGS ATTITUDE MEASUREMENTS TO NEW IMU ALIGNMENT
- UPDATE LM AND CSM STATE VECTORS
- RE-SET AGS TIME DATUM

Fig. 4

NASA-S-66-6573 JUN

LAUNCH

MONITORING

- ALTITUDE ERRORS - COMMAND AND RESPONSE (COMPARE PGNCS AND AGS) ON 8 BALL DISPLAYS
- COCKPIT DISPLAYS: h, $\dot{h}$, V_{CR}
- KEYBOARD DISPLAYS: h_P, V_I , V_{ACC}
- OUT-OF-WINDOW TRACK MONITORING
- RENDEZVOUS RADAR
- MSFN $\dot{r}$ AND $\dot{r}_{CR}$

Fig. 5

NASA-S-66-6559 JUN

LAUNCH (CONCLUDED)

TARGETING

- PGNCS AND AGS EQUIVALENT
- YAW STEER INTO CSM PLANE
- |V| CUTOFF
- γ CONSTRAINED TO ZERO
- CUTOFF ALTITUDE

Fig. 6

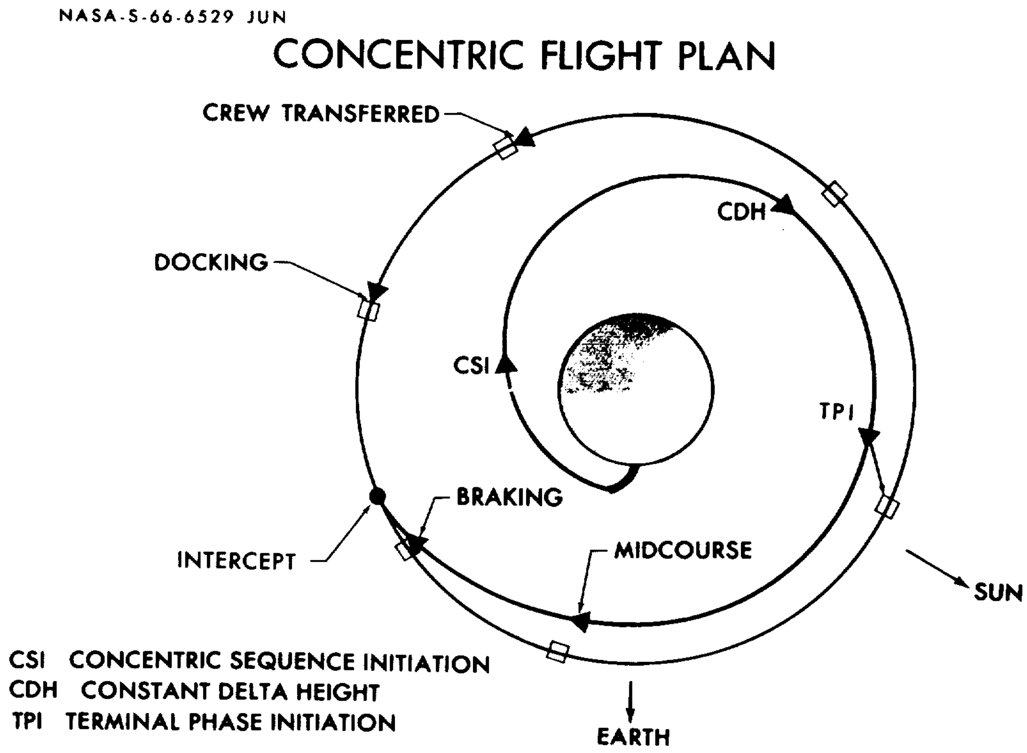

NASA-S-66-6529 JUN
CONCENTRIC FLIGHT PLAN
CREW TRANSFERRED
DOCKING
CDH
CSI
TPI
BRAKING
INTERCEPT
MIDCOURSE
SUN
EARTH
CSI CONCENTRIC SEQUENCE INITIATION
CDH CONSTANT DELTA HEIGHT
TPI TERMINAL PHASE INITIATION

Fig. 7

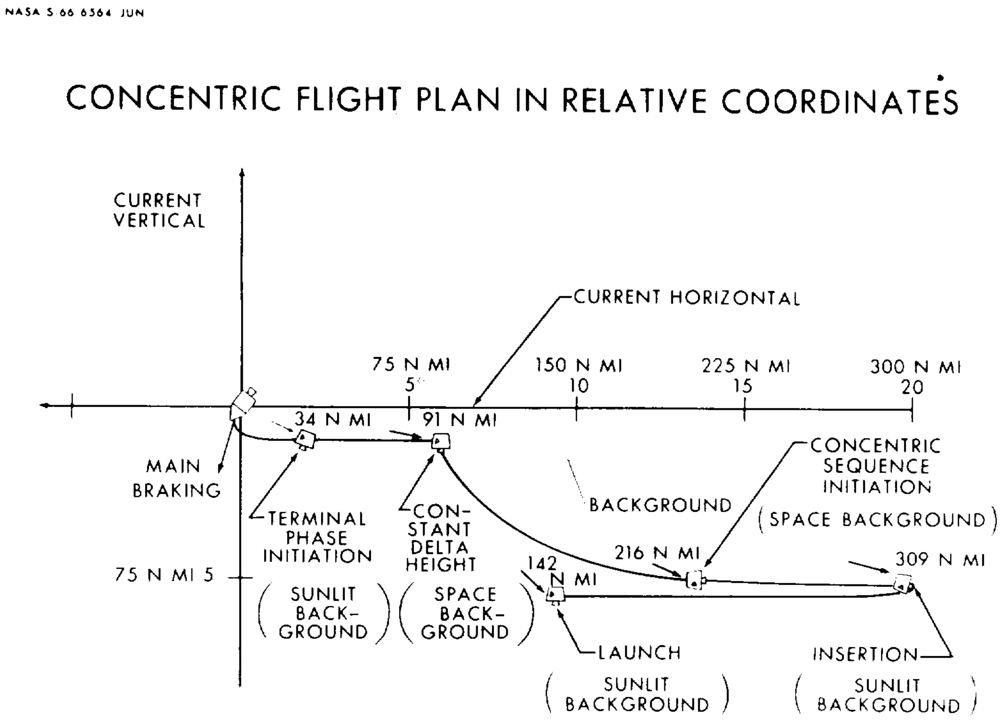

NASA S 66 6364 JUN
CONCENTRIC FLIGHT PLAN IN RELATIVE COORDINATES
CURRENT VERTICAL
CURRENT HORIZONTAL
75 N MI 5
150 N MI 10
225 N MI 15
300 N MI 20
34 N MI
91 N MI
MAIN BRAKING
TERMINAL PHASE INITIATION
(SUNLIT BACKGROUND)
CONSTANT DELTA HEIGHT
(SPACE BACKGROUND)
BACKGROUND
CONCENTRIC SEQUENCE INITIATION
(SPACE BACKGROUND)
75 N MI 5
142 N MI
216 N MI
309 N MI
LAUNCH
(SUNLIT BACKGROUND)
INSERTION
(SUNLIT BACKGROUND)

Fig. 8

NASA-S-66-6536 JUN

UTILIZATION OF PROPULSION SYSTEMS

● LM			
● CONCENTRIC FLIGHT PLAN POWERED ASCENT	MAIN ENGINE RCS	MAIN ENGINE FOR POWER RCS NULLS THRUST OFFSET	APS FUEL
● CONCENTRIC SEQUENCE INITIATION	RCS	RCS TANKS FOR ULLAGE THEN APS FUEL	
● CONSTANT △H	RCS	RCS TANKS	
● TERMINAL PHASE INITIATION	RCS	RCS TANKS	
● TERMINAL THRUSTINGS	RCS	RCS TANKS	
● CSM			
● ALL MANEUVERS			
$V_G > 12$ FT/SEC		RCS ULLAGE THEN MAIN ENGINE	
$V_G < 12$ FT/SEC		RCS e g FINAL TERMINAL THRUSTING IS COMPLETELY CSM ACTIVE RENDEZVOUS	

Fig. 9

NASA-S-66-6521 JUN

TIME EMERGENCY RENDEZVOUS

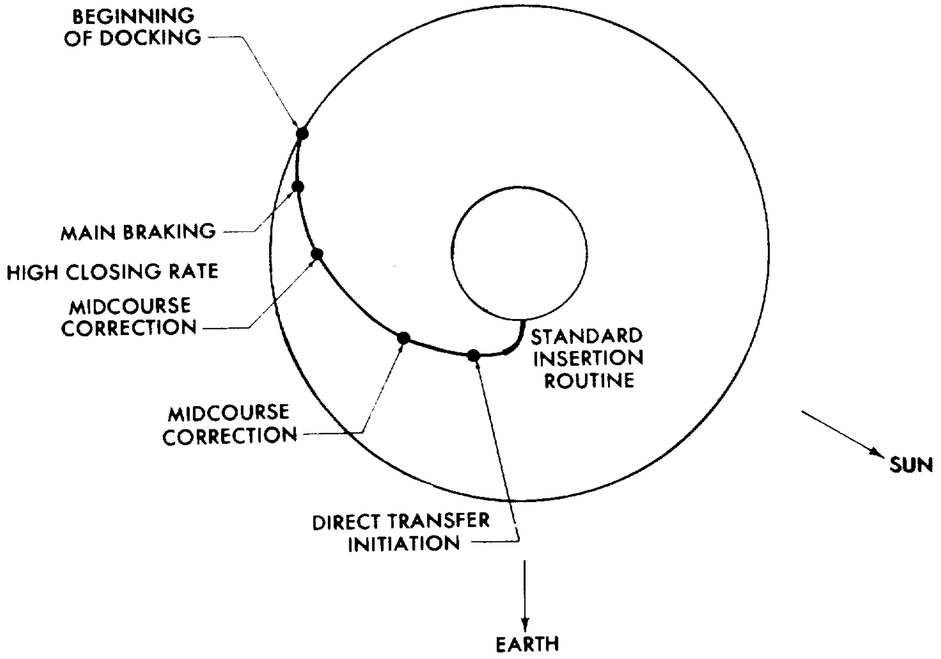

BEGINNING OF DOCKING
MAIN BRAKING
HIGH CLOSING RATE
MIDCOURSE CORRECTION
STANDARD INSERTION ROUTINE
MIDCOURSE CORRECTION
SUN
DIRECT TRANSFER INITIATION
EARTH

Fig. 10

NASA-S-66-6546 JUN

OPTICAL TRACKING ASSUMPTIONS

<table>
<tr><th></th><th colspan="3">ACCEPTABLE CSM SEXTANT VISUAL CONDITIONS</th></tr>
<tr><td>LOS HAS RADIAL COMPONENT OUTWARDS</td><td colspan="3">ACCEPTABLE IF ANGLE BETWEEN SUN DIRECTION AND LOS IS GREATER THAN 20°</td></tr>
<tr><td rowspan="5">LOS HAS RADIAL COMPONENT INWARDS</td><td colspan="3">LUNAR BACKGROUND</td></tr>
<tr><td>RANGE (N MI)</td><td>TARGET</td><td>LUNAR SURFACE LIGHT</td></tr>
<tr><td>250</td><td>FLASHING LIGHT</td><td>EARTH REFLECTED</td></tr>
<tr><td>40</td><td>SUN ILLUMINATED LM</td><td>SUNLIT</td></tr>
<tr><td>500 N MI</td><td>FLASHING LIGHT</td><td>DARK</td></tr>
</table>

Fig. 11

NASA-S-66-6567 JUN

EXPLANATION OF LINE-OF-SIGHT CONVENTION

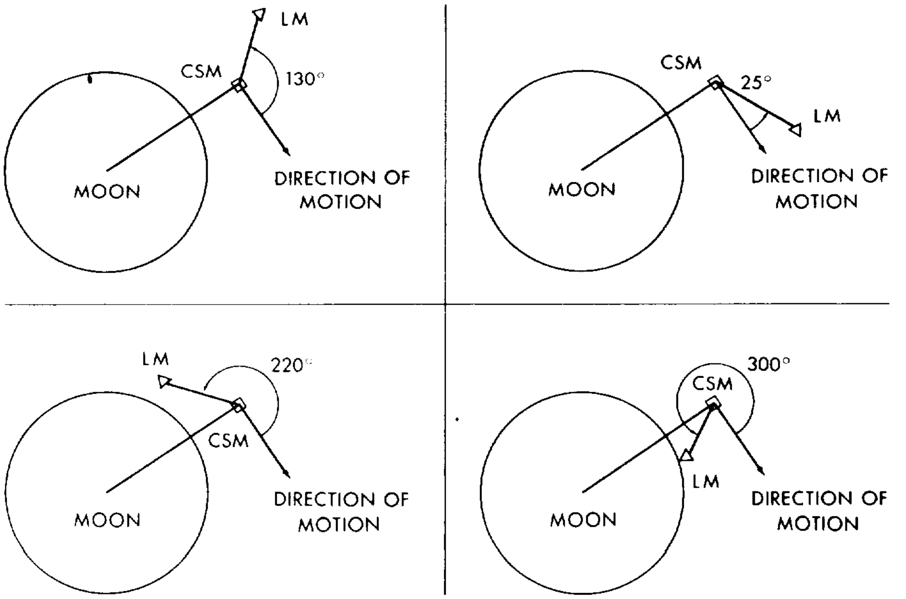

LM
CSM
130°
MOON
DIRECTION OF MOTION
CSM
25°
LM
MOON
DIRECTION OF MOTION
LM
220°
CSM
MOON
DIRECTION OF MOTION
300°
CSM
LM
MOON
DIRECTION OF MOTION

Fig. 12

NASA-S-66-6121 JUN

CSM SEXTANT TRACKING LEM

NOMINAL FLIGHT PLAN

LANDING 0° SUN ELEVATION 18°, CONCENTRIC ALTITUDE 65 N. MI.

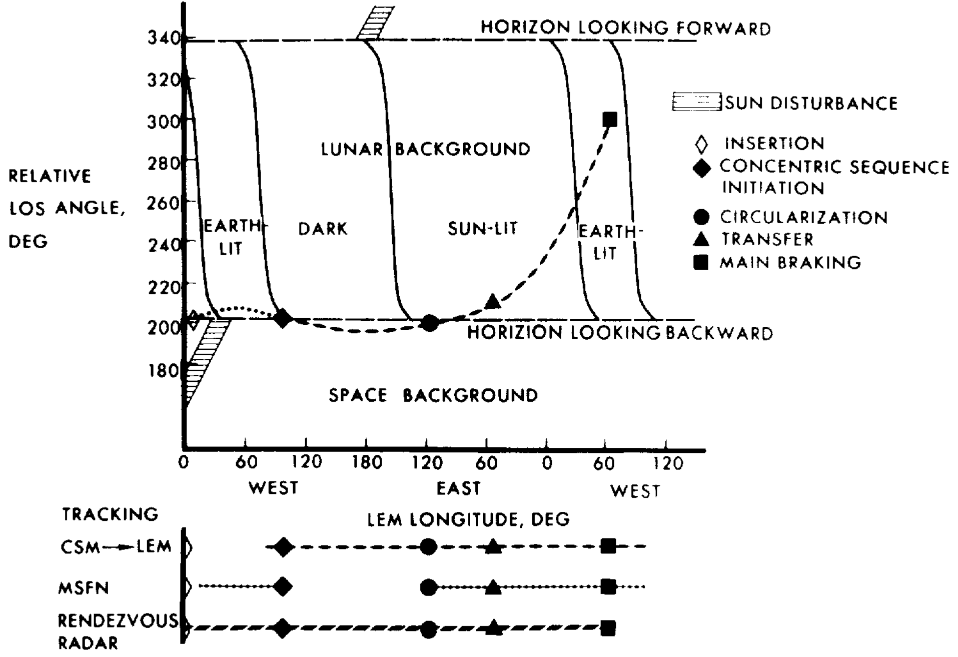


Fig. 1[illegible]

NASA-S-66-6124 JUN

CSM SEXTANT TRACKING LEM

ABORT FROM INITIATION OF POWERED DESCENT

LANDING 0°, SUN ELEVATION 30°, CONCENTRIC ALTITUDE 130 N. MI

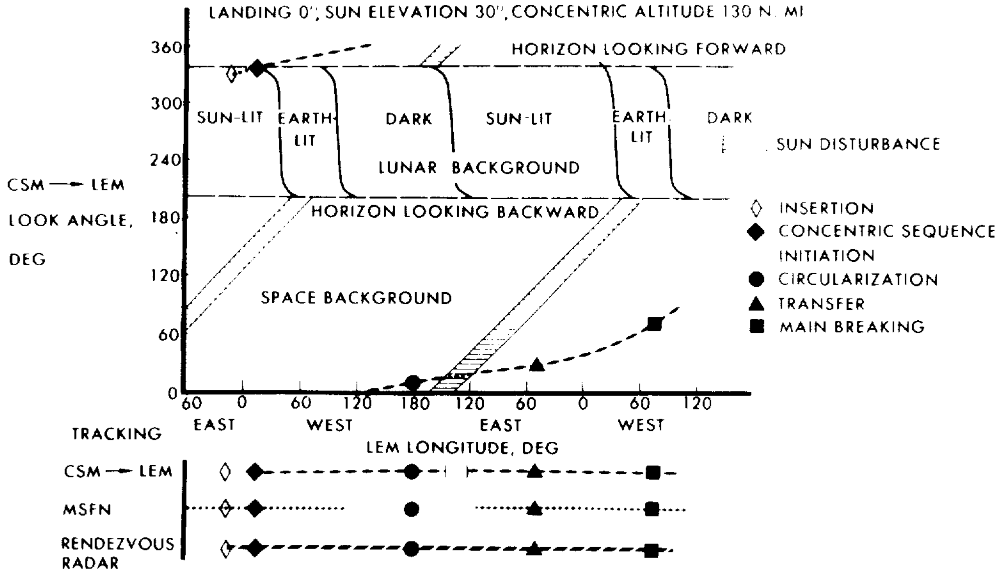


Fig. 14

TRACKING SUMMARY

RENDEZVOUS RADAR, CSM SEXTANT, MSFN

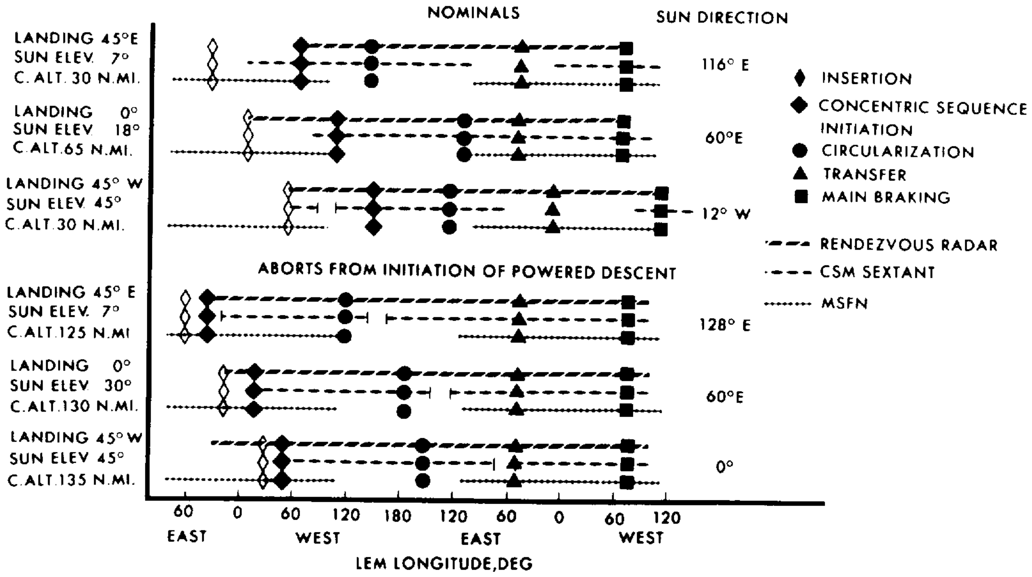


Fig. 15

NASA-S-66-6549 JUN

MSFN ERROR ANALYSIS ASSUMPTIONS

- MSFN TRACKING IN 3 - WAY DOPPLER MODE WITH 1 MASTER STATION AND 2 SLAVE STATIONS
- NOISE AND BIASES ON MEASUREMENTS:

	NOISE (1σ)	BIAS (1σ)
2 - WAY DOPPLER	0.1 FPS	0.07 FPS
3 - WAY DOPPLER	0.1 FPS	0.2 FPS

- STATION LOCATION BIASES AS GIVEN IN APOLLO NAVIGATION WORKING GROUP TECHNICAL REPORT NO. 65 - AN - 1.0, FEB. 5, 1965
- BIAS ON GRAVITATIONAL CONSTANT OF MOON,

$$\sigma_{\mu_M} = 6 \times 10^9 \text{ FT}^3/\text{SEC}^2,$$

- FROM JPL TECHNICAL REPORT NO. 32 - 694, DEC. 15, 1964
- ORBIT DETERMINATION SOLVES FOR
  - 3 COMPONENTS OF VELOCITY
  - 3 COMPONENTS OF POSITION
  - BIAS ON 3 - WAY DOPPLER

Fig. 16

NASA S 66-6563 JUN

MSFN TRACKING UNCERTAINTIES (3σ)
CONCENTRIC FLIGHT PLAN FROM 45° W LAUNCH SITE

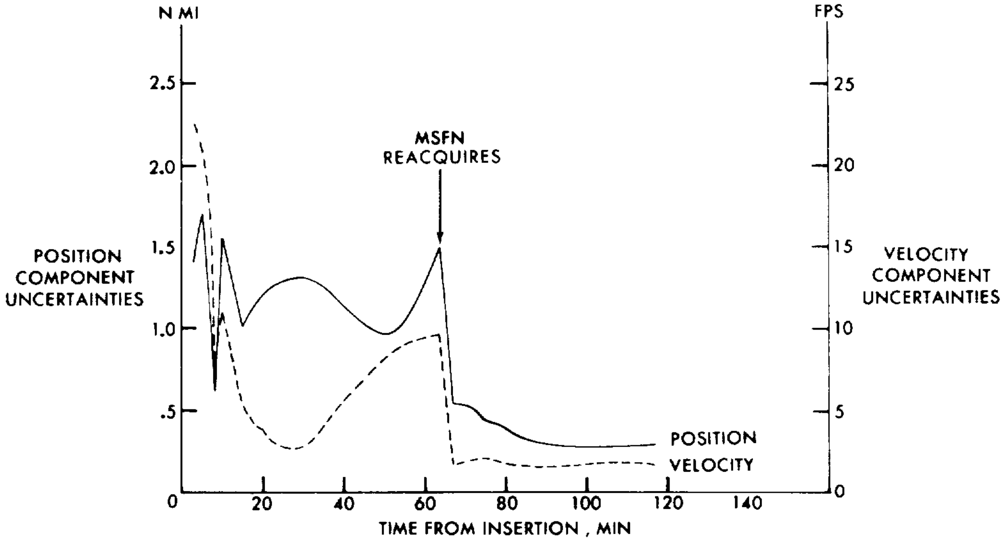


Fig. 17

NASA-S-66-6541 JUN

(3σ) MSFN ERRORS AT INTERCEPT
CONCENTRIC FLIGHT PLAN FROM 45° W LAUNCH SITE

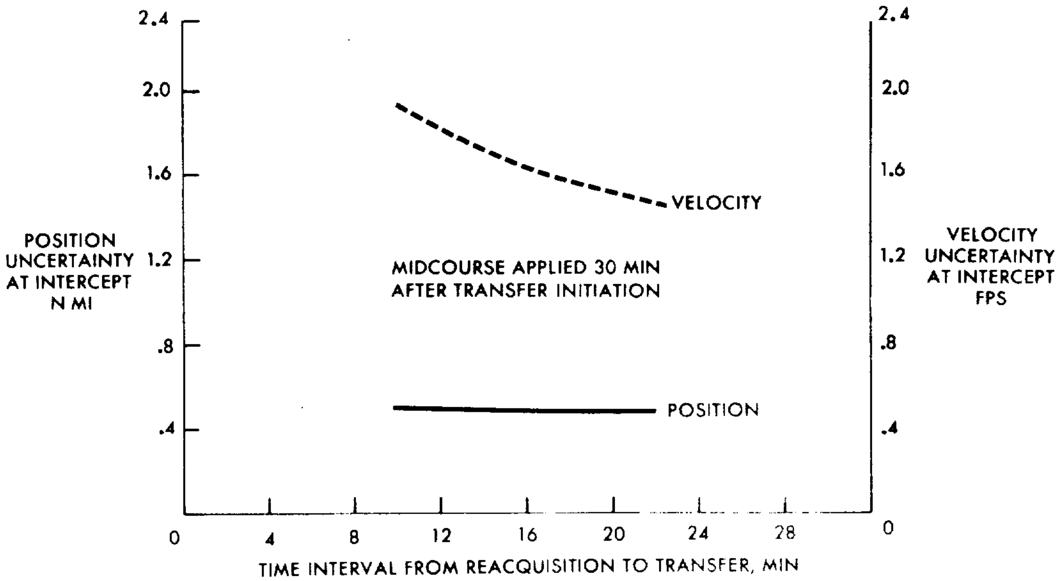


Fig. [illegible]

NASA-S-66-6554 JUN

MAIN COMPONENTS OF ONBOARD RENDEZVOUS PROGRAMS

PARALLEL DETERMINATION ON GROUND

- STANDARD INSERTION
  - LAUNCH; ABORT FROM POWERED DESCENT
- DIRECT TRANSFER
- CONCENTRIC SEQUENCE
  - CSI ⟶ CDH ⟶ TPI ⟶ TERMINAL MANEUVER
  - DETERMINED AS A WHOLE BUT CREWMAN GIVES PROCEED THROUGH DSKY FOR EACH MANEUVER
- EXTERNAL ΔV

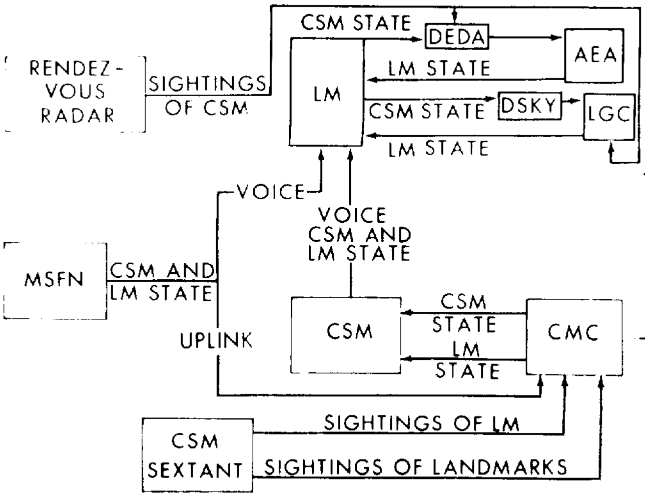
Fig. 19

NASA S 66 6571 JUN

PROVISION OF G AND N INFORMATION FOR THE OPERATION

NAVIGATION

DECISION LOGIC FOR RENDEZVOUS


CSM STATE
DEDA
AEA
LM STATE
RENDEZ-VOUS RADAR
SIGHTINGS OF CSM
LM
CSM STATE
DSKY
LGC
LM STATE
VOICE
VOICE CSM AND LM STATE
MSFN
CSM AND LM STATE
UPLINK
CSM
CSM STATE
LM STATE
CMC
CSM SEXTANT
SIGHTINGS OF LM
SIGHTINGS OF LANDMARKS

COMPLETE ONBOARD CAPABILITY (TERMINAL THRUSTINGS ARE MANUAL) UNLESS TIME OR V LIMITS ARE VIOLATED—THEN RELIANCE ON GROUND

COMPLETE ONBOARD CAPABILITY UNLESS TIME OR ΔV LIMITS ARE VIOLATED—THEN RELIANCE ON GROUND

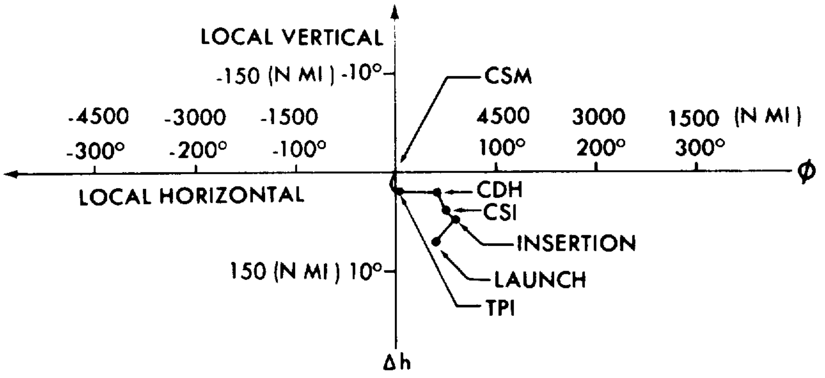
DIRECT TRANSFER CAPABILITY ONLY RELIANCE ON GROUND FOR DIRECTION AS REGARDS INITIATION MULTIPLE ORBITS, ETC

Fig. 20

NASA-S-66-6578 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (60°)


LOCAL VERTICAL
-150 (N MI) -10°
CSM
-4500 -3000 -1500
-300° -200° -100°
4500 3000 1500 (N MI)
100° 200° 300°
Ø
LOCAL HORIZONTAL
CDH
CSI
INSERTION
150 (N MI) 10°
LAUNCH
TPI
Δh

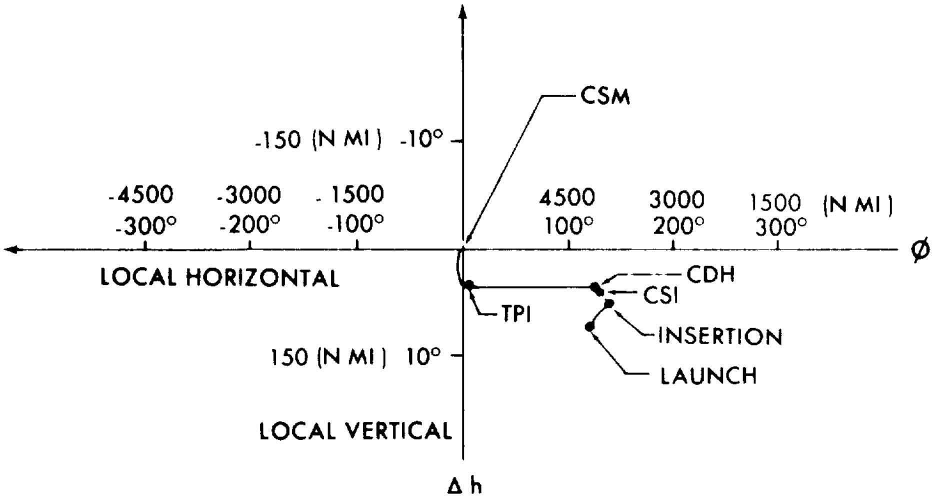
RELATIVE FLIGHT PLAN

Fig. 71

NASA-S-66-6577 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (140°)


CSM
-150 (N MI) -10°
-4500 -3000 - 1500
-300° -200° -100°
4500 3000 1500 (N MI)
100° 200° 300°
Ø
LOCAL HORIZONTAL
CDH
CSI
TPI
INSERTION
150 (N MI) 10°
LAUNCH
LOCAL VERTICAL
Δh

RELATIVE FLIGHT PLAN

Fig. [illegible]

NASA-S-66-6557 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (-50°)

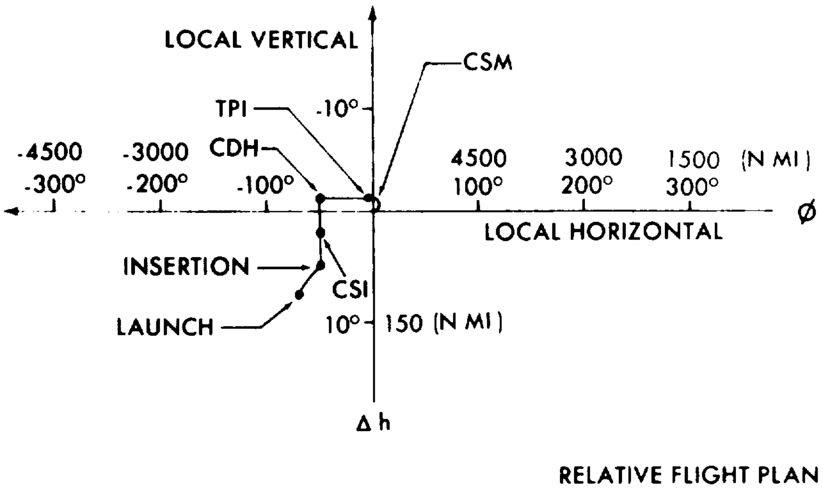

LOCAL VERTICAL
CSM
TPI
-10°
-4500
-3000
CDH
4500
3000
1500 (N MI)
-300°
-200°
-100°
100°
200°
300°
ϕ
LOCAL HORIZONTAL
INSERTION
CSI
LAUNCH
10°
150 (N MI)
Δh
RELATIVE FLIGHT PLAN

Fig. 23

NASA-S-66-6576 JUN

LM ACTIVE CONTINGENCY RENDEZVOUS

PHASING AT INSERTION (-160°)

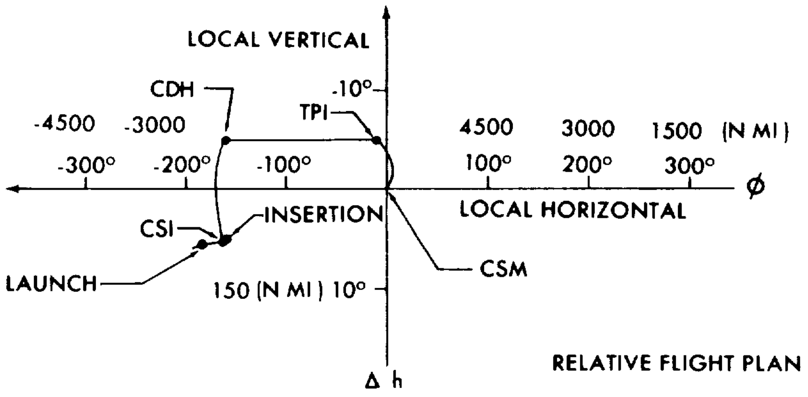
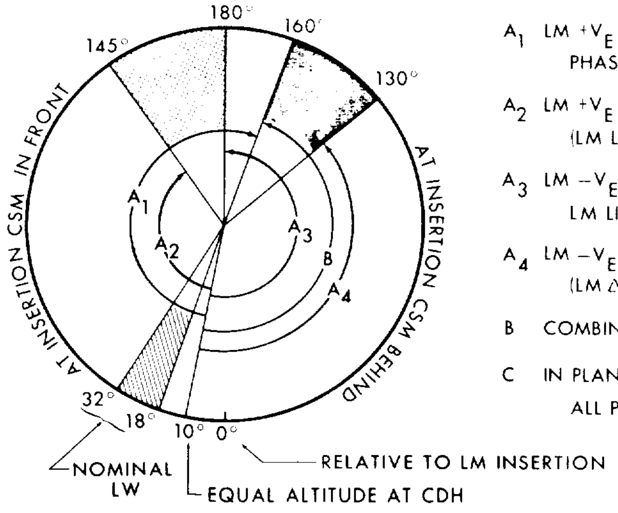

LOCAL VERTICAL
CDH
-10°
TPI
-4500
-3000
4500
3000
1500 (N MI)
-300°
-200°
-100°
100°
200°
300°
ϕ
INSERTION
LOCAL HORIZONTAL
CSI
CSM
LAUNCH
150 (N MI) 10°
Δh
RELATIVE FLIGHT PLAN

Fig. 24

NASA-S-66-6569 JUN

RENDEZVOUS CATEGORIES
ACCORDING TO PHASING


180°
160°
145°
130°
AT INSERTION CSM IN FRONT
AT INSERTION CSM BEHIND
A_1
A_2
A_3
B
A_4
32°
18°
10°
0°
NOMINAL LW
EQUAL ALTITUDE AT CDH
RELATIVE TO LM INSERTION

A_1 LM $+V_E$ CATCHUP WITH TRANSFER FROM PHASING ALTITUDE (LM LIFETIME ONLY)

A_2 LM $+V_E$ CATCHUP AND $\Delta H = 15$ N MI (LM LIFETIME ONLY 11.3 HR)

A_3 LM $-V_E$ CATCHUP USING TOTAL ΔV AND LM LIFETIME (INPLANE)

A_4 LM $-V_E$ CATCHUP 0.5° WEDGE ANGLE (LM ΔV TOTAL AND LIFETIME)

B COMBINED MANEUVERS WITH CSM

C IN PLANE CSM RESCUE POSSIBLE FROM ALL PHASE ANGLES

Fig. 9

NASA-S-66-6562 JUN

RENDEZVOUS LOGIC

CONSTRAINT:

- ASCENT STAGE LIFETIME
- LM ΔV AVAILABLE
- CLEAR PERICYNTHION TRANSFERS FOR BOTH LM AND CSM

DESIRABLE CONDITION:

- SATISFACTORY LIGHTING DURING TERMINAL PHASE

A) LM ACTIVE

1ST PREFERENCE

(1) FIRST ORBIT
  1.1 NON TIME CRITICAL; CONCENTRIC
  1.2 TIME CRITICAL EQUIVALENT DIRECT ASCENT
(2) MULTIPLE ORBITS

TERMINAL THRUSTINGS BY LM

B) COMBINED MANEUVERS

2ND PREFERENCE

MINIMUM AMOUNT OF ENERGY EXPENDED BY CSM, MULTIPLE ORBITS BY LM IF NECESSARY

TERMINAL THRUSTINGS BY LM

C) CSM ACTIVE ONLY

3RD PREFERENCE

(MAXIMUM CAPABILITY OF LM PROPULSION WAS TO INSERT IN MINIMUM SAFE ORBIT)

TERMINAL THRUSTINGS BY CSM

Fig. 10

NASA-S-66-6545 JUN

SUMMARY

- PLANS FOR
  - EVERY PHASE; ALL PRACTICAL OUT OF PLANE CONDITIONS
- NORMAL CASES
  - CONCENTRIC- STANDARD PATTERN OF PROCEDURES
- PLAN FOR TIME CRITICAL SITUATIONS
- REDUNDANCY ON THE LM TO RELIEVE THE CSM
- NON TIME CRITICAL
  - JOINT DECISIONS DURING INTERCEPT TRAJECTORY

Fig. 27

TRANSEARTH INJECTION THROUGH REENTRY

By

M. P. Frank

TRANSEARTH INJECTION THROUGH REENTRY

In this section we will finish off the mission planning considerations and the affects of constraints on trajectory shaping of the nominal mission. All we have to discuss on this topic are the problems of getting the spacecraft back from the moon through the reentry corridor to the recovery area. This section is divided into two phases, transearth and reentry.

Transearth

The geometric restrictions on transearth injection are somewhat similar to those of the translunar injection. However, proper allowances must be made for the fact that the spacecraft trajectory relative to the moon is hyperbolic instead of elliptic and the moon is orbiting about the earth. In the transearth case the target body is relatively stationary and the spacecraft is leaving the body which is in orbit. The spacecraft must break out of the lunar gravitational sphere and fall back to earth. The velocity relative to the moon must be increased in order to escape but the spacecraft inertial velocity must be decreased in order to return to earth.

This is illustrated in Fig. 1. At a distance from the earth equal to the distance to the moon's sphere of influence the earth relative velocity vector required to obtain a given perigee altitude is illustrated by the vector V_{re}. In this figure two cases are shown; a high energy short time trajectory and a low energy long transit time trajectory. There is a continuum of safe return trajectories between there two extremes. The moon's orbital velocity is shown as the vector V_m. The spacecraft must obtain a velocity vector relative to the moon V_{rm} that results in a velocity relative to the earth V_{re}. The minimum velocity relative to the moon occurs when the velocity relative to the earth is a minimum or at apogee of the earth return trajectory. This is not necessarily the longest flight time because the spacecraft could leave the moon on such a trajectory as to have a positive flight-path angle relative to the earth. This type of trajectory is of no interest, however, because the increase of transearth flight time does not allow a decrease in the injection velocity requirements. The conclusions that can be drawn from this figure are that faster return times require larger exit velocities and hence larger transearth injection velocities. It can also be seen that the lower injection velocities require a more nearly retrograde motion upon leaving the moon's sphere of influence.

Figure 2 illustrates the trajectory geometry inside the moon - sphere of influence relative to the exit conditions. In this figure the moon's sphere of influence is approximated by the large diameter circle. The velocity of the moon relative to the earth is given by the vector V_m. Since the sphere of influence moves with the moon any point on this sphere would also have the velocity relative to the earth of V_m. The trajectories inside this sphere are hyperbolic relative to the moon and the energy of these trajectories determines the magnitude of the exit velocity V_{rm}. The direction of this exit velocity is primarily controlled by the exit position which in turn is determined by the longitude at which the injection is performed. To obtain a more retrograde direction of the exit velocity, the injection must occur around on the leading edge of the moon as illustrated by the upper trajectory in Figure 2. To obtain a high energy, short time, earth return trajectory the injection maneuver must be performed on the trailing edge of the moon. This results in exit velocities directed more toward the earth.

It can be seen that once a desired transearth flight time has been selected the exit velocity requirement can be determined and in turn these exit velocity requirements will specify the energy of the transearth injection maneuver and also the location of the transearth injection maneuver. The shaded area in Figure 3 shows the region of longitudes where the transearth injection could be performed. It extends from about 140 degrees west longitude to about 140 degrees east longitude on the far side of the moon.

The affect of the transearth transit time on the injection velocity requirements can be obtained from Figure 4. In this figure the transearth injection ΔV is shown as a function of transit time for two different moon positions in its orbit. The trend of generally decreasing injection velocity requirements with increasing transit time is clearly illustrated. The fact that these curves cross over indicates that there are other factors at work in determining the actual injection velocity required. However, these other factors do not change the basic conclusions.

Now, lets consider how the required transearth flight time is determined. Figure [illegible] illustrates the transearth trajectory drawn in the moon orbit plane. The return perigee location is shown relative to the moon's antipode. In this case the moon's antipode is drawn at the time of the transearth injection. This relative angle between the return perigee and the antipode has very little variation with the return transit time. Therefore, the approximate inertial position of the return perigee is a function only of the moon's position. The landing location relative to the return perigee also has very little variation so that in affect the inertial position of the landing is known well in advance of the actual transearth injection.

ıne recovery forces, of course, are fixed to the earth and hence are rotating around with the earth. This leads to the rather interesting situation in which the inertial location of the landing position is well known. However, the inertial position of the recovery forces is highly dependent on the time at which landing occurs. The determination of the transearth transit time is based on the location of the recovery forces. That is, the reentry and landing must occur when the recovery forces are in the proper position. This occurs once every 24 hours, and within any 24 hour range of return transit times one time can be found which allows rendezvous with the recovery forces.

Entry Phase

For the lunar mission the entry corridor is defined by the variation in flight-path angle at the entry interface altitude of 400,000 feet. Strictly speaking, this allowable variation is a function of the velocity at entry, but there is such a small variation for a nominal lunar mission that the velocity effect is generally omitted. However, for abort returns to earth, this is not the case, and the entry corridor is defined as a function of entry velocity.

The maximum entry angle is defined by the maximum allowable aerodynamic decelleration. Aerodynamic loads encountered during entry increase rapidly with increasingly negative flight-path angles. The high-"g" side of the corridor is called the undershoot boundary. The minimum entry angle, called the overshoot boundary, is defined by the Command Module's capability to prevent an uncontrolled skipout of the atmosphere. In addition to being a function of entry velocity, these corridor boundaries are strongly dependent of the L/D ratio of the entry vehicle.

Figure 6 shows the entry corridor for the nominal L/D of .34 of the Command Module. In this figure, entry flight-path angle is plotted as a function of range from the entry point to the landing point in order to combine the corridor and maneuvering capability information. The skipout limit is shown here to be a flight-path angle slightly less than 5 degrees. In other words, any entry conditions at shallower angles than this would result in an uncontrolled skip. The 10 g undershoot limit is shown to be about 7.3 degrees and any flight-path angle steeper than this would result in the g loads exceeding 10. The emergency limit is defined as a limit below which the aerodynamic load factor would exceed 20 g. This 20 represents the structural design limit of the spacecraft.

The minimum guided range represents the limit to which the reentry range can be controlled with a guidance system. For the overshoot boundary this corresponds to about 1500 miles. The maximum range is limited by the lifting capability of the vehicle. At the steep entry angles the spacecraft could not fly further than this range with maximum positive lift. The maximum range is arbitrarily cut off at 2500 n.m. to comply with the limitations imposed by the heat shield. The heat shield is designed to tolerate a 3500 n.m. reentry. There is an entry monitoring system onboard the spacecraft which gives warning when excessive skip is to be encountered. However, the tolerance on this entry monitoring system is about 1000 miles. This is made large so that it will not unnecessarily take over a trajectory or give warning that an excessive skip is going to occur. To allow for this 1000 mile tolerance the mission can not be planned with reentry ranges exceeding 2500 miles. The inplane maneuvering capability that can be used which would be independent of the entry ocrridor position is given by the 1500-mile limit and the 2500-mile limit so that ± 500 miles of down range maneuver capability is available. The nominal aim point represents the conditions that will be targeted to for the transearth trajectory. That is, the return trajectory will be planned to have a flight-path angle at entry of 6.2 degrees and to have the entry point located some 2000 miles away from the landing point. This gives maximum maneuvering capability and allows maximum tolerance of dispersions both in flight-path angle and in range at entry. The primary purpose of the maneuvering capability is to allow a change in the landing site after the transearth injection has been performed. If bad weather were to develop in the area of the recovery forces such that a landing there was undesirable, the spacecraft would have the capability of going 500 n.m. to either side of this position. Figure 7 shows the total maneuver footprint, both the downrange and crossrange plotted on a map of the Pacific Ocean hemisphere. The reentry point is some 2000 miles away from the nominal touchdown position. The crossrange capability is about 440 miles at the base and some 660 miles at the toe of this footprint.

Now let's examine the considerations involved in locating the recovery areas. Figure 8 shows the inplane geometry of the location of significant points of the reentry trajectory. The angular relations shown between the entry point, the perigee of the return trajectory and the landing point have only a slight variation. The location of the return perigee relative to the antipode also has only a slight variation. The result is that the landing point on the earth's surface will be very near the moon's antipode. All of the earth return trajectories will pass through this antipode regardless of the return inclination. Because of the relatively small angle between the antipode and the landing position, very little latitudinal control of the landing can be obtain by varying the inclination of the earth return trajectory. This is illustrated in Figure 9. In this figure, two typical earth return trajectories are shown. These two inclinations would be obtained by performing pland changes or performing an azimuth change at the transearth injection maneuver. This plan results in a different return inclination for the two trajectories. However, since the antipode is in effect a node through which all of these return trajectories pass and since the landing is fairly close to this node, it can be seen in Figure 9 that only a very small amount of latitude variation can be obtained. The amount of latitude control is further restricted by the fact that the return inclination is limited to 40 degrees or less relative to the earth's equator. The landing position control that is available then can be summarized as one in which very fine control is available for the landing latitude. The longitude can be controlled exactly by merely varying the transit time of the transearth trajectory. However, the latitude will be a function of the moon's declination at the time of the transearth injection maneuver, and only small variations about this latitude are available through the means of changing the return trajectory inclination. These landing point control characteristics have led to the definition of the recovery zones shown in Figure 10.

There are two zones in which recovery from the lunar landing mission may occur. The northern zone extends from a latitude of 35 degrees north to 35 degrees south along a longitude of approximately 160 degrees west. The southern zone extends from five degrees south latitude to 35 degrees south along a longitude of approximately 167 degrees west. The northern zone would be supported by recovery forces staged from Hawaii. The staging base for the southern recovery zone would be Pago Pago in the Samoa Islands. The northern zone extends into the south latitude because of the preference to stage the recovery forces from the Hawaiian vase. The shape of these recovery zones reflects the landing area control capability of the return trajectory. The zones need not extend over a large range of longitudes because the longitude can be controlled precisely by the variation in return time. They do extend over a wide range of longitudes because the longitude can be controlled precisely by the variation in return time. They do extend over a wide range in latitudes since very little latitude control of the landing point is available. The latitude range of these recovery areas is a function of the maximum northern and southern declinations of the moon during any given month.

For any specific mission, of course, it will not be necessary to deploy forces to cover the entire recovery zones. The latitude of the antipode, or the moon's declination, will be confined to a narrow region of latitudes so that the recovery forces for any specific mission will be confined to a narrow region in one of these recovery zones.

Questions and Answers

TRANSEARTH INJECTION THROUGH ENTRY

Speaker: M. P. Frank

1. Dr. Von Braun - Can the same recovery ships be used throughout the launch window?

   ANSWER - Yes. The recovery ships are capable of achieving 30 knots, which is more than adequate to keep up with the recovery area changes.

2. Mr. Green - What is the longest time practical between getting back into lunar orbit and transearth injection?

   ANSWER - The CSM will have sufficient expendables aboard for approximately 14 days. Therefore, the longest time after LEM separation would be approximately 7 days. The trajectory changes required to return to earth during this period are not significant.

NASA-S-66-6465 JUN

EFFECT OF LUNAR SPHERE EXIT VELOCITY ON RETURN TRAJECTORIES

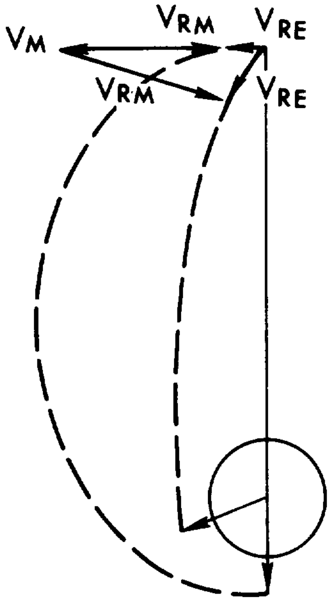

V_M
V_{RM}
V_{RE}
V_{RM}
V_{RE}

Figure 1

NASA-S-66-6413 JUN

LUNAR SPHERE EXIT HYPERBOLAS THAT RETURN TO EARTH

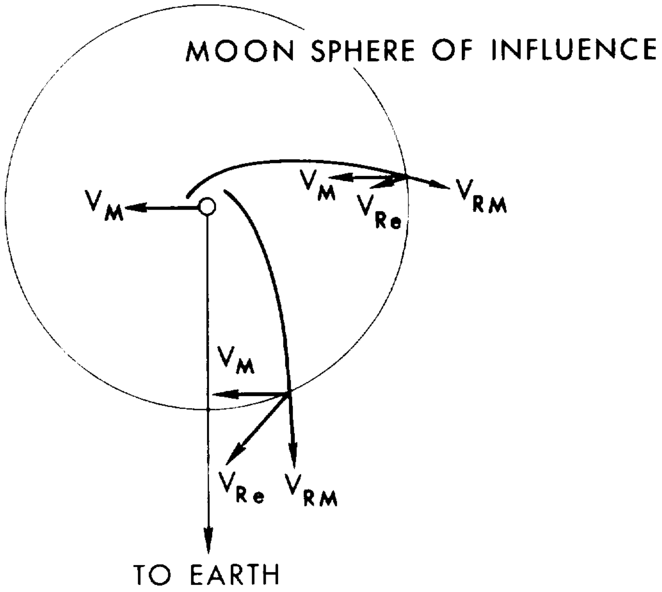

MOON SPHERE OF INFLUENCE
V_M
V_M
V_{Re}
V_{RM}
V_M
V_{Re}
V_{RM}
TO EARTH

Figure 2

NASA-S-66-5019 MAY 31

EFFECT OF TRANSEARTH FLIGHT TIME ON INJECTION POSITION

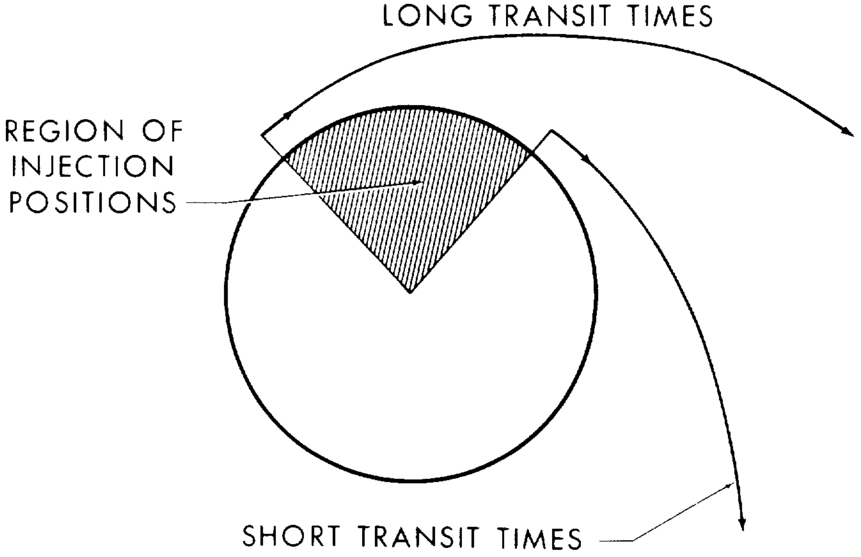
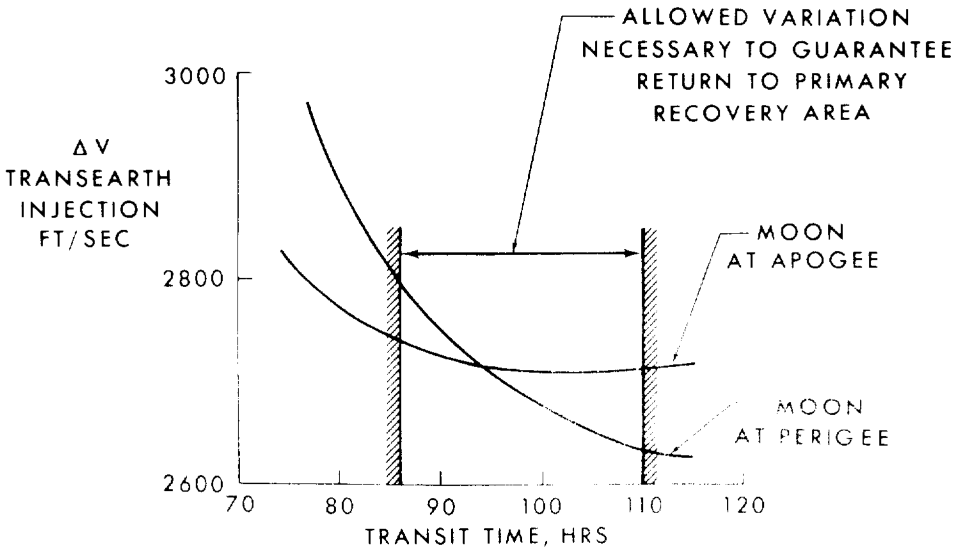

LONG TRANSIT TIMES
REGION OF INJECTION POSITIONS
SHORT TRANSIT TIMES

Figure [illegible]

NASA-S-66-5020 MAY 31

EFFECT OF TRANSEARTH FLIGHT TIME ON INJECTION VELOCITY REQUIREMENTS



NASA-S-66-6051 JUN

TRANSEARTH TRAJECTORY PERIGEE LOCATION

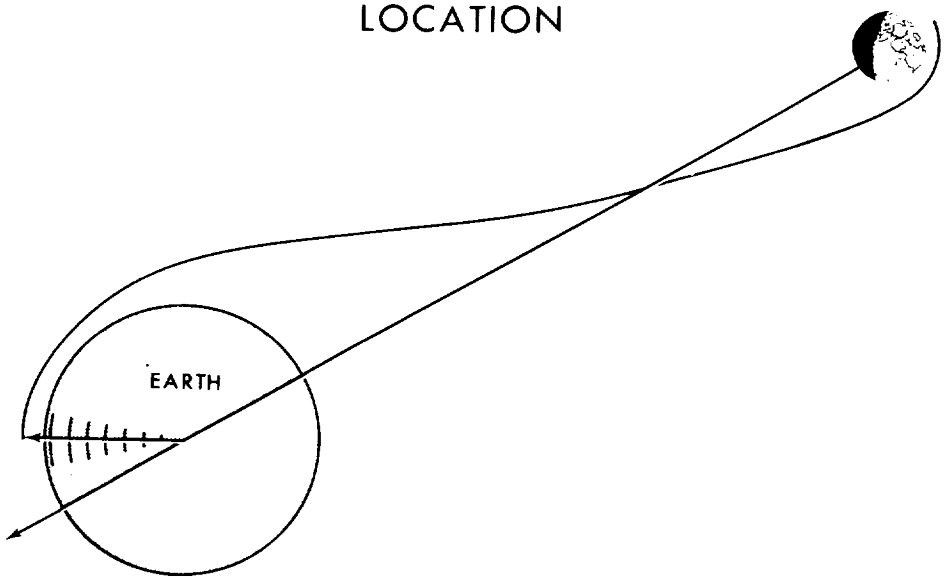


Figure 5

NASA-S-66-5009 MAY 31

ENTRY CORRIDOR

ENTRY SPEED = 36,000 FPS

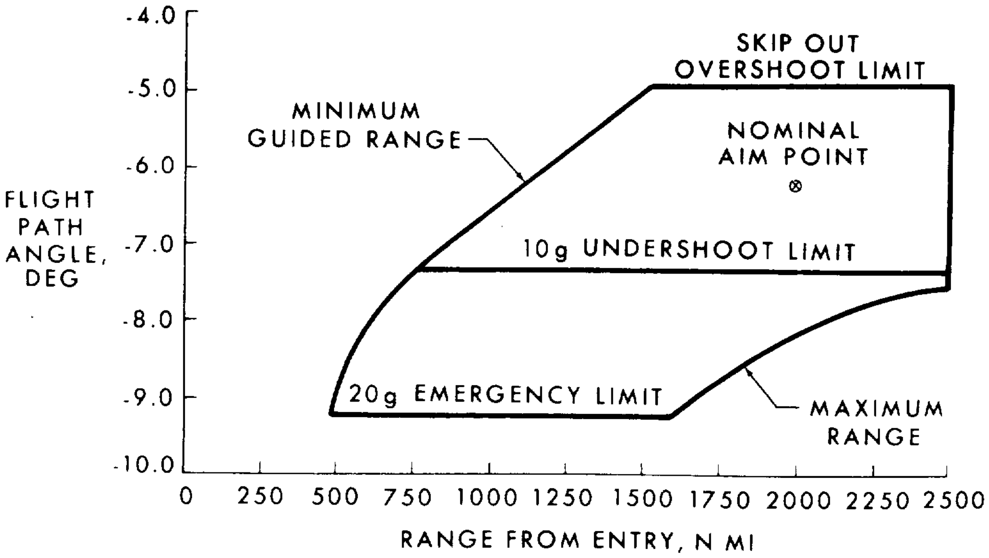
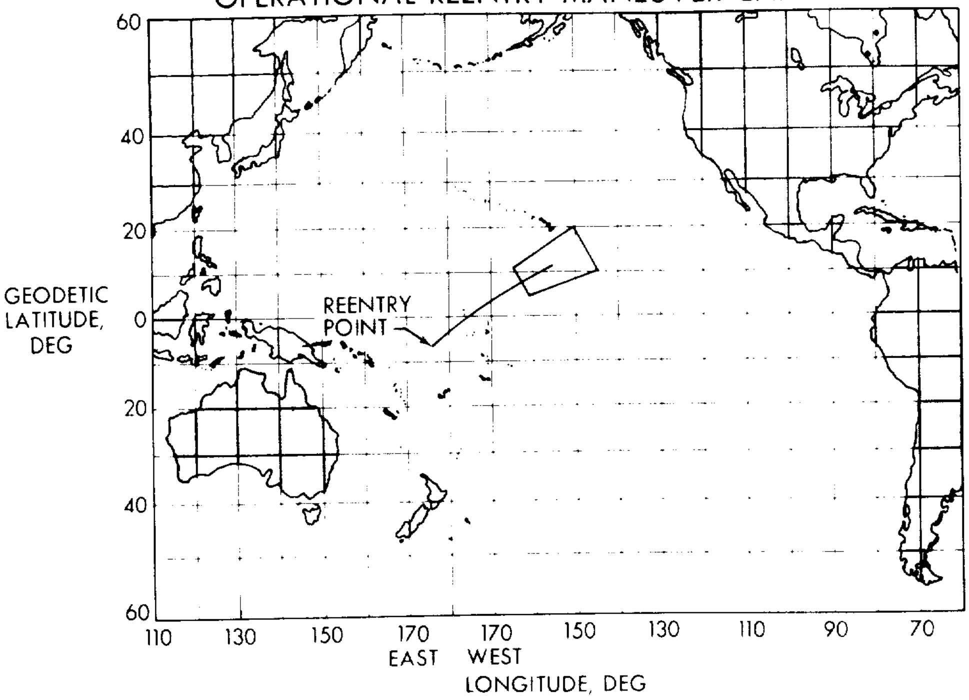


Figure 6

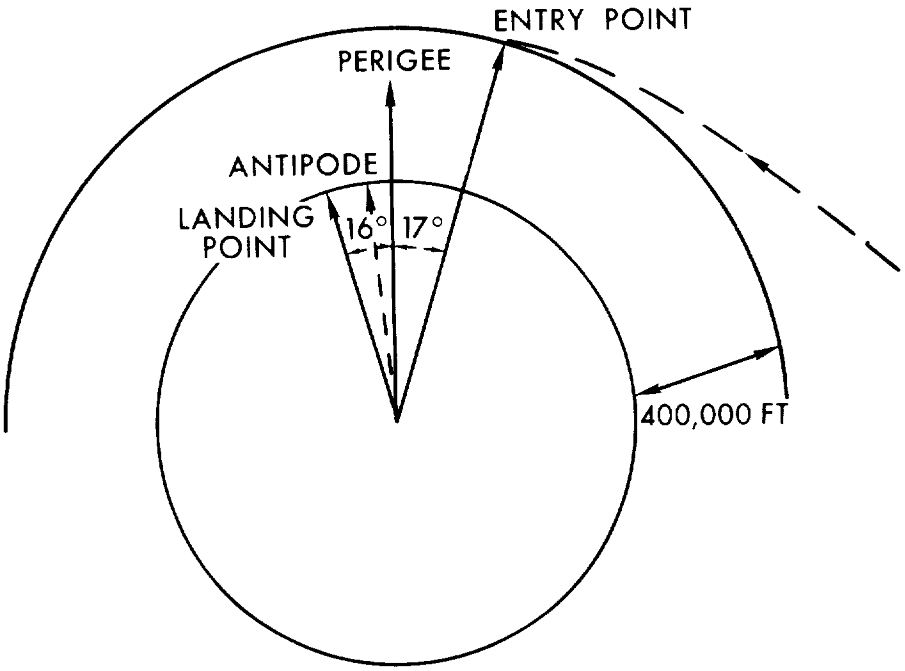
NASA S 66 6041 MAY

OPERATIONAL REENTRY MANEUVER CAPABILITY


60
40
20
0
20
40
60
GEODETIC LATITUDE, DEG
REENTRY POINT
110
130
150
170 EAST
170 WEST
150
130
110
90
70
LONGITUDE, DEG

NASA-S-66-6464 JUN

IN-PLANE TRAJECTORY GEOMETRY


ENTRY POINT
PERIGEE
ANTIPODE
LANDING POINT
16°
17°
400,000 FT

NASA-S-66-6392 JUN

EFFECT OF PLANE CHANGE ON CHANGING THE LANDING POINT

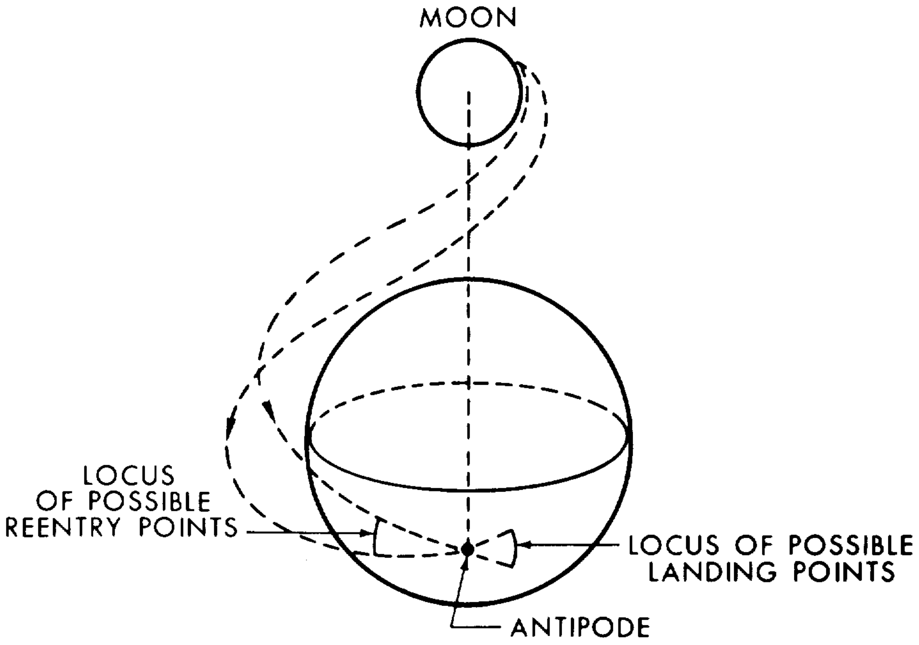

MOON
LOCUS OF POSSIBLE REENTRY POINTS
LOCUS OF POSSIBLE LANDING POINTS
ANTIPODE

Figure 9

NASA-S-66-5018 MAY 31

NOMINAL RECOVERY AREAS FOR MISSION PLANNING

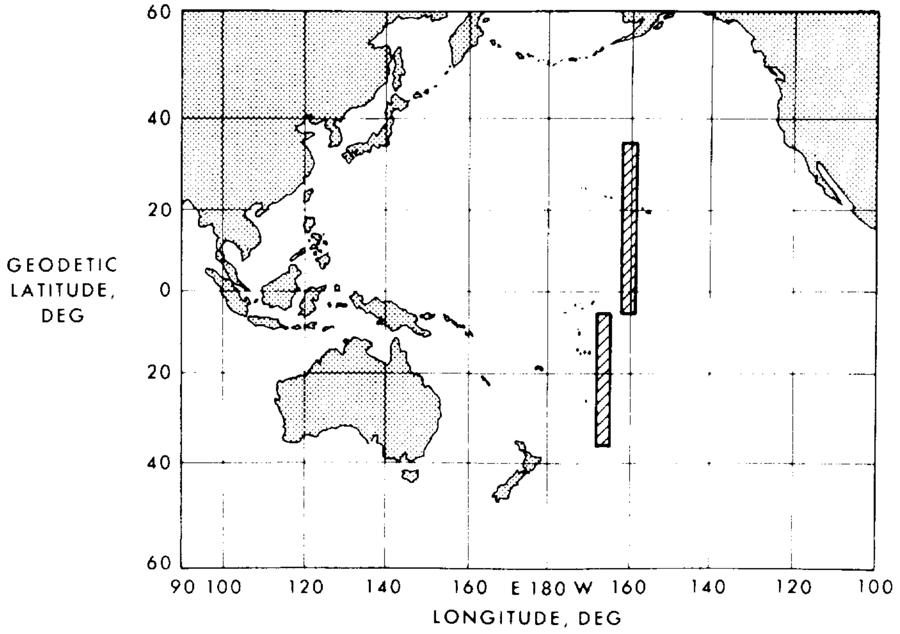

60
40
20
0
20
40
60
GEODETIC LATITUDE, DEG
90
100
120
140
160
E 180 W
160
140
120
100
LONGITUDE, DEG

Figure [illegible]

APOLLO EARTH RETURN ABORT CAPABILITIES

By

Ronald L. Berry

APOLLO EARTH RETURN ABORT CAPABILITIES

I.0 INTRODUCTION

It has always been a stated desire, as regards Project Apollo, to have continuous abort-to-earth capability throughout the entire lunar landing mission. This paper will examine the capability of the spacecraft to satisfy this objective through each mission phase of the lunar landing mission. The abort capability will be discussed primarily from a performance standpoint. In other words, examining whether or not the spacecraft has the necessary performance required for continuous abort capability throughout the mission. Portions of the mission where redundancy exists will be pointed out as well as portions of the mission which are critical or marginal as regards abort capability. The primary ground rules for this discussion are as follows:

a. The only objective considered as regards earth return aborts is the safe return of the crew. No alternate mission objectives are considered.

b. Only spacecraft abort capability after an abort decision has been made will be discussed. In other words, an assessment of the spacecraft capability to recognize an abort situation will not be included.

c. Only one, or at the most two, burn abort maneuvers will be considered since more sophisticated multiple burn maneuver sequences are not required to provide adequate abort capability.

The items which will be discussed for each mission phase are as follows:

a. The characteristics of the trajectories from which aborts could be required will be described. This includes any trajectory which might result from an underburn or an overburn during any of the major power flight maneuvers.

b. The basic abort modes or procedures will also be briefly described. Where possible, the mode considered prime will be pointed out along with the modes which are optional or backup. The computer logic required to provide all of these abort modes and procedures are currently being implemented into the real time ground computer system.

c. The capability of the spacecraft of actually performing abort manuevers will be discussed. Included here will be a description of the propulsion systems available, the ΔV available from each, and the ΔV required for abort.

d. The significant characteristics of abort trajectories will also be discussed. Included will be such things as orientation with respect to the earth and moon, relation between return time ΔV required, and delay time, etc.

The major constraints which shape all abort trajectories are as follows:

a. Reentry corridor - In other words, a velocity/flight-path angle relation at the beginning of the atmosphere (400,000 feet). For the purposes of this paper, only abort trajectories targeted to the center of the reentry corridor will be considered.

b. Maximum reentry speed (less than 36,000 fps) - This constraint is due to heat shield limitations.

c. Return inclination (less than 40°) - This constraint assures landings in temperate zones even in the presence of large reentry dispersions and also reduces heat shield requirements.

d. ΔV available - In other words, the ΔV required for an abort must be within the ΔV capability of the spacecraft. This value will probably be set at some amount less than the total ΔV available to allow a pad for midcourse and possible contingencies.

e. Return time - This constraint must always be less than the lifetime of the spacecraft systems such as power supply, life support, etc.

The mission phases considered in this discussion will be (1) launch-to-earth parking orbit, (2) earth parking orbit coast, (3) translunar injection, (4) translunar coast, (5) lunar orbit insertion, (6) lunar orbit coast, (7) transearth injection, and (8) transearth coast. Note that aborts during LM maneuvers are not considered since the abort would be to rendezvous.

2.0 ABORTS DURING LAUNCH PHASE

This phase consists of the launch vehicle three-stage burn from the pad-to-earth parking orbit insertion, a burn duration of approximately 700 seconds. An abort during this phase requires immediate action since the spacecraft trajectory is suborbital for the majority of the burn.

The propulsion systems available for abort during the launch phase are as follows:

a. LES - Launch escape system propulsion systems mounted on the tower atop the spacecraft.

b. SPS - The service propulsion system of the CSM.

c. S-IVB - The third stage of the launch vehicle.

Figure 1 shows a summary of the abort modes for the launch phase and through what region of the launch burn they apply. These modes are shown as a function of the burn time as well as the respective launch vehicle stage. The black bar designates the prime mode while the striped bar represents the optional or backup modes of abort.

Note that the first mode is the LES or launch escape system mode which is prime from the pad throughout the S-I stage and extending on for a few seconds into the S-II burn before LES jettison. As shown in figure 2, a LES abort consists of the LES propulsion system separating the Command Module from the stacked launch vehicle configuration and providing an adequate altitude and downrange translation. This is followed by the orientation of the Command Module with heat shield forward for reentry. Landing occurs in a continuous Atlantic recovery area along the flight azimuth up to a maximum downrange of approximately 400 nautical miles.

The next mode is the suborbital free-fall abort. This mode, as shown in the summary chart, begins where the LES is jettisoned and remains available until approximately halfway through the S-IVB burn. Note that this mode is prime through approximately the first half of the S-II burn and through half of the S-IVB burn. It is considered an optional mode through the second half of the S-II burn because of the availability of a contingency; orbit insertion with the S-IVB, which will be discussed below.

As shown in figure 3, the suborbital free-fall mode consists of CSM separation from the launch vehicle using Service Module RCS, a 10-second SPS burn to gain further separation from the launch vehicle, Service Module jettison, and Command Module orientation (heat shield forward) for reentry. Landing would be in the continuous Atlantic recovery area along the flight azimuth up to 3,200 nautical miles downrange.

An extension of the suborbital abort mode can be achieved by addition of another SPS burn for landing area control, as shown in the summary of abort modes in figure 1. This mode is available as an option during the second half of the S-IVB burn when the suborbital free-fall mode is no longer available.

Figure 4 shows how the suborbital mode with SPS landing control differs from the free-fall mode. Note that the procedure is identical except for an additional retrograde SPS burn. This burn is a variable length depending on the time of abort and causes landing to be at the end of the continuous Atlantic recovery area, approximately 3,200 nautical miles downrange.

The next mode of interest, as shown in figure 1, is the S-IVB contingency orbit insertion followed by an SPS deorbit to reentry. This mode, as shown, is available and prime for approximately the second half of the S-II burn. The reason that this mode is considered prime over the suborbital free-fall mode is that it allows landing to be pinpointed precisely to a given recovery force and would allow additional "thinking" time to consider alternate missions.

The next mode is similar in nature to the S-IVB contingency orbit insertion, except the SPS is used for both the COI and the deorbit burn. This mode, as shown in figure 1, is only possible during approximately the latter half of the S-IVB burn, but it is the prime mode for this time period.

Figure 5 shows the basic features of these latter two abort modes. As shown, insertion into earth parking orbit is completed by either the S-IVB or the SPS. After a certain coast time in earth parking orbit, during which CSM S-IVB separation occurs (if it has not already), an SPS coplanar deorbit burn is performed to return the spacecraft to reentry. The time of deorbit is chosen so as to result in landing at a discrete recovery area, as done in Projects Mercury and Gemini. The SPS deorbit burn is targeted so as to place the earth's horizon at a specified point in the spacecraft window for monitoring purposes.

In summary, then, as regards launch aborts, the main things to remember are: (1) abort capability in one mode or another is available throughout the entire launch phase on a continuous basis; (2) contingency orbit insertion followed by deorbit is always prime when it is available.

3.0 ABORTS FROM EARTH PARKING ORBIT COAST

This mission phase consists of coasting in a circular earth parking orbit from earth orbit insertion to the initiation of translunar injection, a duration which is usually one orbit or longer. As might be expected for this phase, the abort procedures being planned are very similar to those used in Projects Mercury and Gemini.

The propulsion systems available and capable of performing an abort during this phase with their respective ΔV capabilities are shown in Table I. Note that the Service · Module RCS propulsion system does not appear on this table. The reason for this is that when considering only abort trajectories targeted to the center of the reentry corridor, the Service Module RCS does not have sufficient ΔV capability to perform an abort maneuver from the nominally planned earth parking orbit. However, a procedure is currently being evaluated where the abort maneuver is targeted for very near the overshoot boundary of the reentry corridor. Preliminary indications are that an abort using this technique will be available using the Service Module RCS propulsion system, although it will be marginal. As shown in Table I, the propulsion systems which are available and capable of performing an abort are the SPS, the LM propulsion systems, and the S-IVB.

Figure 6 summarizes the abort modes for earth parking orbit. The first and primary mode of interest, as shown, is a single coplanar deorbit burn targeted to provide horizon monitoring and which results in landing at a discrete area. The initiation time of this type of abort is carefully selected to provide the landing area control. This mode is, of course, very similar to that planned for Projects Mercury and Gemini. As shown in the chart of figure 6, the abort can be performed by either the SPS or S-IVB throughout the entire phase with, of course, the SPS being the prime propulsion system. Use of the S-IVB is not desirable due to possible recontact problems during reentry and, thus, would never be considered as an abort mode unless there had been a definite indication by the instrumentation that an SPS failure had occurred prior to CSM separation from the S-IVB. Also, if use of the Service Module RCS to deorbit proves feasible by targeting reentry near the overshoot boundary, the possible use of the S-IVB as an abort propulsion system would seem even more remote. Thus, use of the S-IVB as an abort propulsion system during this phase, seems very improbable even though it is available.

Figure 7 shows the major features of the SPS and S-IVB abort mode during this phase. Note that the transfer angle from the abort maneuver point to reentry is much less than 90°, which means the time from abort to reentry is on the order of 15 to 20 minutes. Command Module/Service Module separation occurs during the coast period from abort to reentry followed by the Command Module orienting itself for reentry. The burn attitude is such that the maneuver is coplanar and such that the earth's horizon remains at a fixed position in the Command Module window for crew monitoring purposes. The ΔV required for abort is approximately 500 fps, which is, of course, well within the SPS capability. The time of deorbit is selected so as to cause landing in a discrete recovery area, as mentioned previously.

The other abort mode possible for this phase is a DPS coplanar deorbit burn. This mode, as shown in figure 6, is available throughout the entire phase, but obviously requires transposition and docking in earth parking orbit.

Figure 8 shows the basic difference between this mode and the previously described one. Note that the deorbit maneuver is such that the spacecraft passes through apogee in order to provide enough time for the crew to transfer from the LM to the CSM prior to reentry. Also, the horizon cannot be easily monitored during the abort burn due to the spacecraft docked configuration. Thus, this attitude restriction is deleted for this mode. As for the S-IVB mode, the use of this mode is very improbable if the Service Module RCS deorbit proves feasible due to the same type of recontact problems upon reentry as would be experienced with the S-IVB deorbit mode.

An alternate DPS abort mode not shown on the summary chart is currently being investigated. This new DPS mode would consist of using the DPS to lower perigee to very near the atmosphere so that the resulting spacecraft trajectory would then be within Service Module RCS capability to deorbit. In other words, the procedure would require maneuvers by both the DPS and the Service Module RCS. This procedure would eliminate recontact problems during reentry, since the LM would be jettisoned between the DPS burn and the Service Module RCS burn.

Summarizing for this phase, one can say that more than adequate abort capability exists from strictly a performance standpoint with essentially three independent propulsion systems capable of providing the abort ΔV required, continuously through the mission phase. However, the use of the S-IVB and DPS, as described in this section, would be very undesirable due to the recontact problems during reentry. If use of the Service Module RCS system to deorbit proves feasible, then the abort modes using the S-IVB or the DPS can essentially be eliminated from consideration.

Since the SPS mode is very similar to that planned for Projects Mercury, Gemini, and the early Apollo orbital flights, the detailed procedures and computer programs are already available and checked out for the ground computers. The SPS deorbit modes could be executed using either the onboard G&N system, the SCS system, or a strictly manual-type abort using visual attitude reference. The DPS abort mode would normally be executed using the onboard G&N system, although it could also be executed using the backup AGS system.

4.0 ABORTS DURING TRANSLUNAR INJECTION PHASE

This phase consists of the S-IVB burn which injects the spacecraft on a highly elliptical trajectory to rendezvous with the moon. The duration of this phase is approximately 340 seconds for a typical lunar mission. What is of interest here is to consider abort capability from orbits which would result from a premature or early burnout of the S-IVB stage.

Figure 9 shows the type of preabort orbits which would result from an S-IVB underburn during this phase of the mission. Note that the family of orbits are elliptical, having very nearly coincident lines of apsides with ever increasing apogee altitude up to and beyond lunar distance. The perigee altitude, however, remains relatively fixed, very near that of the original circular earth parking orbit altitude. The periods of these orbits, as shown in figure 10, vary all the way from $1\frac{1}{2}$ hours to approximately 400 hours as burn time increases. Actually, for free-return translunar profiles, the moon's gravitation perturbs the trajectory resulting from nominal burnout such that return to earth requires much less than 400 hours. Note also in figure 10 that the period remains relatively small (less than 10 hours) for more than three quarters of the way through the burn.

Table II shows the propulsion systems available which are capable of performing abort maneuvers during one portion or another of this phase as well as the subsequent phase, translunar coast. Note that there is a large ΔV capability with the service propulsion system even when the LM is attached. Also, there is moderate capability available with the LM propulsion systems, although use of them requires transposition and docking prior to abort. Note, however, that very little capability is available with the Service Module RCS. Because of this, aborts using the Service Module RCS are marginal at best, as will be shown later.

Figure 11 presents a summary of the abort modes available for translunar injection. Note that redundant abort capability exists throughout the entire phase and even double redundancy for the latter part of the burn. The first and primary abort mode shown in the summary chart consists of a single burn to return the spacecraft directly to reentry, as shown sketched in figure 12. This mode is available throughout the phase with either the SPS or DPS propulsion systems. Note also that the burn attitude is not constrained to be either coplanar or to enable horizon monitoring. A constrained attitude for aborts from this family of ellipses could result in excessive ΔV penalties and could also prevent landing at a desired recovery area.

Seven sub-modes exist which would be available from the ground for abort in this one basic mode. In other words, there exists seven different classes of abort trajectories which will be possible with a single unconstrained attitude burn. There was really no requirement for this kind of flexibility in the previous phase--in other words, during the earth parking orbit phase--because the spacecraft was always in the close proximity of the earth, and there was really no significantly different ways to return to reentry. Beginning with this phase, however, the spacecraft could be in a preabort orbit which extends a considerable distance from the earth. This fact, combined with the large spacecraft performance capability, means that abort trajectories having significantly different characteristics as regards ΔV, return time, and landing point are possible. Therefore, plans are being made to take advantage of this situation to provide greater flexibility in real time planning.

The seven sub-modes available are as follows:

a. Returns to primary recovery sites - This class of abort trajectory returns the spacecraft to a specific primary recovery site within the major constraints mentioned previously. For premature S-IVB shutdown during all but the very last few seconds of the nominal burn, this class of abort trajectories requires that the abort initiation time must be selected very carefully to cause landing at a discrete site, i.e., there will exist only small regions of abort initiation time which will allow returns to a discrete site. For late premature S-IVB shutdowns, however, reasonably placed recovery sites can be reached for any abort initiation time. This sub-mode also required, in general, that a plane change be made in order to reach a primary recovery site, unless there exists sufficient time for the spacecraft to remain in the preabort orbit until the primary recovery site rotates into the proper position for a coplanar abort maneuver. The delay time to abort, however, is usually limited if the apogee of the spacecraft preabort orbit extends into the radiation belts.

b. Time critical returns to a contingency recovery area - This class of abort returns the spacecraft to a contingency recovery area in the quickest possible return time consistent with the constraints of the situation. A contingency recovery area is a continuous line extending from a far northerly latitude, usually 30°, to a far southerly latitude, also usually 30°. These lines will approximately follow the contours of major continents such as the North American and South American continents and, thus, will be very close to being a fixed longitudinal line. Landings on or near these defined lines can be supported by contingency recovery forces, resulting in recovery times nearly as fast, if not as fast, as those associated with the primary recovery sites. The expected existence of four or five of these contingency areas or lines placed equidistant around the world means that no long delay times will be required before the abort can be initiated if a time critical situation exists. The existence of these recovery lines also means that plane change manuevers will never buy a great amount of return time. For these reasons, this sub-mode is defined to be a coplanar maneuver with the abort initiation time selected to cause landing at the first accessible contingency recovery area or line.

c. Fuel critical returns to contingency recovery areas - This class of abort returns the spacecraft to the same contingency recovery areas or lines as described above, on a trajectory which requires the minimum possible ΔV or fuel. This class of abort return would only be used in a situation where there had been a serious degradation in the propulsion system. The very nature of the sub-mode limits or restricts the region of abort initiation to the vicinity of apogee of the preabort orbit.

d. Time critical returns to a water landing - This class of abort returns the spacecraft to a water landing in the quickest possible time. Note that the only restriction on the landing point is that it be on water. The landing is not restricted to a particular latitude, longitude, or combination thereof. It is assumed in this sub-mode, as it was in the previous time critical sub-mode, that the savings in return time made possible by plane changes were not worth the added logic complexity. Therefore, returns in this abort sub-mode are restricted to coplanar maneuvers. The selection of the abort initiation time is not as critical for this sub-mode as for those described above, and for a large percentage of the cases, the abort maneuver can be executed as soon as possible, i.e., as soon as the crew and systems are prepared and checked out. The only instances when the abort initiation time must be be delayed is to prevent a land landing. It should be pointed out that the use of this sub-mode is very improbable due to the small savings in return time possible when compared to the time critical returns to a contingency recovery area, as described above.

e. Fuel critical returns to a water landing - This class of returns is restricted to a water landing, as above, but provides the minimum ΔV solution rather than the quickest return time. Again, as in the fuel critical sub-mode described above, this class of return trajectory would only be used in the situation where a serious degradation has occurred in the propulsion system. As would be expected, the region of abort initiation times for this sub-mode is restricted to the vicinity of apogee of the pre-abort orbit. The use of this sub-mode is more probable than its time critical counterpart due to the fact that a return to a contingency recovery area or line may not be possible when the abort initiation is restricted to the vicinity of apogee. Thus, a fuel critical return to a water landing would be the next best possible way to return.

f. Time critical returns to unspecified landing area - This class of abort returns the spacecraft in the quickest possible time with no restriction on the point of landing, i.e., the landing could be on land or water. These returns, by their very nature, are required to be coplanar and are initiated as soon as possible, i.e., as soon as the crew and systems are prepared and checked out. The use of this type of return is even more improbable than the time critical water landing sub-mode due to the small savings in return time as compared to the other possible sub-modes, and due to the possible degradation of crew safety associated with a land landing. Only a very extreme contingency situation would justify such drastic action.

g. Fuel critical returns to an unspecified area - This class of abort returns the spacecraft to reentry with the minimum possible ΔV required. The definition of this sub-mode requires that the maneuver be performed coplanar at apogee of the preabort orbit. This sub-mode would be used when only the Service Module RCS system was available and when the abort initiation time could not be delayed to some subsequent apogee passage due to radiation dosage or a preabort reentry condition, i.e., when the perigee of the preabort orbit is so low as to cause reentry during that particular orbit.

A straightforward procedure for selection of the best sub-mode is virtually impossible due to the complexity of the spacecraft systems involved, the possible failures or combination of failures which could have occurred, and the wide range of possible preabort orbits including the wide range of possible geographic orientations possible. For these reasons, flight controller judgment in real time must be heavily relied upon to make the sub-mode selection based on the exact situation at hand. The flight controllers will, of course, develop many mission rules to aid him in this decision. The current plan is to display the significant characteristics of abort trajectories in these various categories or sub-modes to the flight controller after a decision to abort has been made. These characteristics would include such items as (1) ΔV required,

(2) return time, (3) landing location, and (4) whether landing is in daylight or darkness. The flight controller would then select the one which, in his judgment, provides the greatest margin of crew safety. It is obvious that, all things being equal, an abort to a primary recovery site would be selected if at all possible. In order of decreasing priority, then, would be aborts to a contingency recovery area, aborts to a water landing, and finally, for an extreme contingency situation, aborts to unspecified areas. However, all things may not be equal for a given abort situation, and a tradeoff may be required between the possible ways of returning in order to maximize crew safety.

Note that the above described sub-modes would be available from the ground not only for this phase but for all the remaining mission phases to reentry. The onboard backup capability will be restricted to the latter two sub-modes (f and g) due to computer storage limitations, but certain manual iterations can be performed by the crew to provide some measure of control over longitude of landing.

As shown in figure 11, the only other basic abort mode for this phase is the Service Module RCS coplanar deorbit returning the spacecraft directly to reentry using the minimum possible ΔV. This mode of abort is available in one of three possible sub-modes, as described below.

a. Fuel critical returns to contingency recovery areas.

b. Fuel critical returns to water landing.

c. Fuel critical returns to an unspecified area.

The definitions of these sub-modes are identical to those described previously for the SPS and DPS propulsion systems. Note from figure 11 that this mode of abort is available only during the latter part of the TLI burn. Again, it should be emphasized that this latter statement assumes that targeting for the abort maneuver would be for the middle of the reentry corridor. Targeting for near the overshoot boundary could conceivably make this made of abort available throughout the entire phase.

Another abort mode not shown in the summary chart is currently under investigation and evaluation. This new mode would consist of two burns and would be considered only if the SPS were available. The first burn would be a circularization maneuver at perigee of the preabort ellipse. The second burn would be a standard deorbit maneuver as described in the earth parking orbit phase, i.e., a coplanar burn targeted so as to provide the crew with horizon monitoring. This procedure would allow the crew to retain the simplified abort procedures of the earth parking orbit phase through a considerable portion of the translunar injection phase. The use of this abort procedure would be curtailed for preaborts occurring during the latter portion of the translunar injection burn. This curtailment would occur due to an excessive ΔV requirement to circularize. In other words, the ΔV required to circularize would not leave enough ΔV remaining to deorbit. Another possible limitation of the use of this abort procedure would be that it would only be considered if the primary G&N system of the CSM were operating. A manual-type circularization maneuver would never be attempted due to the magnitude of the maneuver required. However, if the current investigation proves this mode to be feasible, it would probably become the prime abort mode for the portion of the translunar injection phase where it is possible, if the primary G&N and SPS are available.

Figure 13 shows the general characteristics of a fuel critical abort trajectory regardless of the propulsion system being considered. Note that the burn is in the near vicinity of apogee and that the maneuver is of the Hohmann type to lower perigee into the atmosphere.

Figure 14 presents the ΔV requirements for fuel critical aborts with unspecified landing area as a function of the time of premature S-IVB shutdown. In other words, this curve represents the absolute minimum ΔV required to abort of all the possible modes and sub-modes discussed above if targeting is restricted to the center of the reentry corridor. Note that the ΔV required begins at a value of approximately 500 pfs at zero burn time--when the spacecraft is essentially still in a circular orbit--to a value very nearly zero at nominal cutoff. Shown as limit lines on the plot are the various ΔV capabilities associated with the available propulsion systems. Note that the Service Module RCS is the only propulsion system which can be considered critical for this phase. As shown, use of this system can only be considered feasible for shutoffs occurring during the latter part of the translunar injection phase, but it should be emphasized again that this is assuming targeting for the center of the reentry corridor. The ΔV capabilities associated with the SPS (even with the LM) and the DPS far exceed the minimum ΔV requirements. This large ΔV capability margin, which exists with the SPS or DPS propulsion systems can be utilized to control the abort landing point and/or to speed up the return trajectory.

Figure 15 shows the general characteristics of an abort trajectory when this excess ΔV margin is used to speed up the return. As shown, a time critical abort trajectory during this mission phase is usually characterized by a post-apogee type return. In other words, the abort trajectory does not pass through apogee between the abort maneuver point and reentry.

Another general characteristic of a time critical return is that the overall quickest return time usually results when the abort is performed as soon as a solution is available. For the time critical unspecified area sub-mode, this means that the abort should be performed as soon as preparation time permits to minimize the return time.

Figure 16 shows the minimum possible return time in the time critical unspecified area sub-mode as a function of the time of premature shutdown during translunar injection. As shown, a ΔV of approximately 10,000 fps was assumed in the generation of this curve. Since this is the maximum ΔV possible from any of the available propulsion systems, this curve is a limiting or bounding line. In other words, this curve represents the absolute minimum possible return time for a single burn procedure. The other assumption made in the generation of this data was that abort occurred one-half hour following translunar injection burnout.

As shown, the minimum possible return time is always less than approximately $3\frac{1}{2}$ hours no matter where premature shutdown occurs. Note, also, as would be expected, the return time increases with increasing translunar injection burn time. It should be understood that an abort maneuver this severe would probably never be attempted due to the very size of the maneuver required and due to the small probability of it being necessary.

Figure 17 presents the minimum possible ΔV requirements to abort to a typical primary recovery site during the translunar injection phase. This type data are very mission dependent so only very generalized conclusions can be drawn. The ΔV required to abort is shown plotted in contour form as a function of the translunar injection burn time on the vertical axis, and the abort delay time or the time of abort as measured from translunar injection burnout on the horizontal axis. The eccentricity of the preabort orbit is also shown plotted on the vertical scale with the translunar injection burn time. The particular case shown is for returns to Bermuda for the DRM II lunar landing mission.

Note that for early premature burnout times, the ΔV requirements are cyclical with delay time and that these requirements vary very rapidly with delay time. Note also that there are regions which exist where no abort solutions are possible even assuming a ΔV of 10,000 fps available. As shown, the bands of possible solutions become more and more narrow with delay time for this particular case. This characteristic is due to the fact that the landing site is rotating farther and farther out of the preabort orbit plane. If delay time were extended further than is shown on this plot, the bands would eventually disappear and then reappear in ever-widening bands as the landing site again rotated into the favorable position with respect to the inertial preabort orbit plane. This effect, of course, is very mission dependent and the exact opposite could have occurred. In other words, the bands could have become wider and wider with delay time instead of more narrow if the landing site chosen for consideration had been rotating into the plane rather than away from the plane. Another noticeable trend on this data plot which is also true for the general case is that the ΔV requirements decrease with increasing translunar injection burn time. Also, for late premature shutdowns, abort solutions are available continuously with delay times and no gaps or voids occur.

Figure 18 shows typical return times to the same primary recovery site as considered on the previous plot. As for the previous plot, the data is shown plotted as a function of translunar injection burn time and abort delay time. These data assume that the entire SPS capability is available for use. Note that although a few regions are available where returns in a few hours are possible, in general, returns are forced to at least one day in return time.

Summarizing the translunar injection phase, then, one can state that: (1) redundant abort capability exists continuously throughout the translunar injection phase; (2) the spacecraft performance margin is great enough to allow considerable flexibility in selection of return times and choice of a landing area; (3) the only critical or marginal abort mode is when the Service Module RCS system must be relied upon to perform the abort maneuver. This mode would only be required in the event of an SPS failure as well as a failure to perform transposition and docking to obtain the LM propulsion systems. Thus, the use of this mode would seem very improbable.

5.0 ABORTS DURING TRANSLUNAR COAST PHASE

This mission phase consists of the coast period from translunar injection burnout to the initiation of lunar orbit insertion, a period of approximately 62 to 74 hours for free-return trajectories. Since the nominal translunar coast trajectory is a free return to earth, the minimum ΔV to abort is essentially zero. Thus, as will be shown later, large ΔV capability margins exist which can be used in an abort situation to speed up the return to earth and/or control the point of landing.

Figure 19 presents a summary of the abort modes for the translunar coast phase. Note that redundant abort capability exists throughout the entire phase due to the presence of three independent propulsion systems.

The first abort mode listed is a direct abort available with either the SPS or LM DPS propulsion systems. The basic features of this abort mode are shown sketched in figure 20. This mode consists of a single unconstrained attitude abort burn which returns the spacecraft directly to reentry without circumnavigating the moon. Shown sketched, are the two extreme types of returns possible in this mode--time critical and fuel critical. A fuel critical return usually passes through apogee following the abort maneuver; whereas, a time critical or a fast return usually does not pass through apogee following the abort maneuver. All of the seven sub-modes described previously for a translunar injection phase are available for this mode as well as for all of the other modes for this phase. If the SPS is available, this mode of abort produces the fastest possible returns to the earth for aborts performed prior to reaching approximately the lunar sphere of influence. This is the reason why this particular mode is considered prime for approximately the first three-fourths of the coast period from the earth to the moon.

The DPS direct abort mode, as shown in the summary chart, is available as a backup mode to the SPS direct mode for approximately the first two-thirds of the coast period from the earth to the moon. Use of this mode during the latter portion of the translunar coast phase would result in excessively long return times. Thus, the use of this mode as a backup during this latter period is not considered as a possibility. As will be shown in more detail later, the LM DPS direct abort mode, when used in the event of an SPS failure, will result in the fastest possible return time only if the abort maneuver is performed during approximately the first 25 hours of translunar coast.

The next abort mode listed in the summary chart consists of delaying the abort maneuver until the vicinity of pericynthion or perilune is reached, as shown sketched in figure 21. This mode of abort is available continuously throughout the entire translunar coast phase with either the SPS, DPS, or marginally with the Service Module RCS system. If the SPS is available, this mode of abort will generally result in the fastest possible return to the earth for abort decisions made near the sphere of influence or thereafter. For this reason, it is shown in the summary chart as the prime mode of abort for this portion of the translunar coast phase. In the event of an SPS failure, the LM DPS propulsion system used in this mode will result in the fastest possible return time for abort decisions made approximately 25 hours or later out along the translunar coast trajectory. In the event of an SPS failure and failure to obtain the LM DPS propulsion systems, the Service Module RCS would be marginally available for this mode of abort due to the fact that the nominal translunar coast is a free-return trajectory.

The third abort mode listed in the summary chart is a circumlunar abort available with the SPS, DPS, or marginally with the Service Module RCS continuously throughout this mission phase. Figure 22 shows the basic features of this abort mode to be a single unconstrained burn performed at some point along the translunar coast trajectory which returns the spacecraft to the earth after circumnavigating the moon. The altitude of pericynthion or perilune is allowed a certain amount of freedom in order to obtain landing area control upon return to earth. This mode of abort is not as yet thoroughly understood. Preliminary analysis indicates that this type of abort many times produces the absolute minimum ΔV required to return to a particular landing area, especially if the translunar coast profile is a non-free-return type. However, for just as many cases the minimum ΔV required to return to a particular recovery area has been found to be minimized by use of the delay to pericynthion abort mode. A complete understanding of this effect is currently under investigation.

Figure 23 shows a special case of the circumlunar abort mode. This particular case consists of merely using midcourse corrections to correct back to the nominal free-return trajectory, assuming this type of return is the nominal profile. Use of the circumlunar abort mode in this fashion is essentially equivalent to the fuel critical unspecified area sub-mode described previously.

The last mode listed on the summary chart is a two-burn mode for the special case of when the translunar coast trajectory is on an impact course with the moon. As shown, this mode of abort is available continuously throughout the entire phase with either the SPS, DPS, or marginally with the Service Module RCS. Figure 24 shows the basic features of this abort mode. The first maneuver, usually a small one, is used to raise perilune or pericynthion altitude to an acceptable value. The second burn is then performed in the vicinity of perilune to return the spacecraft to earth in one of the seven sub-modes described previously. The most probable need for this abort mode would be in the event of a badly executed second midcourse correction near the sphere of influence due to a G&N or an SPS failure.

Figure 25 shows a very significant characteristic of translunar coast abort trajectories, expecially as regards returns to a primary recovery site. As shown, the transfer angle from abort to reentry is relatively insensitive to the time of abort or to the type of abort return. This angle varies from approximately 170° to 180° throughout the entire translunar coast phase. This fact, combined with the fact that the reentry ranging capability is also relatively fixed, means that the inertial position of landing is approximately fixed for a given return plane. This means that the abort trajectory problem is essentially a timing or a rendezvous problem. In other words, the return must be timed such that, as the spacecraft reaches its relatively fixed inertial landing point, the desired landing area is just rotating underneath. If the fastest possible abort return just misses rendezvous with the desired landing area, then a delay in landing time of nearly 24 hours results in order to allow the desired landing area to make a complete revolution to again reach the desired inertial rendezvous point. Thus, more than one solution is possible, but the possible solutions occur in 24-hour increments as regards landing time.

Figure 26 shows that this 24-hour landing effect is also present when the time of abort is added as an additional degree of freedom to the problem. This effect occurs because the inertial direction of the abort point changes very little with abort time along the translunar coast trajectory. An abort from point 1, as shown in figure 26, to a particular landing site might require relatively low energy in order to achieve rendezvous with a desired landing site. An abort at a later time, as represented by point 2, would require a higher energy return trajectory if the spacecraft is to rendezvous with the same landing site at the same time as the abort trajectory 1. Similarly, an abort trajectory performed at still a later time, point 3, would require even more energy resulting in an even faster return trajectory in order to account for the time lost in delaying the abort maneuver. Obviously, a limit will be reached as regards delay time when either the Δ V required for abort or the velocity at reentry violates its respective limit. When this limiting or critical delay time is reached, an abort would be forced to a landing 24 hours later, as represented by point 4 in figure 26.

An interesting feature of this 24-hour effect for aborts to a particular landing site is that returns will either all be in daylight or all in darkness. In other words, if an abort solution to a primary recovery site such as Hawaii lands in darkness, a solution cannot be found for any other abort delay time or any other ΔV required which will cause landing at Hawaii to be in daylight.

Figure 27 shows this 24-hour effect in the form of actual abort trajectory performance data computed for a typical translunar coast trajectory. Shown plotted is the time of landing measured from translunar injection as a function of the time of abort measured from translunar injection. Data is shown for three primary recovery sites--Indian Ocean, Hawaii, and Bermuda. The data shown for aborts an SPS abort in the same situation. Note also that the switchover line dividing direct aborts from post-pericynthion aborts is now back to approximately 22 hours after translunar injection due to the much smaller ΔV capability associated with the DPS propulsion system.

Figure 29 shows another significant characteristic of translunar coast aborts as regards landing accessibility. Note that the abort return geometry requires that all possible abort trajectories from a given abort position must pass through the antipode or negative of the abort position vector. Since the transfer angle from abort to reentry is always approximately 170 to 180 degrees, then the reentry positions must be in the immediate vecinity of the antipode. The fact that the reentry ranging capability is relatively limited together with the constraint on return inclination forces the landing point to a relatively restricted band of possible latitudes.

Figure 30 shows typical landing area accessibility for aborts from a point midway along the translunar coast trajectory. The shaded region represents the possible geographic landing areas for aborts from this one position vector. Note that the region of possible landing latitudes is relatively restricted whereas any longitude is accessible. The widening of the accessible area from east to west is associated with slower and slower return trajectories. A slower return means that more energy can be utilized for plane changes which widens the possible latitude band. The particular case shown is an abort position vector at a southerly latitude or declination; therefore, the abort antipode, as well as all possible landing points, are notherly in latitude or declination. Thus, recovery sites should be placed at northerly latitudes when the lunar mission is planned for a southerly lunar declination and vice versa. Proper positioning of recovery sites is represented in figure 30 by the "X" marks.

Summarizing, then, for the translunar coast phase: (a.) Redundant abort capability exists throughout the entire phase. (b.) Spacecraft ΔV capability is large compared to the minimum required due to the free-return profile. (c.) This large margin of capability can be utilized to speed up returns and/or to control the landing area. (d.) No portion of this mission phase is critical or marginal unless both SPS and DPS failures occur.

6.0 ABORTS DURING LUNAR ORBIT INSERTION PHASE

This phase consists of the SPS maneuver which transfers the spacecraft from the approach hyperbola at the moon into a circular parking orbit about the moon. The prior to the time of pericynthion were computed in the direct abort mode. Note that the time of landing for a particular landing site is relatively insensitive to the time of abort until a limit is reached (Δ V required or reentry velocity). At this point, aborts to that same landing site are forced to a time of landing 24 hours later where it remains fixed with delay time until one or the other of the limiting constraints is reached again. Note, however, that if a different landing site is considered when a limit is reached, the jump or discontinuity in time of landing would be something less than 24 hours. Therefore, if the recovery sites were placed approximately equidistant around the world, the incremental increases in the time of landing to a specific site would always be some fraction of 24 hours. Another interesting feature of figure 27 is that, after an abort time of approximately 57 to 58 hours, an abort delayed to the vicinity of pericynthion or beyond will result in the same time of landing as direct aborts prior to pericynthion. Also, since aborts performed following pericynthion will always require much less ΔV than direct aborts prior to pericynthion, the obvious course to follow is to delay abort to post-pericynthion after this switchover line is reached on the translunar coast trajectory. It should be pointed out that this switchover line from direct abort to delay abort to post-pericynthion is a strong function of ΔV required for abort. The particular switchover line shown in this figure assumes 90% of the entire SPS capability (9,000 fps) available for abort.

Another interesting characteristic to note from this figure is that returns from the vicinity of pericynthion are limited to approximately 40 to 50 hours return time due to the reentry velocity constraint. Without this constraint, aborts could be performed post-pericynthion with return times of approximately 25 to 35 hours, assuming 9,000 fps available.

Figure 28 shows the same type of data for DPS aborts from the same translunar coast trajectory as used for figure 27. Note that although the times of landing still demonstrate the 24-hour effect, the times are much later than those for SPS aborts. For example, consider an abort performed at 20 hours after translunar injection targeted to return to Hawaii. Figure 28 shows that a DPS abort in this situation would result in a time of landing of approximately 120 hours after translunar injection. However, figure 27 shows that a SPS abort in this same situation would result in a landing at Hawaii at approximately 48 to 50 hours after translunar injection. Thus, in this particular case, a DPS backup abort would return three days later than length of the lunar orbit insertion burn is approximately 380 seconds for a typical mission. Of interest here is the abort capability from preabort trajectories resulting from premature or late SPS burnout during the lunar orbit insertion burn. As will be described later, the abort procedures are dependent upon the time of SPS burnout during this phase.

Figures 31 and 32 show the different kinds or classes of preabort trajectories which could result from an underburn or an overburn during lunar orbit insertion and the associated burnout times at which they would occur. For early premature burnout, approximately 0 to 130 seconds, the resulting trajectories escape the influence of the moon and return into the earth's influence. These escape trajectories are either hyperbolic or extremely elliptical relative to the moon.

For premature burnouts occuring from approximately 130 to 145 seconds, the resulting preabort trajectories are unstable in character. The apogees of this class of preabort orbit extend to the vicinity of the lunar sphere of influence where the trajectory loses its central force field or two-body type of motion due to the large perturbations from the earth. These perturbations are so severe as to sometimes cause the motion to reverse from retrograde to posigrade with respect to the moon, as shown in fugure 31. The periods of these preabort trajectories, if such can be defined, are generally greater than 100 hours.

For premature shutdowns from approximately 145 to 160 seconds, the resulting trajectories are relatively stable ellipses but are pertubed sufficiently by the earth to cause impact after one revolution. Periods for this class of trajectories range from approximately 35 to 100 hours.

For burnouts occurring from approximately 160 seconds to 10 seconds past the nominal burnout, the resulting trajectories are non-impact stable ellipses with periods ranging from approximately 35 to 2 hours.

Premature burnouts occurring 10 seconds or more after nominal burnout result in impact ellipses after less than one-half revolution. This class of preabort trajectory is quite obviously the most critical from an abort standpoint, but also is the most improbable one to occur due to the redundant monitoring devices available to the crew onboard the spacecraft.

Before discussing the various abort modes and procedures for thses preabort trajectories, it would seem worth while

The last major abort mode associated with this phase is a two-burn abort procedure for LOI overburns which result in preabort ellipses which impact in less than one-half revolution. Use of either the SPS or the LM propulsion systems can be considered with this mode although use of the LM propulsion systems is marginal. Figure 38 shows that the first burn of this two-burn procedure is executed as soon as possible to clear perilune or pericynthion altitude to an acceptable value. The second burn is executed on some later orbit in the approximate quadrant shown to return the spacecraft to earth. It again should be emphasized that it is very improbable that such an overburn could occur due to the redundant monitoring capabilities available to the crew during the lunar orbit insertion burn.

Figure 39 shows the effect of delay time on abort V required for preabort escape trajectories. Shown plotted is the minimum ΔV required to abort as a function of the time of abort measured from lunar orbit insertion burnout Data for two different escape trajectories are shown-- one associated with a lunar orbit insertion premature burnout after 40 seconds and the other after 80 seconds. Note that the ΔV requirements tend to flatten out after the first one or two hours with the later burnout producing the steepest initial slope. Although the abort requirements for the two trajectories shown are well within the LM propulsion system capability, the trend should be obvious that later and later premature burnouts will eventually cause the LM propulsion system capability to be exceeded for direct-type ASAP aborts.

Figure 40 shows the same type data for a typical preabort ellipse having a period of approximately 4 hours. Note that, although the initial slope of the abort ΔV required/delay time curve is extremely steep, the ΔV required is cyclical with delay time. As shown, this effect permits aborts with the LM propulsion system if the proper time of abort is chosen. This proper time of abort corresponds to the vicinity of perilune of the preabort ellipse.

Figure 41 displays the abort capability using the LM propulsion systems throughout a typical lunar orbit insertion phase. Shown plotted is the ΔV required to abort as a function of the time of premature lunar orbit insertion burnout for several delay times. Also shown on the plot are two limit lines associated with the LM propulsion system capability. First notice that the minimum ΔV required for an immediate abort (essentially zero delay time) is always within the capability of the LM propulsion systems throughout the entire lunar orbit insertion phase. Note, however, that for an abort delay performed. Thus, precious "thinking" and preparation time can be had to obtain a better assessment of the situation before action is taken. Note from the summary chart that this mode using the LM propulsion systems is shown available through the first 200 seconds of burn time. This assumes that the LM can be prepared for an abort maneuver in approximately one-half hour. For longer preparation times, use of the LM in this mode would be eliminated at an earlier LOI burn time.

The next mode of interest is a two-burn procedure for use when the spacecraft is on a preabort ellipse which impacts the moon after one revolution. As shown in the summary chart, this mode is available with either the SPS or the LM propulsion systems for a period of approximately 15 to 20 seconds near the middle of the lunar orbit insertion burn. This time period, of course, corresponds to the premature burnout times which would result in impact ellipses. Figure 36 shows the basic features of this abort mode. Note that the first burn is made in the vicinity of apogee to raise perilune to an acceptable value with the second burn being executed on the back side of the moon in the vicinity of perilune to return the spacecraft to earth. This mode is considered prime only during the latter part of its region of availability due to the excessive delay times associated with the preabort ellipses during the earlier portion. A very significant feature of this abort mode, as shown in the summary chart, is that it overlaps the immediate direct abort mode even when considering use of the LM propulsion systems. The amount of this overlap, however, is dependent upon the LM preparation time required in the direct abort mode. It should be recalled that the LM propulsion system direct abort mode shown in the summary chart assumes a LM preparation time of one-half hour. Longer preparation times would reduce this degree of overlap and, as will be shown later in more detail, could even eliminate the overlap between the two modes.

The next abort mode to be considered consists of delaying one or more orbits to abort when premature SPS burnout results in a stable non-impact ellipse. As shown, this mode is available using either the SPS or LM propulsion systems throughout the entire latter half of the burn phase. The basic features of this abort mode are shown sketched in figure 37. As shown, the maneuvers are always delayed at least one orbit with execution occurring in the vicinity of perilune to return the spacecraft to earth. The longest delay time which would ever be encountered before the abort maneuver could be executed is approximately one and a half days corresponding to the maximum period of the possible non-impact ellipses.

to describe the two basic ways of returning from the moon and why one can be ruled out. Figure 33 shows the two basic ways of returning in a simplified "patched conic" model. As shown, after exiting the moon's sphere of influence, the spacecraft will either pass through apogee along the earth phase conic or it will not pass through apogee. In order to return post-apogee, the spacecraft must exit the moon's sphere of influence in the quadrant shown such that the exit velocity with respect to the moon will combine with the velocity of the moon with respect to the earth to result in the required velocity of the vehicle with respect to the earth. Similarly, if the spacecraft is to return pre-apogee in the earth phase conic, the exit point must be on the far side of the moon in order that the velocity of the spacecraft relative to the moon can combine with the velocity of the moon relative to the earth to result in the required spacecraft velocity relative to the earth. This latter class of return trajectories which pass through apogee along its earth phase conic can be ruled out from consideration due to prohibitive return time and very difficult targeting requirements.

The propulsion systems available for abort during the lunar orbit insertion phase are shown in table III along with their respective ΔV capabilities. Note that there are basically two independent propulsion systems available during this phase--the SPS system and the LM propulsion systems. The ΔV capabilities vary through the phase due to the change in mass brought about by SPS fuel expenditure.

Figure 34 summarizes the abort modes for the lunar orbit insertion phase. As shown, redundant abort capability exists continuously throughout the entire phase, if the proper abort modes are chosen.

The first abort mode listed is an immediate (as soon as possible) direct abort, as shown sketched in figure 35. This mode is available with either the SPS, the LM propulsion systems, or for a very small region, the Service Module RCS System. This mode consists of a single burn performed as soon as preparation is completed which returns the spacecraft directly to earth. As shown in the summary chart, this mode is available throughout the entire phase when the SPS is available and is shown as prime for approximately the first 150 seconds of lunar orbit insertion burn. This mode is not prime, though available, for the latter portion of the lunar orbit insertion burn due to the fact that the preabort trajectories are elliptical with periods short enough to reasonably allow one complete revolution before the abort maneuver is time of one-half hour, direct aborts cannot be considered using the LM propulsion systems for premature burnout after approximately 190 seconds. The cutoff for delays of one hour and two hours occurs at approximately 160 and 135 seconds, respectively. However, as shown, LM DPS aborts using the delay-one-orbit mode for burnouts in the vicinity of 140 to 150 seconds result in extremely long delay times before the abort maneuver to return the spacecraft to earth can be executed. These delay times are on the order of approximately 100 hours. Thus, in order to eliminate the possibility of being forced to these extremely long delay times, the LM preparation time should be minimized to produce as large an overlap as possible between the direct abort mode and the delay-one-orbit abort mode. Under investigation now is the possibility of checking out at least some of the LM systems prior to lunar orbit insertions to expedite abort preparation time. Other studies now being performed in an attempt to alleviate this problem are:

a. Evaluation of a two-burn abort procedure where the first burn would shorten the period of the preabort ellipse with the second burn being executed in the vicinity of perilune to return the spacecraft to earth.

b. Evaluation of the use of the Service Module RCS to continue the lunar orbit insertion burn in the event of a SPS premature burnout as long as possible in order to shorten the period of the preabort ellipse.

Figure 42 presents abort performance data for returns to primary recovery sites from a preabort ellipse about the moon, having a period of approximately 19 hours. Shown plotted is the time of landing as measured from lunar orbit insertion burnout as a function of the time of abort measured from lunar orbit insertion burnout. Again, as for translunar coast, three recovery sites are considered--Indian Ocean, Hawaii, and Bermuda. Note the the 24-hour effect appears here as it did in the translunar coast phase; and also note that the times of landing are cyclical (repeatable after one orbit).

Summarizing, then, for the lunar orbit insertion phase it can be seen that:

(a.) Redundant abort capability exists throughout the entire lunar orbit insertion phase if the proper abort mode is chosen.

(b.) The only critical regions are:

  1. Premature burnouts in the vicinity of 150 seconds where the delay-one-orbit mode would result in extremely delay times before an abort maneuver could be executed.

2. Overburns which would result in impact ellipse in less than one-half revolution.

However, different techniques are being studied to eliminate the possibility of the long delay times associated with critical region "a" and critical region "b" is very improbable due to crew monitoring techniques.

7.0 ABORTS DURING LUNAR ORBIT COAST PHASE

This phase of the lunar landing mission consists of the spacecraft coast in lunar orbit from lunar orbit insertion to initiation of transearth injection. As would be expected, the abort procedures are very similar to the normal transearth injection procedures.

The propulsion systems available for aborts during the lunar orbit coast phase are shown in table IV with their ΔV capabilities. Again, two independent systems are available--the SPS and the LM propulsion systems.

Figure 43 summarizes the abort modes for the lunar orbit coast phase. As shown, continuous abort capability exists using the SPS, but redundant abort capability exists only for the early portion of the phase prior to initiation of LM descent. The abort mode for this mission phase consists of a single abort burn on the far side of the moon to return the spacecraft to reentry in a manner similar to that of earth injection, as shown in figure 44.

Figure 45 shows the affect of delay time on abort ΔV required for aborts for the lunar orbit coast phase. Plotted is the minimum ΔV required for abort as a function of the time of abort and the longitude of abort for a typical landing mission. Although the slopes are steep, the ΔV requirements are cyclical in phase with the period of the orbit. Note also that the minimum ΔV required occurs at a longitude just prior to 180°.

Summarizing the abort capability from the lunar orbit coast phase, then, one can see that continuous abort capability exists throughout the entire phase only if the SPS is available.

8.0 ABORTS DURING TRANSEARTH INJECTION PHASE

This phase consists of the SPS burn which injects the spacecraft from lunar orbit onto a return trajectory to earth. The length of this phase for a typical lunar landing mission is approximately 120 seconds. The aborts of interest are those performed from trajectories resulting from premature SPS burnout during this phase. As would be expected, the abort procedures are very similar to those used for the lunar orbit insertion phase.

Figures 46 and 47 show the different classes of preabort trajectories and their respective times of occurrence for the transearth injection phase. As shown, the same classes of trajectories exist as for the lunar orbit insertion phase with the exception of the overburn impact ellipse class.

Table V shows the only propulsion system available for the transearth injection phase with its ΔV capability. As for the lunar orbit insertion phase, the ΔV capability varies due to the change in mass through the TEI burn.

Figure 48 summarizes the abort modes for the transearth injection phase. Note that continuous abort capability exists if the SPS can be restarted. As shown, the modes are very similar to those planned for lunar orbit insertion. The delay-one-orbit mode is prime during the earlier portion of the transearth injection burn to allow additional delay time for a better assessment of the situation. However, as shown, this abort mode is not considered prime after approximately 70 seconds due to the exessively long delay times required before the abort maneuver could be initiated.

Likewise, the two-burn, delay-one-orbit procedure is not considered prime, even though it is available for a small region, due to the long delay times required before the abort could be initiated. There is more reason to avoid these long delay time regions during this phase than for the lunar orbit insertion phase, due to the lifetime of the spacecraft's systems becoming more critical as the length of the of the mission increases.

The immediate direct abort is available throughout the entire phase and is considered prime during approximately the last half of the phase to avoid excessive delay times associated with the delay-one-orbit modes.

Figure 49 shows the SPS abort capability during a typical transearth injection phase. Shown plotted is the minimum ΔV required for abort as a function of the time of premature transearth injection burnout. Shown plotted are the ΔV requirements for an immediate abort, an abort delayed one hour, and an abort delayed two hours. Also shown are the abort requirements for a delay-one-orbit abort which happened to be identical to the immediate abort ΔV requirements for that portion of the mission phase. Shown as a limit line is the curve representing SPS capability for this particular mission. It should be recognized that this limit line is mission dependent as the amount of SPS fuel reserve will vary from mission to mission. As shown, aborts delayed two hours or longer will completely eliminate the overlap between the direct abort mode and the delay-one-orbit to abort mode.

Summarizing the abort capabilities for the transearth injection phase, then:

a. Continuous abort capability exists only if the SPS can be restarted.

b. The only critical region exists for premature burnout in the vicinity of 80 seconds where allowable delay time before the SPS must be restarted is on the order of two hours. Premature burnout before or after this critical region allows delay times greater then two hours before restart is required.

9.0 ABORTS DURING TRANSEARTH COAST PHASE

This mission phase consists of the coast trajectory from transearth injection to reentry. The nominal transearth coast results in landing at prime recovery site. An abort could (1) speed up the return to the same site or to an alternate site or (2) change the landing site to avoid bad weather.

Since the transearth coast is already returning to a prime recovery site, however, the probability of ever aborting during this phase is small.

Table VI lists the propulsion system available and capable for abort during this mission phase and its ΔV capability. Note that the only system available is the SPS with a typical ΔV capability of 1,500 fps. This ΔV capability is, of course, very mission dependent but will usually be of this order of magnitude, at least for the earlier missions.

Figure 50 is a summary of the abort modes for the transearth coast phase. Note that only one mode is available, that being a single SPS burn returning the spacecraft directly to reentry, as shown sketched in figure 51. The mode is available continuously throughout the entire phase, although it is obvious that the closer to reentry, the less likely an abort would be executed.

Figure 52 presents actual abort performance data for aborts to primary recovery sites for a typical transearth coast trajectory. Shown plotted is the time of landing as measured from transearth injection as a function of the time of abort measured from transearth injection. Data is shown for three primary recovery sites--Indian Ocean, Hawaii, and Bermuda. As shown, the nominal transearth trajectory is targeted to return to Hawaii at the time of landing of 90 hours as measured from transearth injection. Aborts with a time of landing 24 hours earlier to Hawaii can be performed during the first 10 hours of transearth coast. Note also that the faster returns are available to alternate landing sites during the first 45 hours of transearth coast. Also shown is the availability of returns to an alternate (Indian Ocean) site with a later time of landing up through approximately the first 55 hours of transearth coast. These data, then, show considerable flexibility or abort capability to speed up the return and/or to change the point of landing. It is also of interest that returns to primary recovery sites still display the 24-hour effect, as in the previous phases.

10.0 CONCLUSIONS

A. Continuous abort capability exists for all mission phases under the following conditions:

1. For the lunar orbit insertion and transearth injection phases, the selection of the proper abort mode as a function of the time of premature burnout is a requirement.

2. Following the initiation of LM descent in lunar orbit, the SPS must be operable to provide abort capability.

B. Continuous redundant abort capability exists for all mission phases prior to the initiation of LM descent with the exception of the launch phase.

Questions and Answers

EARTH RETURN ABORT CAPABILITIES

Speaker: Ronald L. Berry

1. Dr. Rees - Why not use the APS for a backup propulsion system?

   ANSWER - Since the ascent engine is not gimballed, the possible c.g. offset effects cannot be controlled.

2. What is the RCS engine burn time limitation?

   ANSWER - Specifications limit is 1000 seconds.

3. Dr. Mueller - Have methods to restart the SPS engine in the event of an early shutdown during transearth injection burn been investigated?

   ANSWER - If the problem is a guidance or control problem, the engine can be restarted and controlled manually. If the problem is with the SPS engine itself, nothing can be done. This is one of the accepted risks in the program.

4. Dr. Haeussermann - Have we looked into using a DPS-SPS combination?

   ANSWER - No. The main reason for using the DPS as backup is because of an SPS failure.

5. Mr. Richter - Can't we use ADS some of the time - isn't it load dependent?

   ANSWER - The control authority is marginal in most cases for the APD since the APS engine is not gimballed.

6. Mr. Richter - Can we use two SPS burns for fast earth return transfer?

   ANSWER - This is being looked into. This may not be a desirable way to achieve a gain in return time, since it could be dangerous. If the SPS did not fire the second time, you might exceed acceptable entry velocity or conditions.

7. Mr. Green - When the DPS is used for backup propulsion, do we jettison the SPS propellants?

   ANSWER - There is no capability to jettison SPS propellants.

8. Dr. Mueller - Have the procedures, etc., have been worked out for all of the abort possibilities?

   ANSWER - No.

9. Dr. Von Braun - Why the difference in the translunar and transearth transit times?

   ANSWER - The translunar phase uses a free return trajectory which limits the transfer time to a narrow band. The transearth time is primarily limited by the energy available and the amount of consumables remaining.

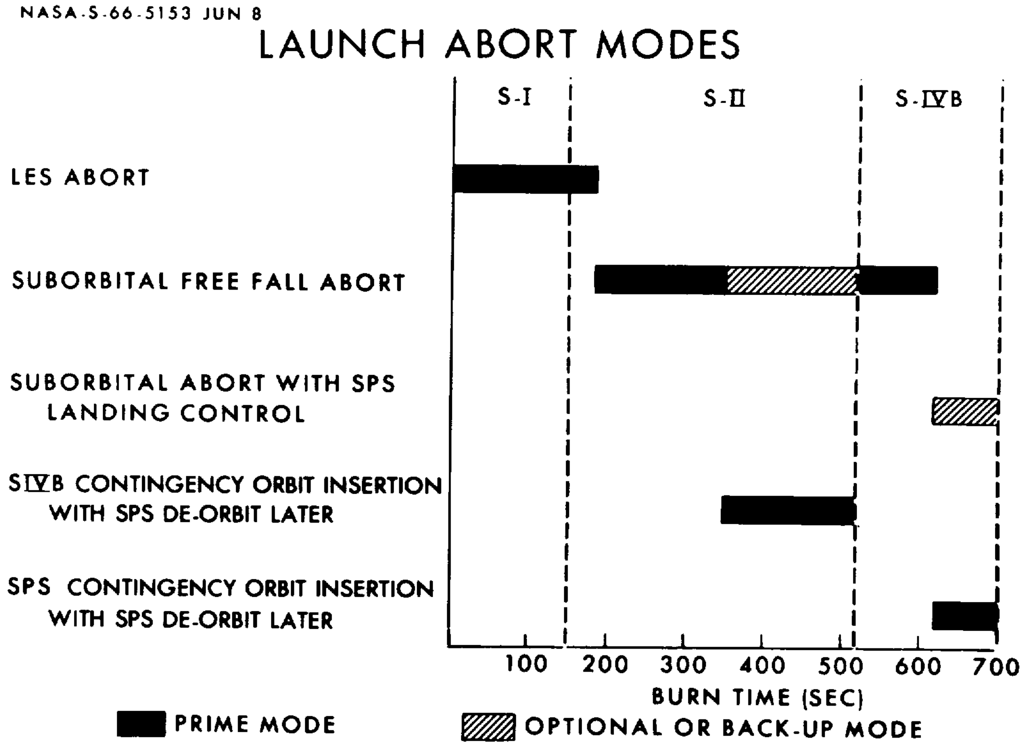

NASA-S-66-5153 JUN 8
LAUNCH ABORT MODES
S-I
S-II
S-IVB
LES ABORT
SUBORBITAL FREE FALL ABORT
SUBORBITAL ABORT WITH SPS LANDING CONTROL
SIVB CONTINGENCY ORBIT INSERTION WITH SPS DE-ORBIT LATER
SPS CONTINGENCY ORBIT INSERTION WITH SPS DE-ORBIT LATER
100
200
300
400
500
600
700
BURN TIME (SEC)
PRIME MODE
OPTIONAL OR BACK-UP MODE

Fig. 1

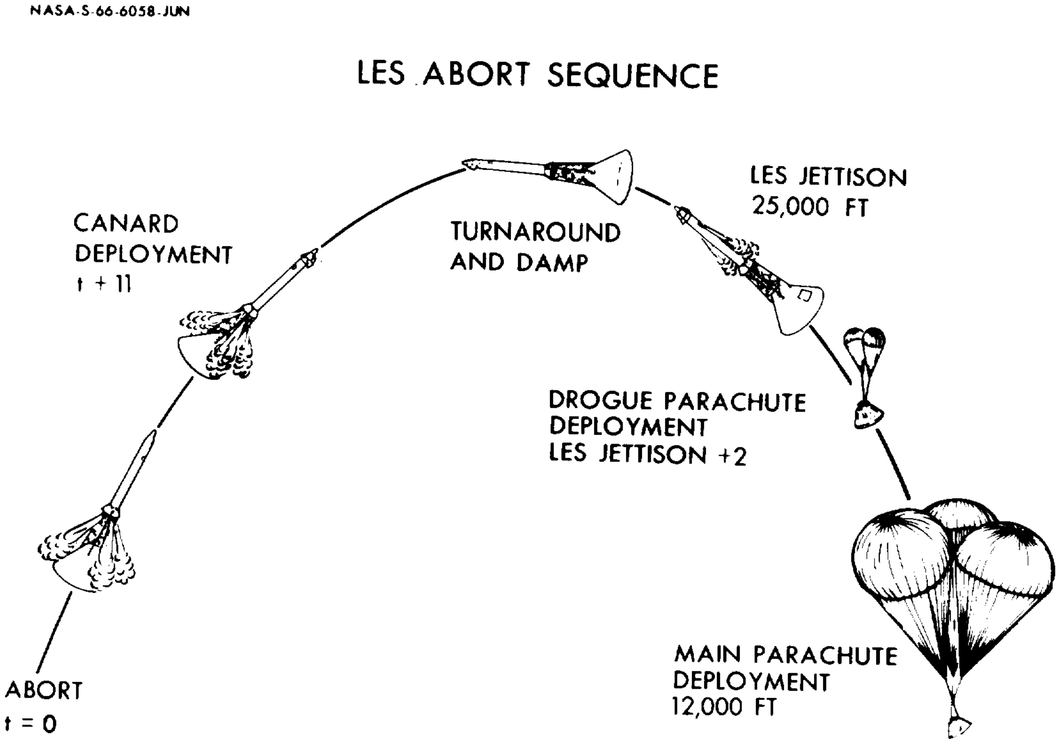

NASA-S-66-6058-JUN
LES ABORT SEQUENCE
CANARD DEPLOYMENT t + 11
TURNAROUND AND DAMP
LES JETTISON 25,000 FT
DROGUE PARACHUTE DEPLOYMENT LES JETTISON +2
MAIN PARACHUTE DEPLOYMENT 12,000 FT
ABORT t = 0

Fig. 2

NASA-S-66-6059-JUN

SUBORBITAL ABORT WITH NO SPS LANDING AREA CONTROL

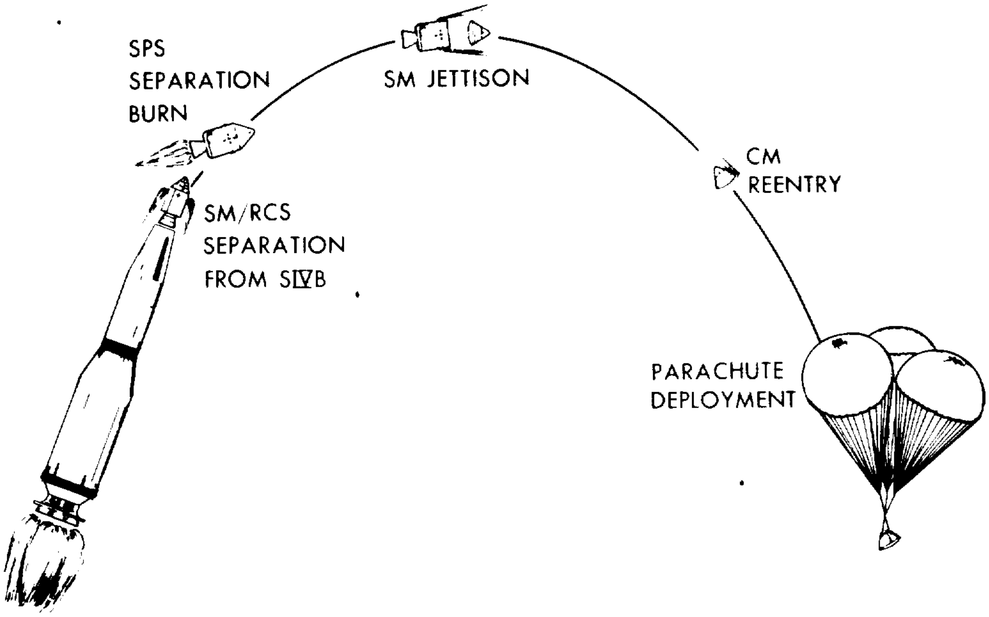

SPS
SEPARATION
BURN
SM JETTISON
CM
REENTRY
SM/RCS
SEPARATION
FROM SIVB
PARACHUTE
DEPLOYMENT

Fig. 3

NASA-S-66-6803-JUN

SUBORBITAL ABORT WITH SPS LANDING AREA CONTROL

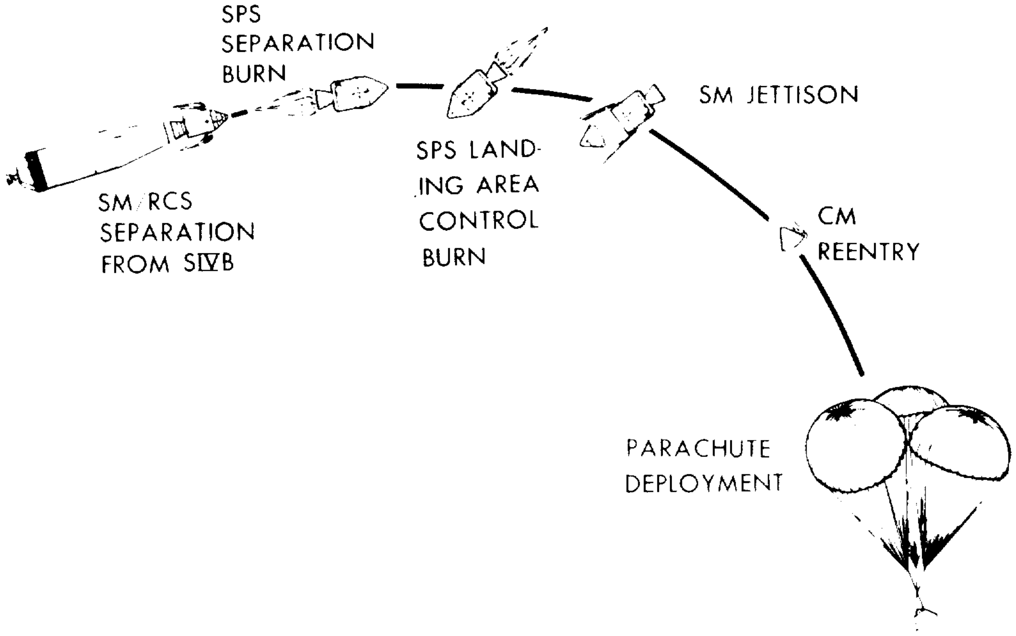

SPS
SEPARATION
BURN
SM JETTISON
SPS LANDING AREA
CONTROL
BURN
SM/RCS
SEPARATION
FROM SIVB
CM
REENTRY
PARACHUTE
DEPLOYMENT

Fig. 4

NASA-S-66-6486 JUN

S-IVB OR SPS CONTINGENCY ORBIT INSERTION FOLLOWED BY SPS DE-ORBIT TO REENTRY

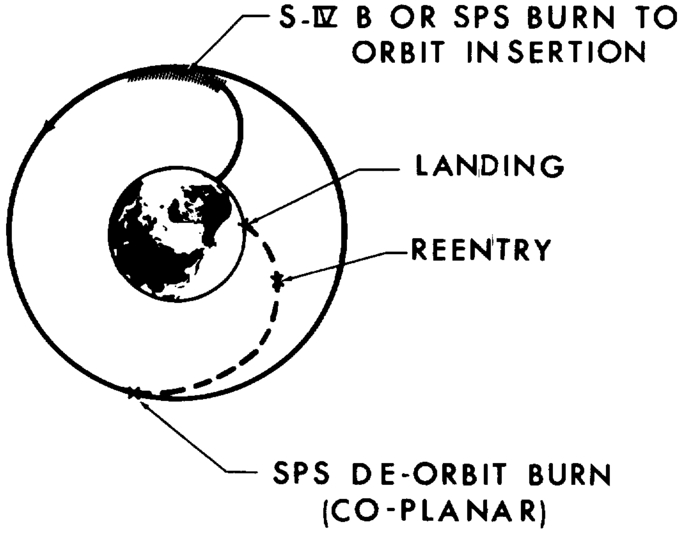

S-IV B OR SPS BURN TO ORBIT INSERTION
LANDING
REENTRY
SPS DE-ORBIT BURN (CO-PLANAR)

Fig. 5

NASA-S-66-6415 JUN

EARTH PARKING ORBIT ABORT MODES

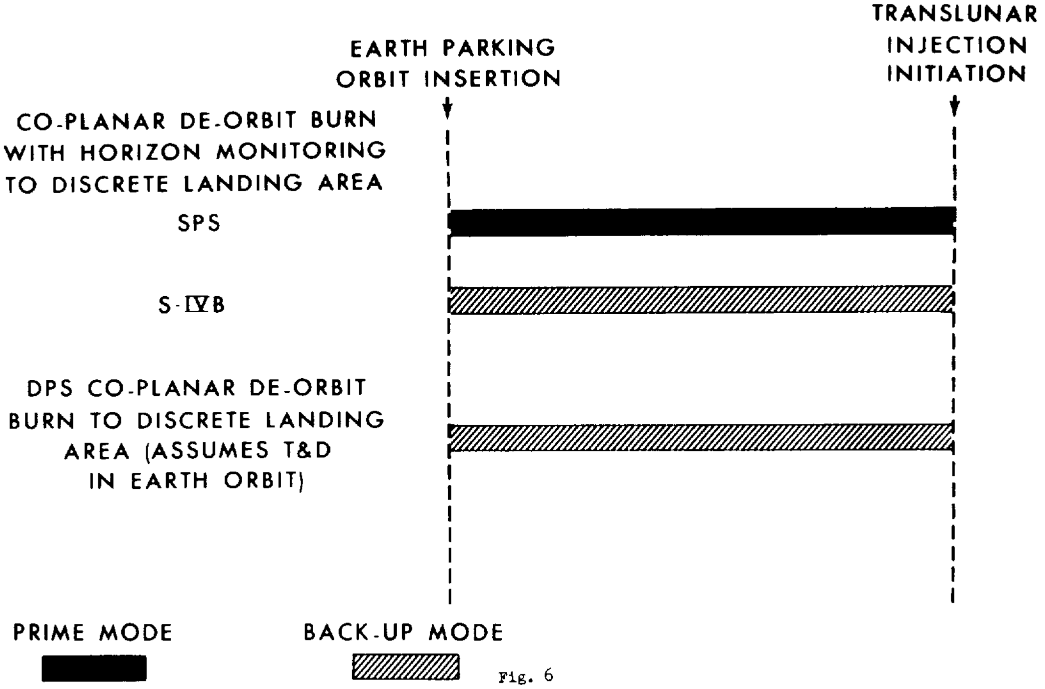


Fig. 6

NASA-S-66-4987 MAY 26

SPS OR S-IVB DE-ORBIT ABORT FROM EARTH PARKING ORBIT

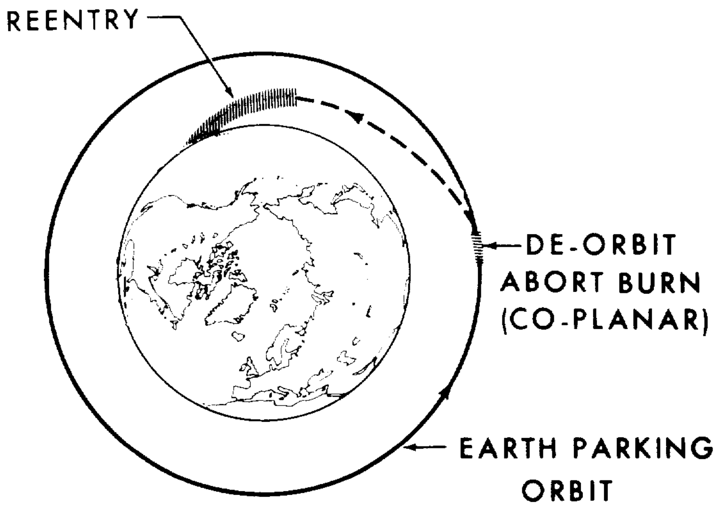

REENTRY
DE-ORBIT ABORT BURN (CO-PLANAR)
EARTH PARKING ORBIT

Fig. 7

-66-6499 JUN

LM DPS DE-ORBIT FROM EARTH PARKING ORBIT

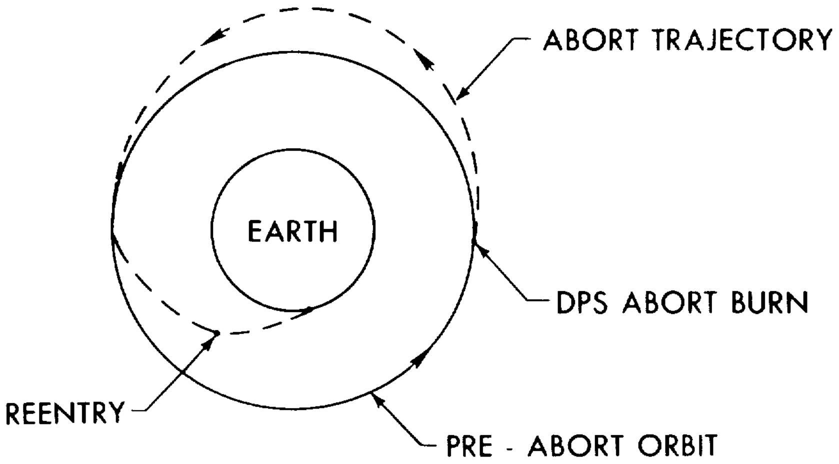

ABORT TRAJECTORY
EARTH
DPS ABORT BURN
REENTRY
PRE - ABORT ORBIT

Fig. 8

NASA-S-66-4977 MAY 25

PRE-ABORT ORBITS DUE TO PREMATURE BURN-OUT DURING TRANSLUNAR INJECTION

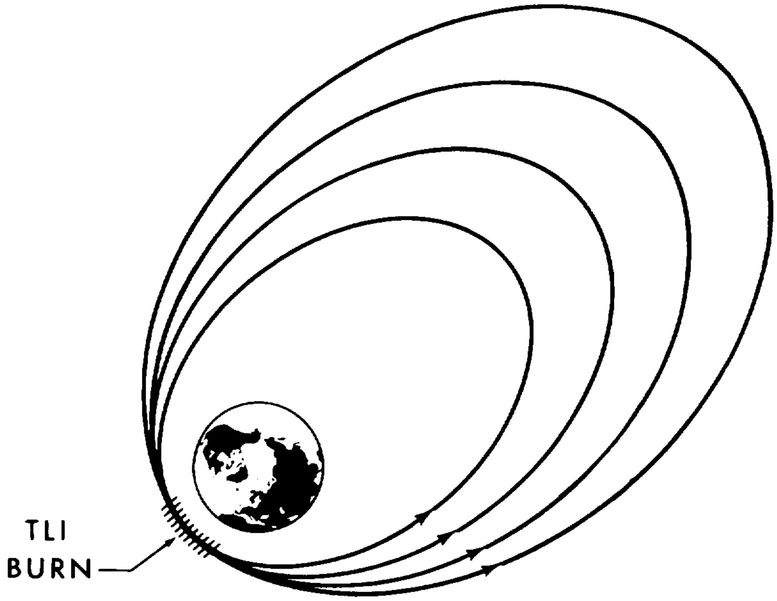


Fig. 9

NASA-S-66-6397 JUN

THE ORBITAL PERIOD AS A FUNCTION OF S-IVB BURN TIME

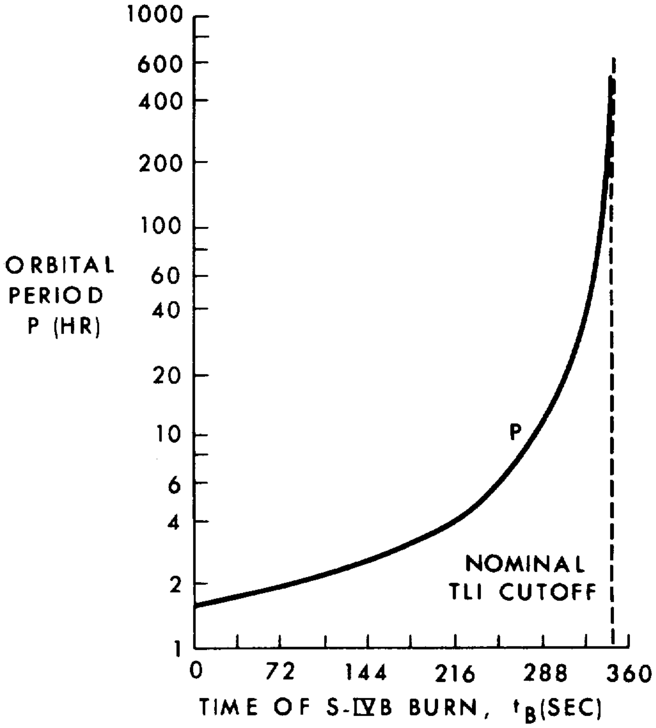


Fig. 10

NASA-S-66-6410 JUN

TRANSLUNAR INJECTION ABORT MODES

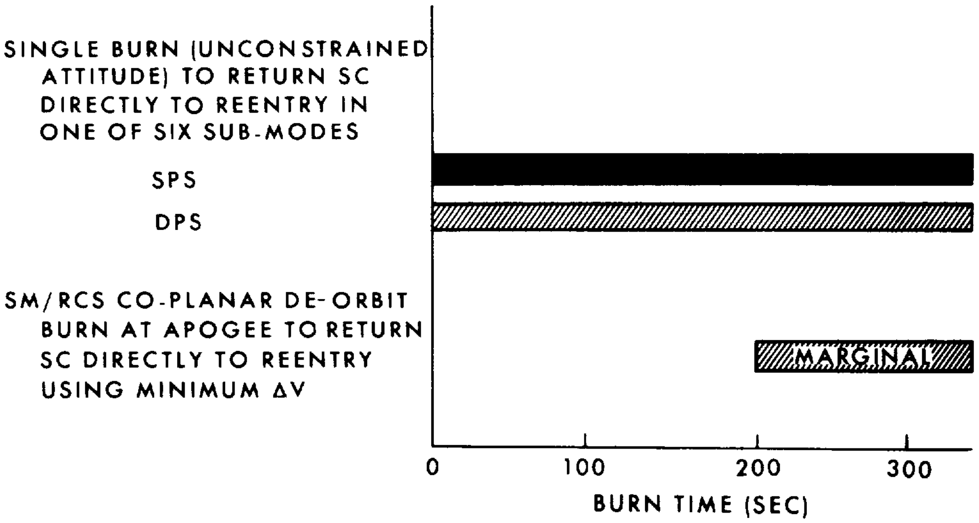


Fig. 11

NASA-S-66-6500 JUN

ABORTS FROM TRAJECTORIES RESULTING FROM TLI FAILURE

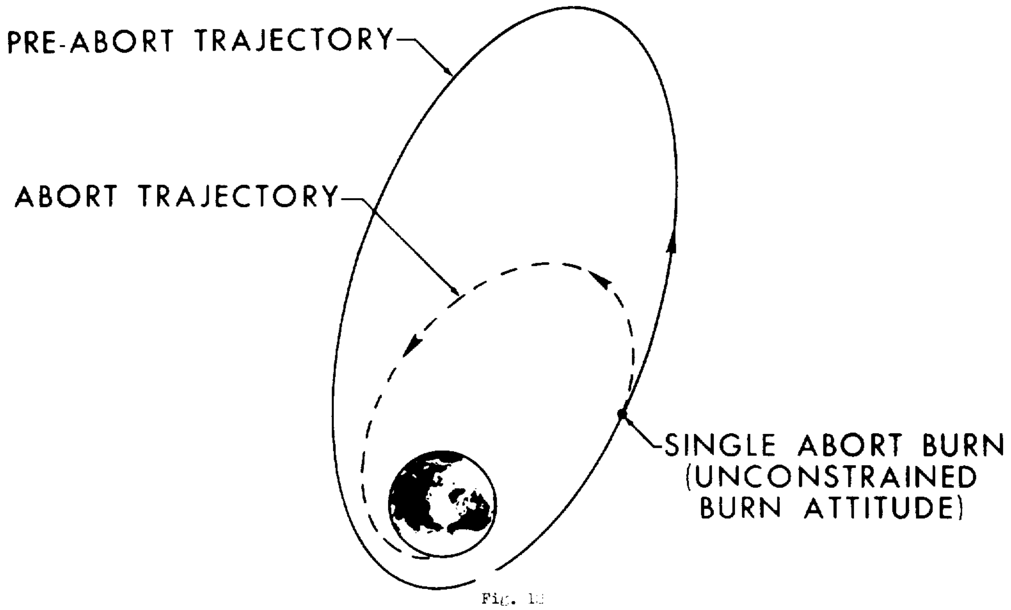

PRE-ABORT TRAJECTORY
ABORT TRAJECTORY
SINGLE ABORT BURN (UNCONSTRAINED BURN ATTITUDE)

Fig. 12

NASA-S-66-4994 MAY 26

FUEL CRITICAL ABORT FOLLOWING TLI FAILURE

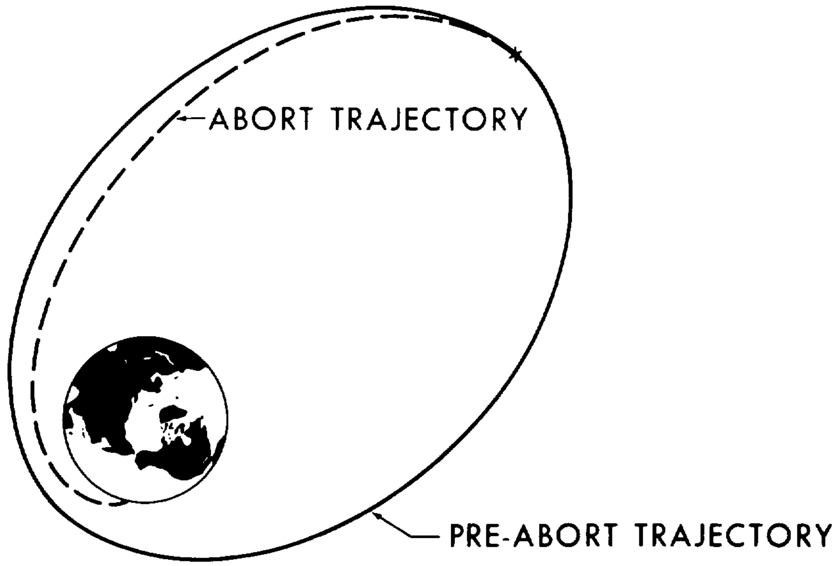

ABORT TRAJECTORY
PRE-ABORT TRAJECTORY

Fig. 13

NASA-S-66-4982 MAY 25

FUEL CRITICAL ΔV REQUIREMENTS FOR ABORTS FOLLOWING PREMATURE SHUTDOWN DURING TLI

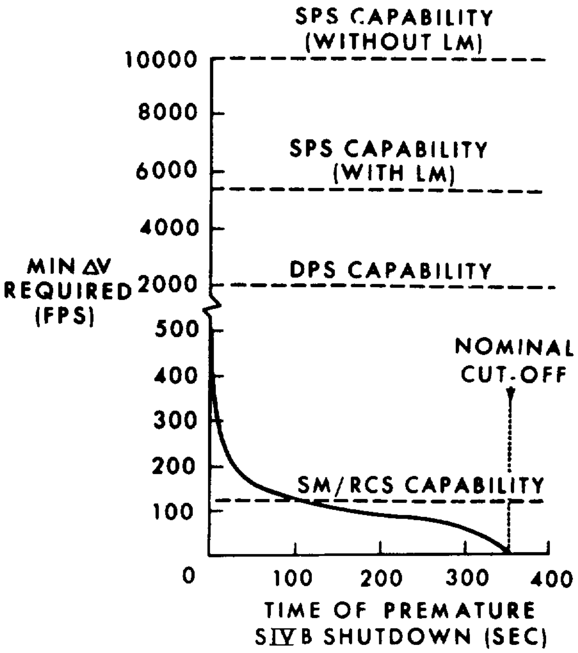


Fig. 14

NASA-S-66-4984 MAY 25

TIME CRITICAL ABORT FOLLOWING TLI FAILURE

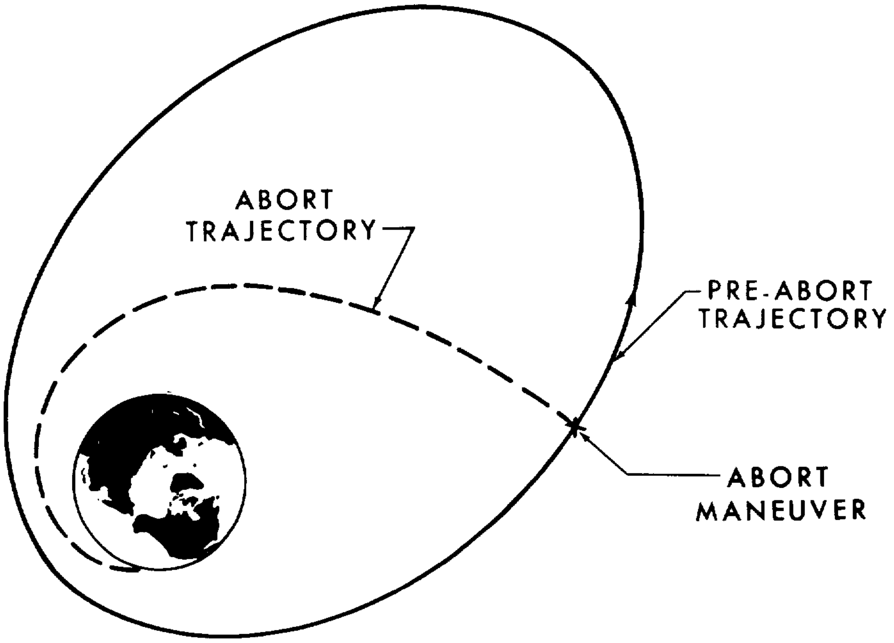

ABORT TRAJECTORY
PRE-ABORT TRAJECTORY
ABORT MANEUVER

Fig. 15

NASA-S-66-4975 MAY 25

TIME CRITICAL RETURN TIMES FOR ABORTS FOLLOWING PREMATURE SHUTDOWN DURING TLI & ASSUMING SPS AVAILIBLE ($\Delta V = 10,000$ FPS)

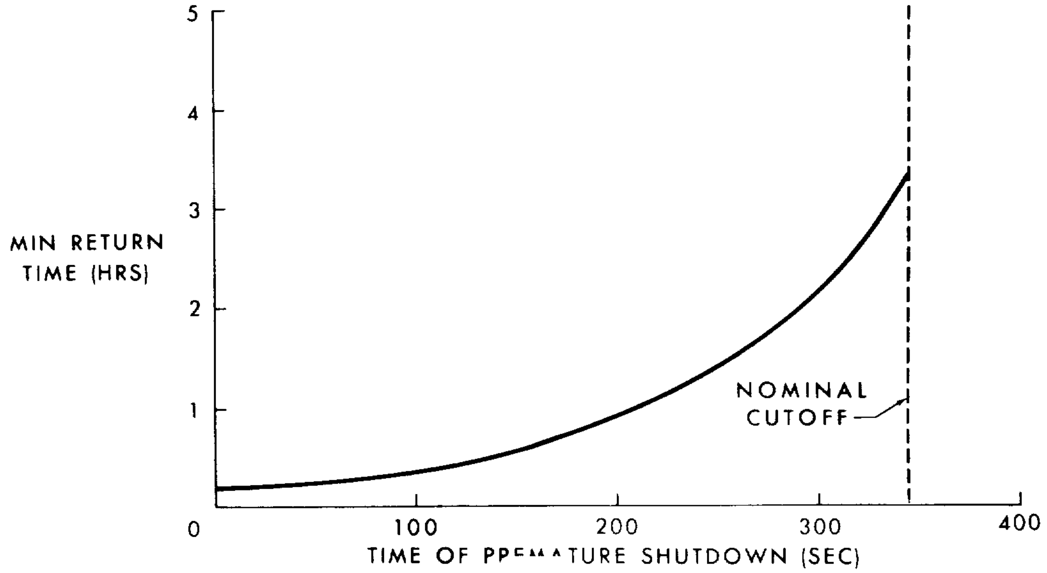


Fig. 16

NASA-S-66-4990 MAY 26

FUEL CRITICAL ABORT VELOCITY INCREMENTS, ΔV(FT/SEC) - BERMUDA

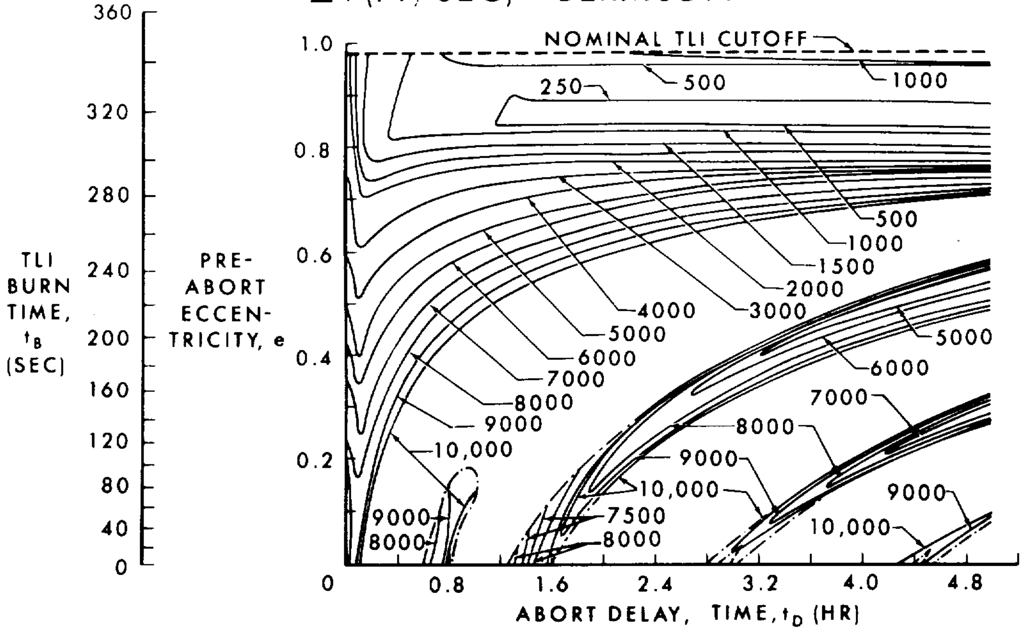


Fig. 17

NASA-S-66-5000 MAY 26

TIME CRITICAL TOTAL TRIP TIME REGIONS

BERMUDA – ΔV MAX = 10,000 FPS (ENTIRE SPS CAPABILITY)

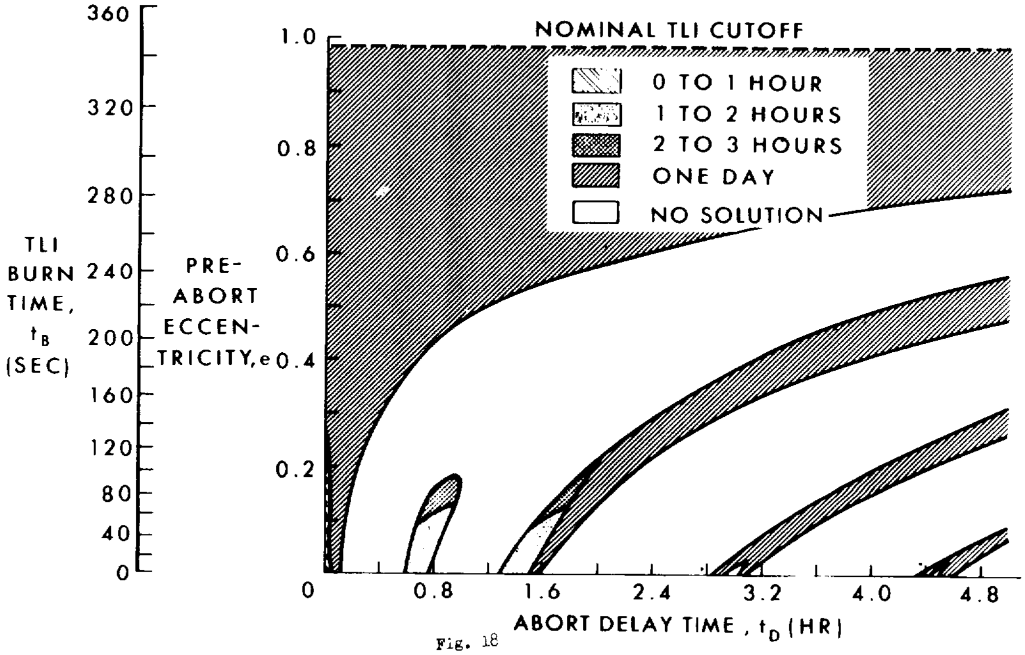


Fig. 18

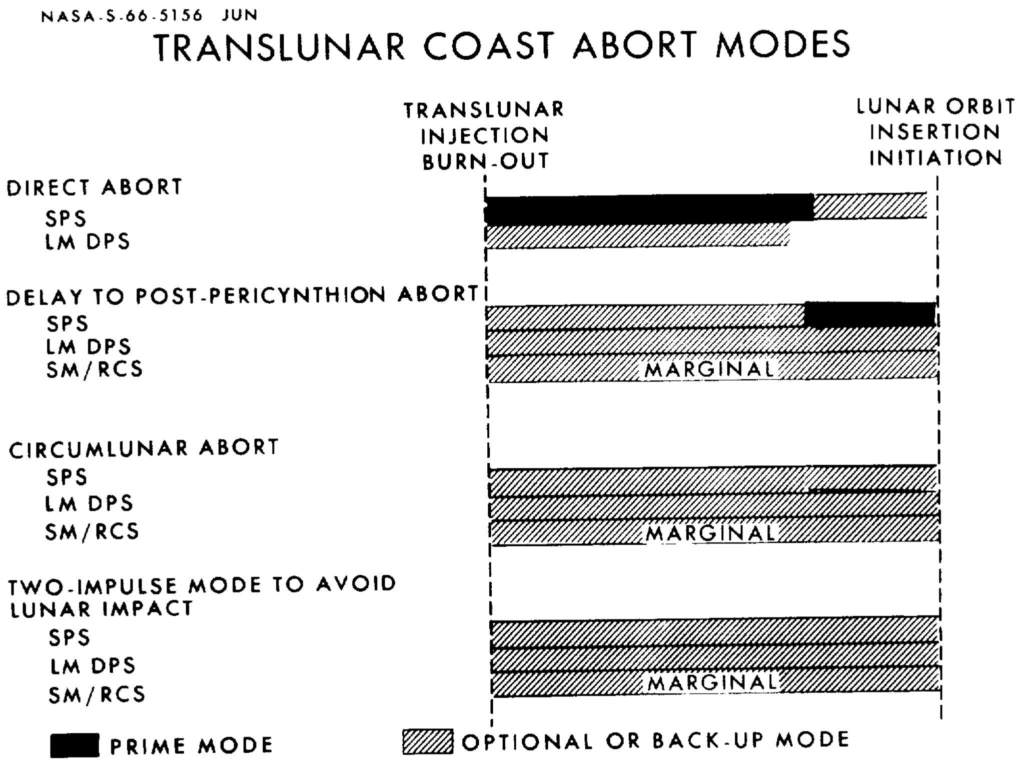

NASA-S-66-5156 JUN
TRANSLUNAR COAST ABORT MODES
TRANSLUNAR INJECTION BURN-OUT
LUNAR ORBIT INSERTION INITIATION
DIRECT ABORT
SPS
LM DPS
DELAY TO POST-PERICYNTHION ABORT
SPS
LM DPS
SM/RCS
MARGINAL
CIRCUMLUNAR ABORT
SPS
LM DPS
SM/RCS
MARGINAL
TWO-IMPULSE MODE TO AVOID LUNAR IMPACT
SPS
LM DPS
SM/RCS
MARGINAL
■ PRIME MODE
▨ OPTIONAL OR BACK-UP MODE

Fig. 19

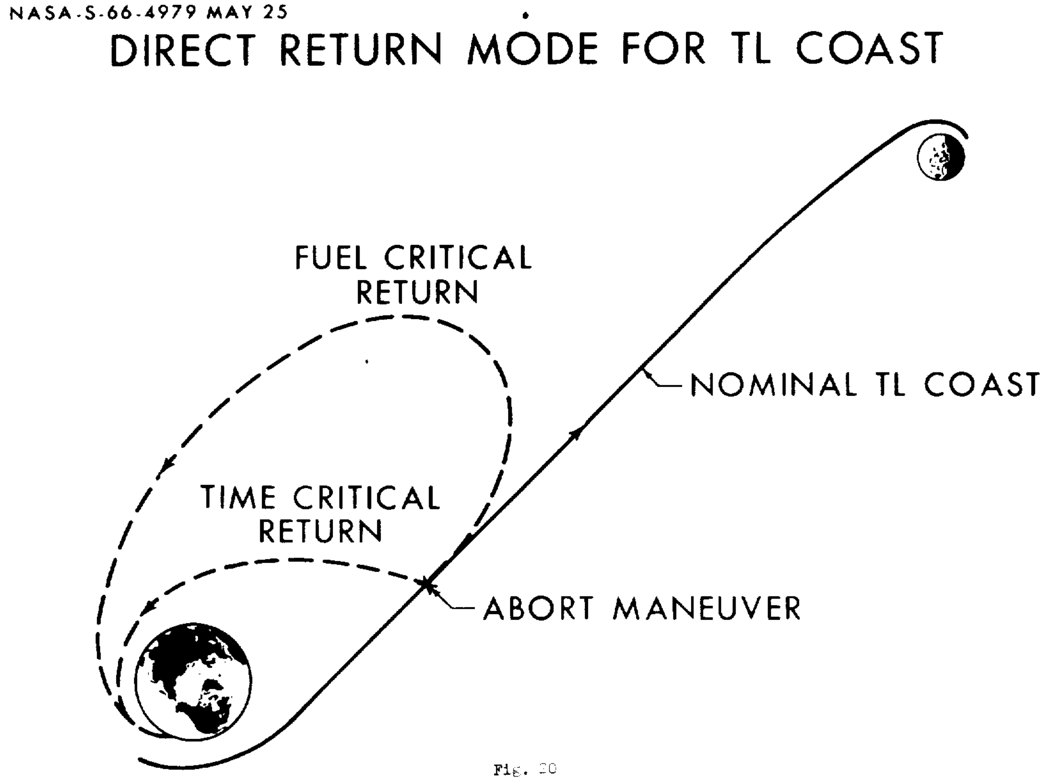

NASA-S-66-4979 MAY 25
DIRECT RETURN MODE FOR TL COAST
FUEL CRITICAL RETURN
NOMINAL TL COAST
TIME CRITICAL RETURN
ABORT MANEUVER

Fig. 20

NASA-S-66-4993 MAY 26

DELAY ABORT TO POST-PERICYNTHION MODE FOR TL COAST

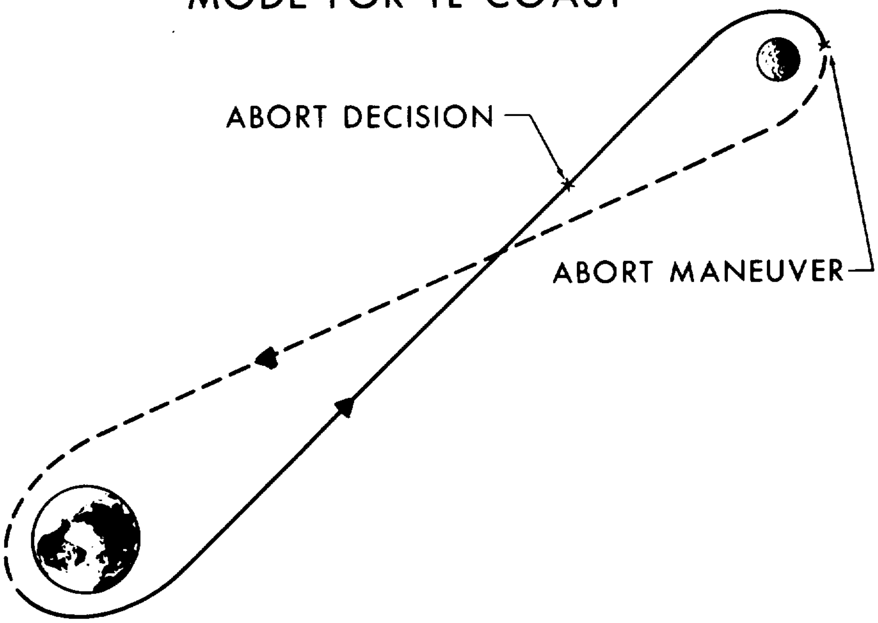

ABORT DECISION
ABORT MANEUVER

Fig. 21

NASA-S-66-4996 MAY 26

CIRCUMLUNAR ABORT MODE FOR TL COAST

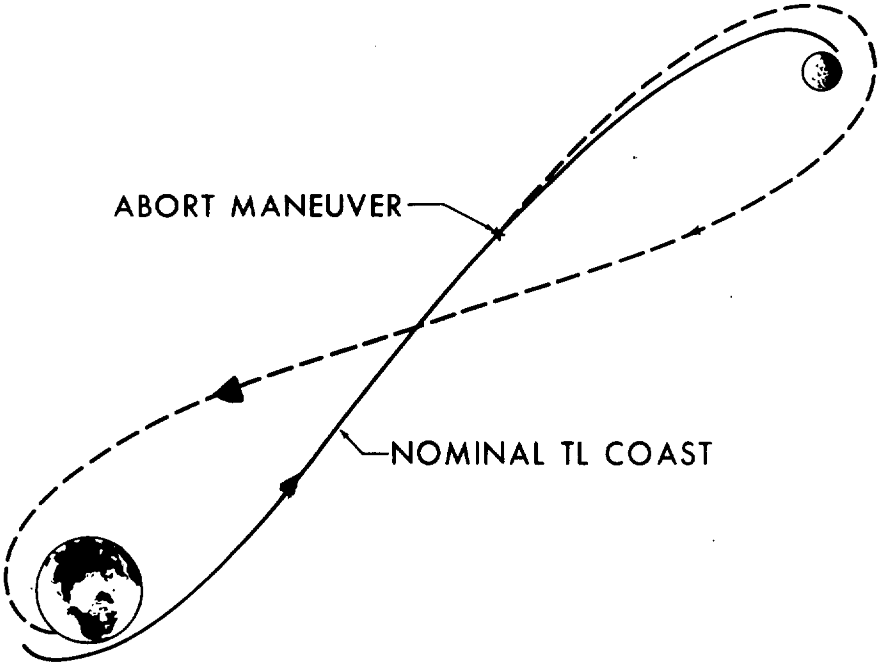

ABORT MANEUVER
NOMINAL TL COAST

Fig. 22

NASA-S-66-5001 MAY 26

CORRECT BACK TO NOMINAL FREE RETURN MODE

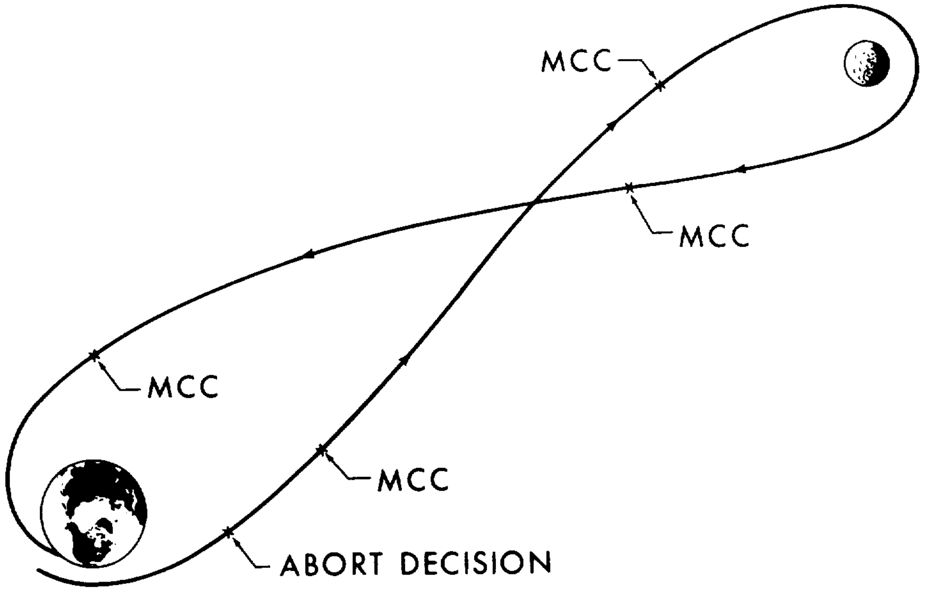

MCC
MCC
MCC
MCC
ABORT DECISION

Fig. 23

NASA-S-66-5002 MAY 26

TWO-BURN MODE TO AVOID LUNAR IMPACT

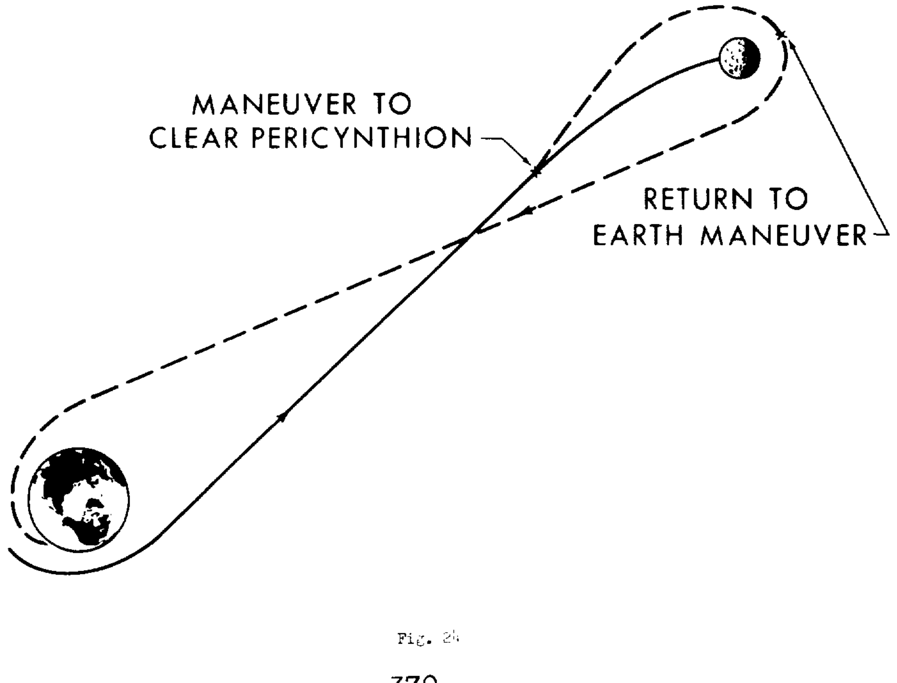

MANEUVER TO
CLEAR PERICYNTHION
RETURN TO
EARTH MANEUVER

Fig. 24

NASA-S-66-4997 MAY 26

ABORT TRANSFER ANGLE FOR TL COAST

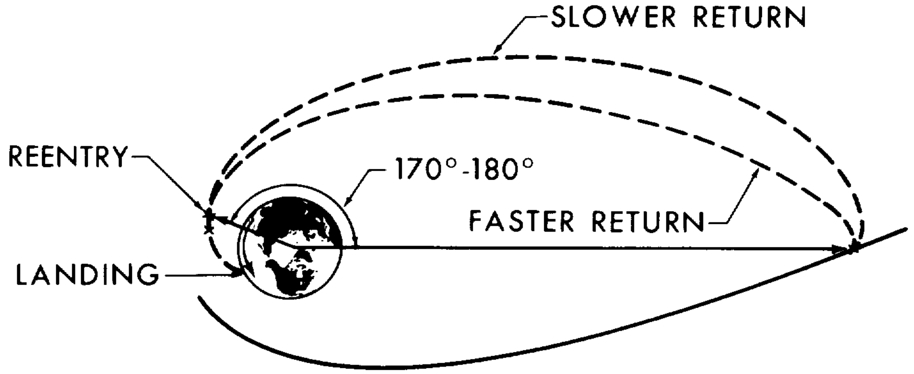

SLOWER RETURN
REENTRY
170°-180°
FASTER RETURN
LANDING

Fig. 25

NASA-S-66-4998 MAY 26

'24 HR' EFFECT FOR ABORTS TO SPECIFIC LANDING SITES

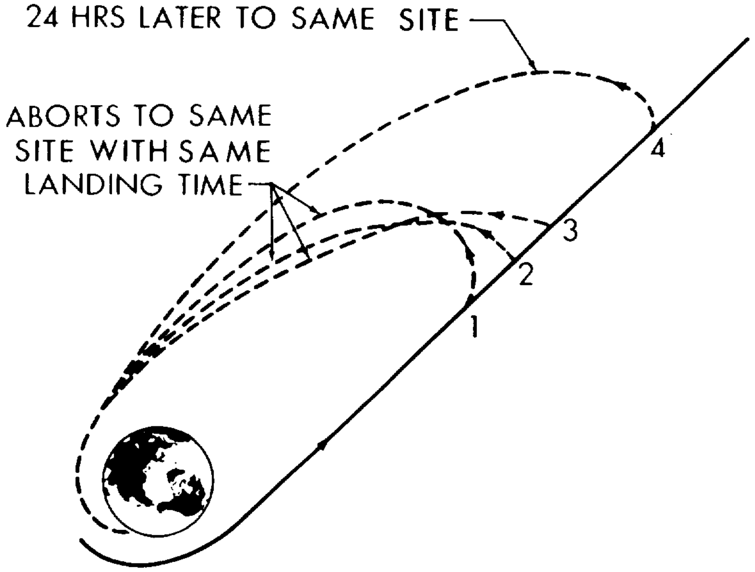

24 HRS LATER TO SAME SITE
ABORTS TO SAME SITE WITH SAME LANDING TIME
1
2
3
4

Fig. 26

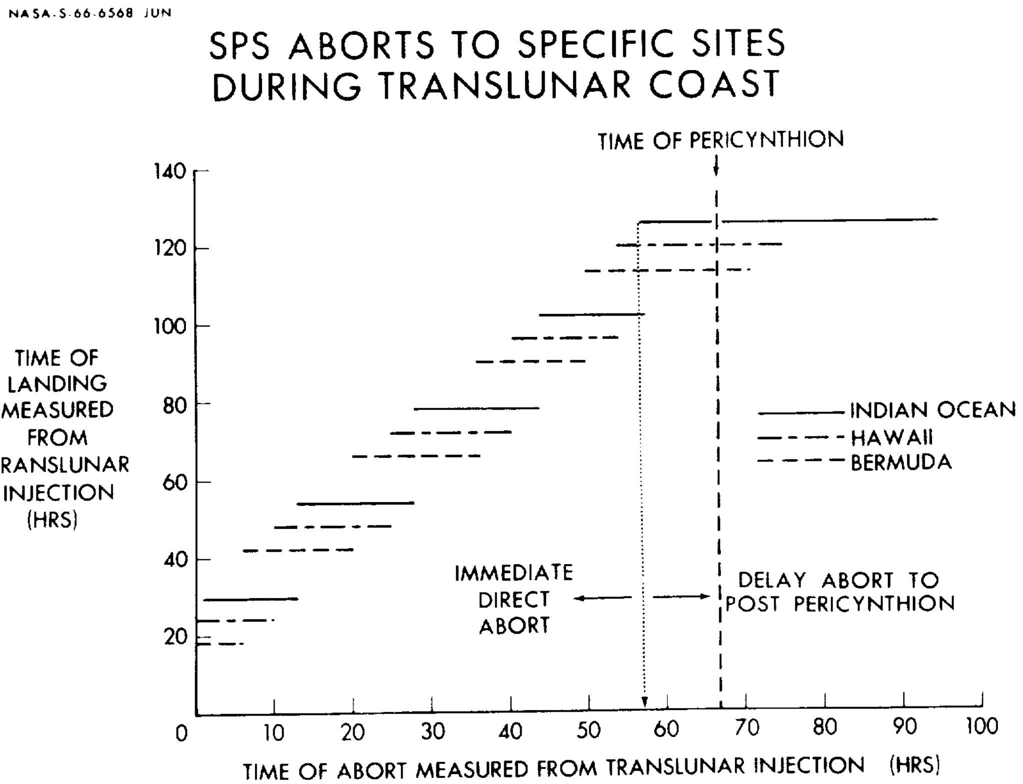

NASA-S-66-6568 JUN
SPS ABORTS TO SPECIFIC SITES DURING TRANSLUNAR COAST
TIME OF PERICYNTHION
TIME OF LANDING MEASURED FROM TRANSLUNAR INJECTION (HRS)
0
20
40
60
80
100
120
140
10
20
30
40
50
60
70
80
90
100
TIME OF ABORT MEASURED FROM TRANSLUNAR INJECTION (HRS)
INDIAN OCEAN
HAWAII
BERMUDA
IMMEDIATE DIRECT ABORT
DELAY ABORT TO POST PERICYNTHION

Fig. 27

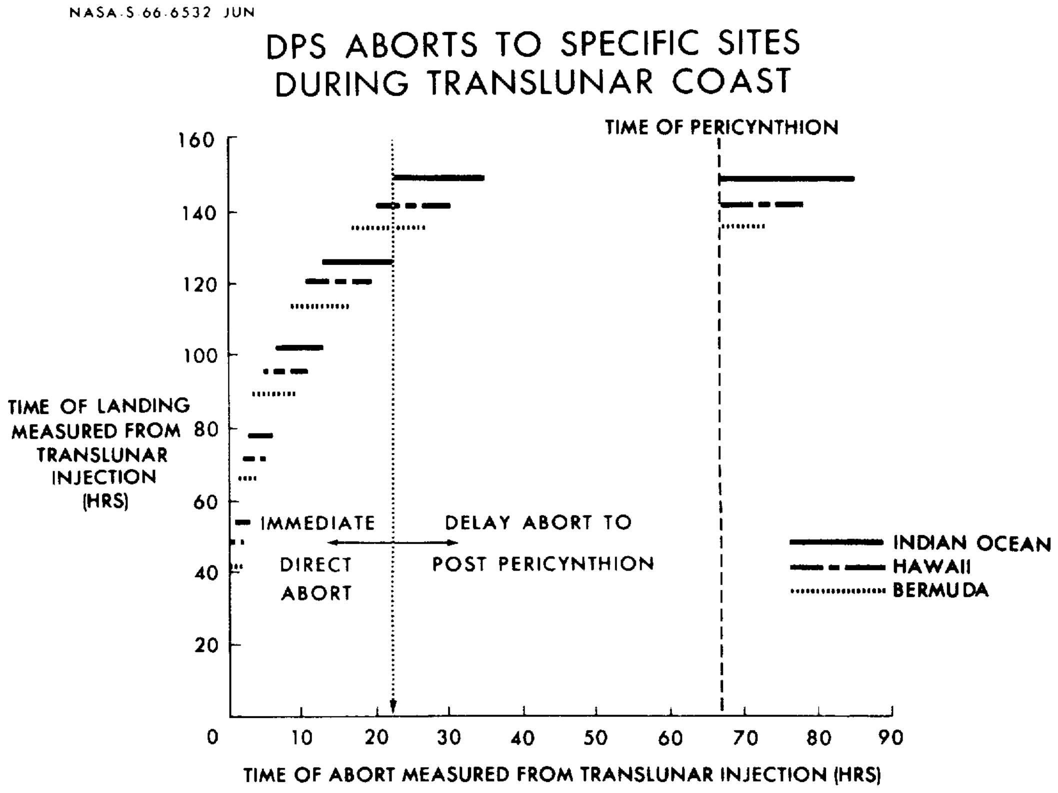

NASA-S-66-6532 JUN
DPS ABORTS TO SPECIFIC SITES DURING TRANSLUNAR COAST
TIME OF PERICYNTHION
TIME OF LANDING MEASURED FROM TRANSLUNAR INJECTION (HRS)
0
20
40
60
80
100
120
140
160
10
20
30
40
50
60
70
80
90
TIME OF ABORT MEASURED FROM TRANSLUNAR INJECTION (HRS)
IMMEDIATE DIRECT ABORT
DELAY ABORT TO POST PERICYNTHION
INDIAN OCEAN
HAWAII
BERMUDA

Fig. 28

NASA-S-66-6501 JUN

ABORT RETURN GEOMETRY

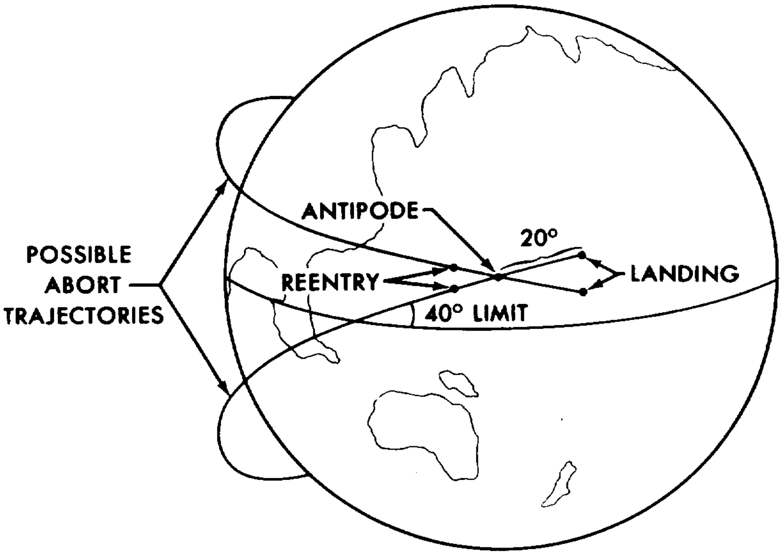

ANTIPODE
20°
POSSIBLE
ABORT
TRAJECTORIES
REENTRY
LANDING
40° LIMIT

Fig. 29

NASA-S-66-4999 MAY 26

TYPICAL LANDING AREA ACCESSIBILITY FOR ABORTS FROM TL COAST

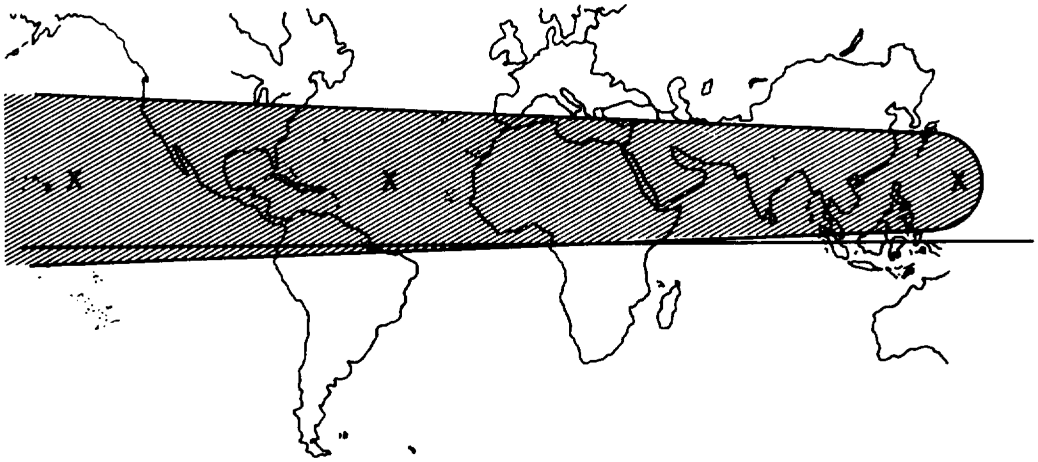


Fig. 30

NASA-S-66-4983 MAY 25

LOI PREMATURE SHUTDOWN TRAJECTORIES

PRE-ABORT

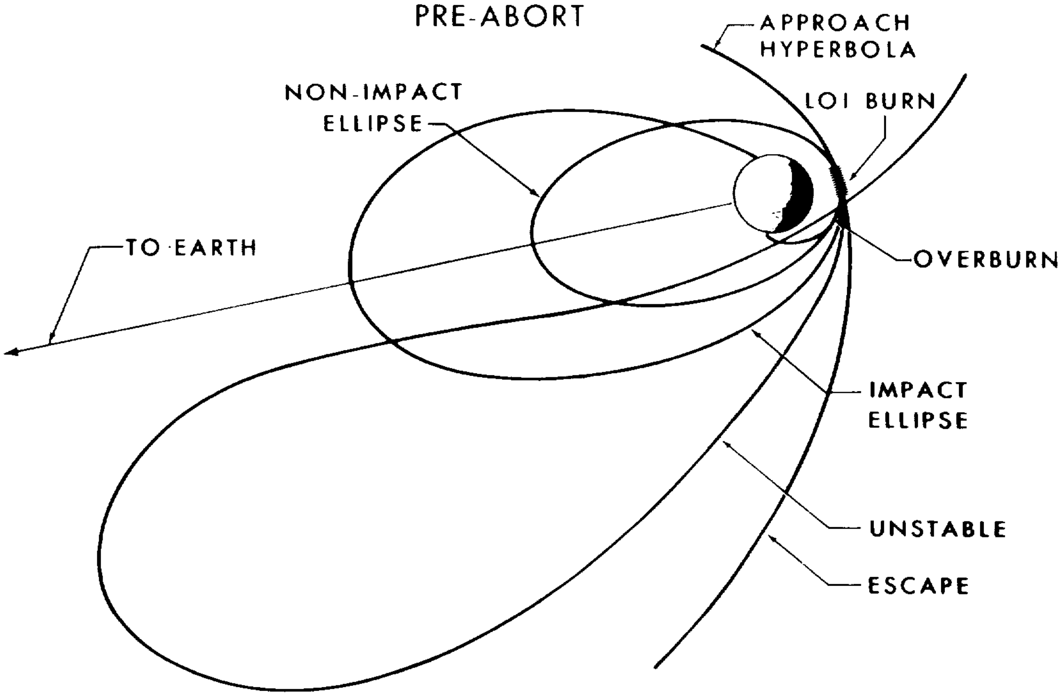

APPROACH HYPERBOLA
NON-IMPACT ELLIPSE
LOI BURN
TO EARTH
OVERBURN
IMPACT ELLIPSE
UNSTABLE
ESCAPE

Fig. 31

NASA-S-66-6395 JUN

PRE-ABORT TRAJECTORY FAMILIES DURING LOI

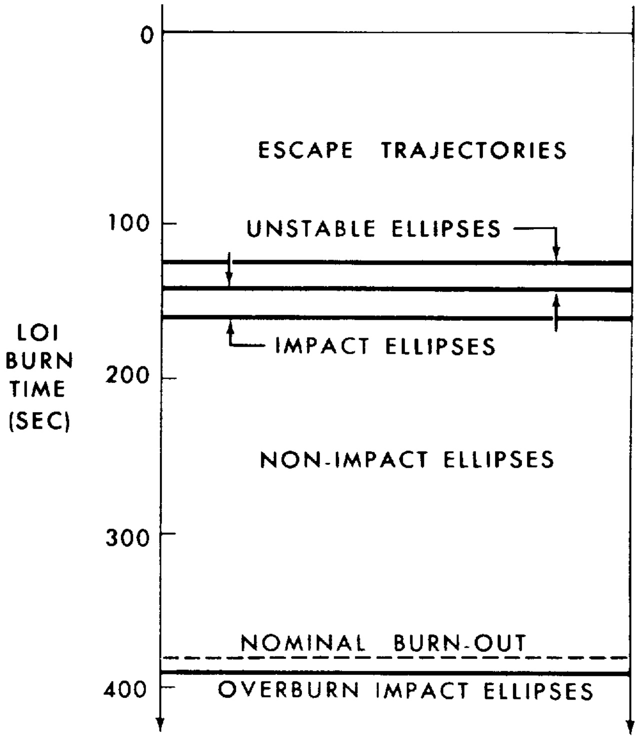
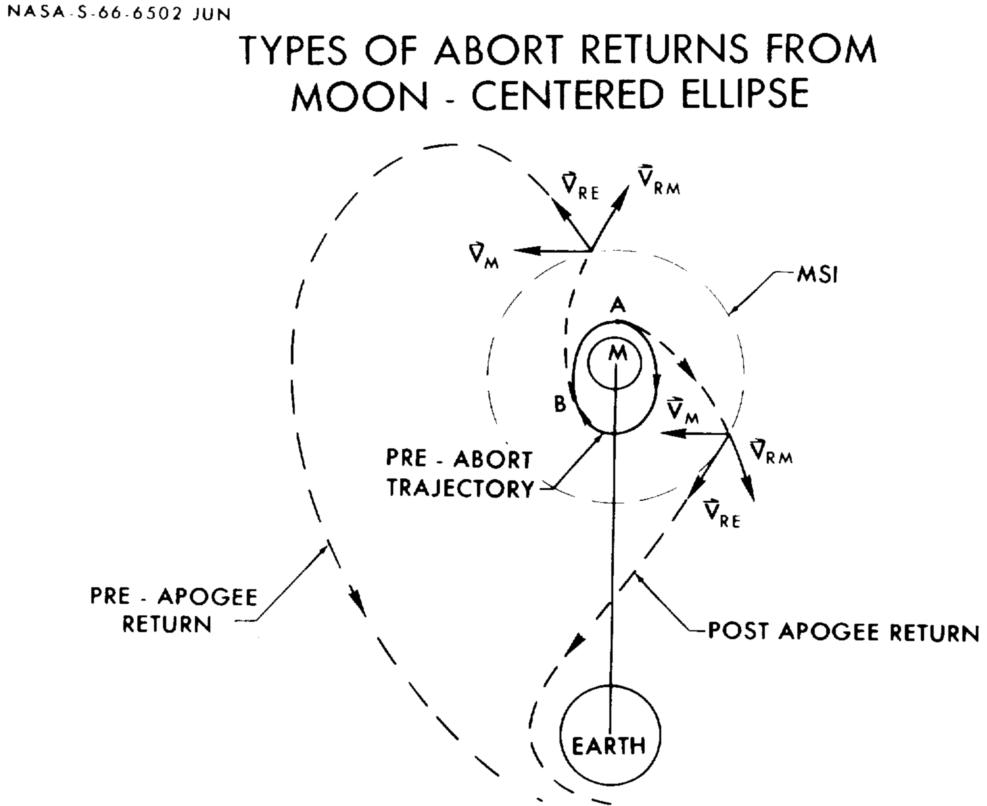


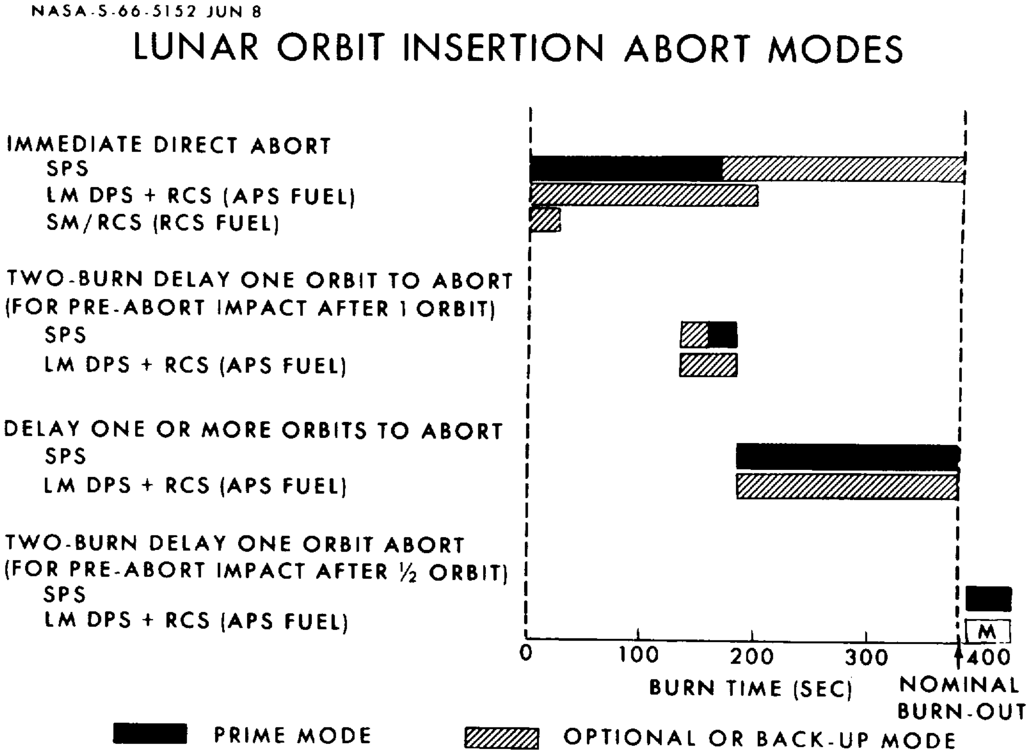
Fig. [illegible]


NASA-S-66-6502 JUN

TYPES OF ABORT RETURNS FROM MOON - CENTERED ELLIPSE

$\vec{V}_{RE}$
$\vec{V}_{RM}$
$\vec{V}_M$
MSI
A
M
B
$\vec{V}_M$
$\vec{V}_{RM}$
PRE - ABORT TRAJECTORY
$\vec{V}_{RE}$
PRE - APOGEE RETURN
POST APOGEE RETURN
EARTH

Fig. 33


NASA-S-66-5152 JUN 8

LUNAR ORBIT INSERTION ABORT MODES

Fig. 34

NASA-S-66-4988 MAY 26

IMMEDIATE DIRECT ABORT MODE

MANDATORY FOR ESCAPE OR UNSTABLE PRE-ABORT
TRAJECTORIES OPTIONAL FOR OTHERS

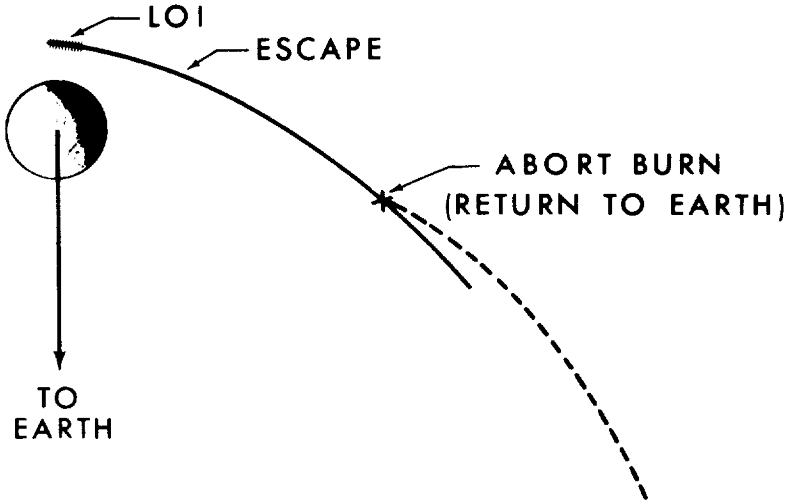

LOI
ESCAPE
ABORT BURN
(RETURN TO EARTH)
TO
EARTH

Fig. 35

NASA-S-66-4991 MAY 26

TWO-BURN DELAY ONE ORBIT MODE

FOR PRE-ABORT IMPACT ELLIPSES

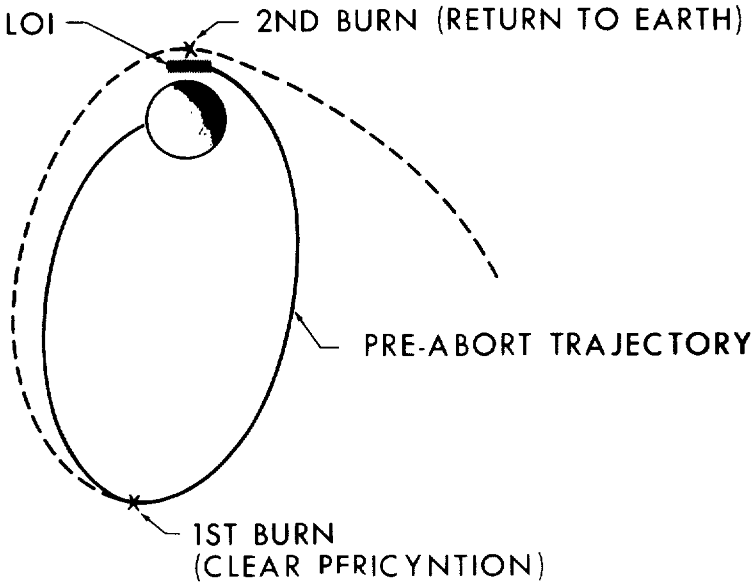

LOI
2ND BURN (RETURN TO EARTH)
PRE-ABORT TRAJECTORY
1ST BURN
(CLEAR PFRICYNTION)

Fig. 36

NASA-S-66-4980 MAY 25

DELAY ONE ORBIT TO ABORT MODE

NON-IMPACT, SHORT PERIOD PRE-ABORT ELLIPSES

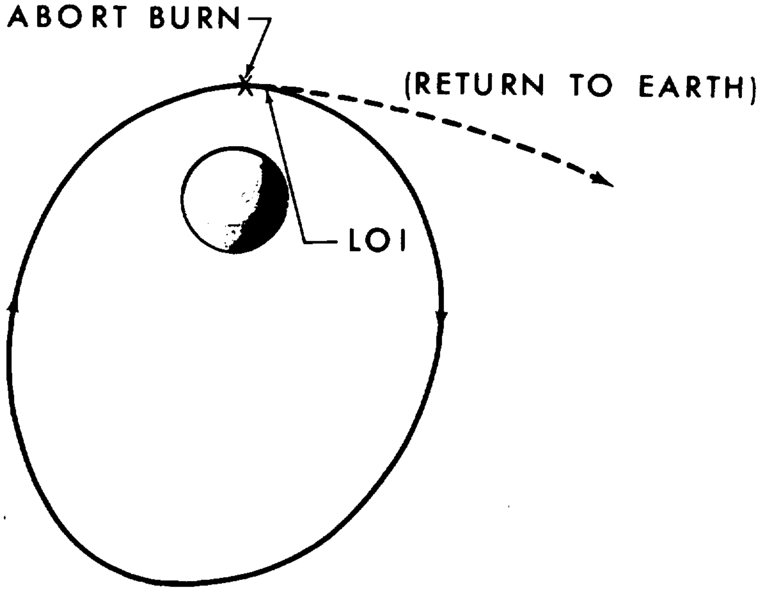

ABORT BURN
(RETURN TO EARTH)
LOI

Fig. 37

NASA-S-66-4989 MAY 26

TWO-BURN DELAY ONE ORBIT MODE

OVERBURN CASES

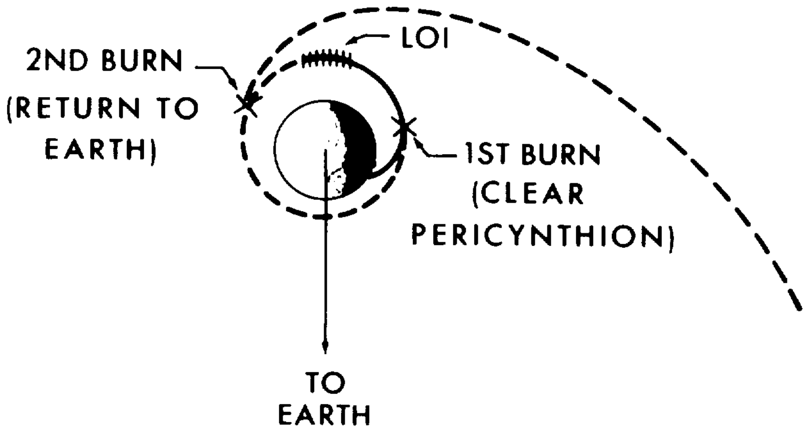
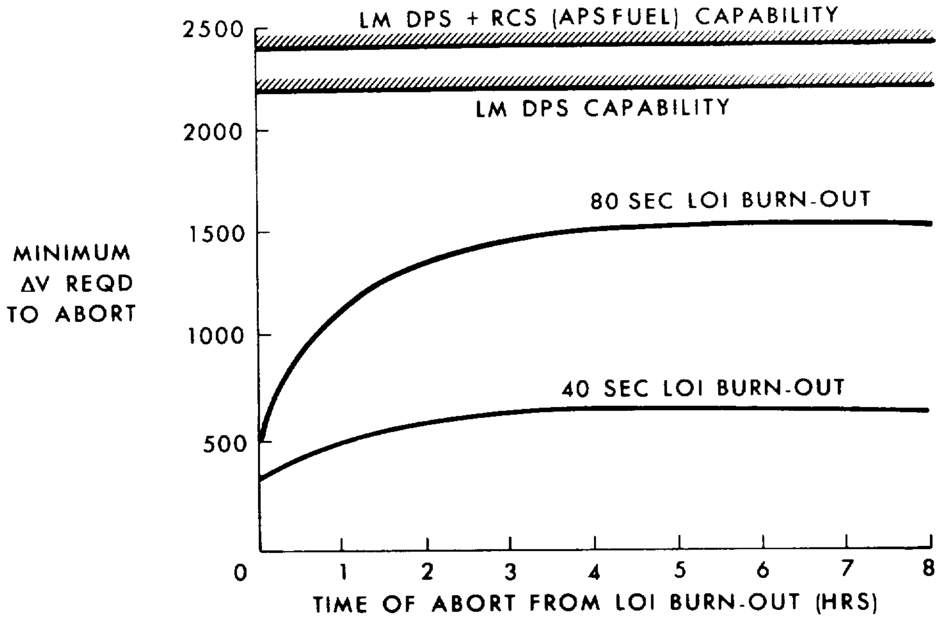
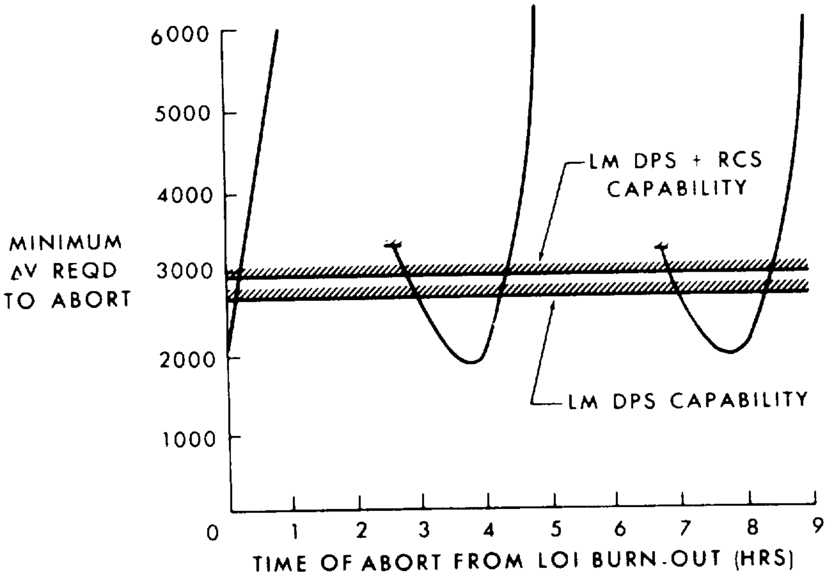

2ND BURN
(RETURN TO EARTH)
LOI
1ST BURN
(CLEAR PERICYNTHION)
TO EARTH

Fig. 38


NASA-S-66-6400 JUN

EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR PRE-ABORT ESCAPE TRAJECTORIES

Fig. 39


NASA-S-66-6399 JUN

EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR PRE-ABORT ELLIPSE

Fig. 40

NASA-S-66-6511 JUN

ABORT CAPABILITY USING LM PROPULSION SYSTEMS IN LUNAR ORBIT INSERTION PHASE

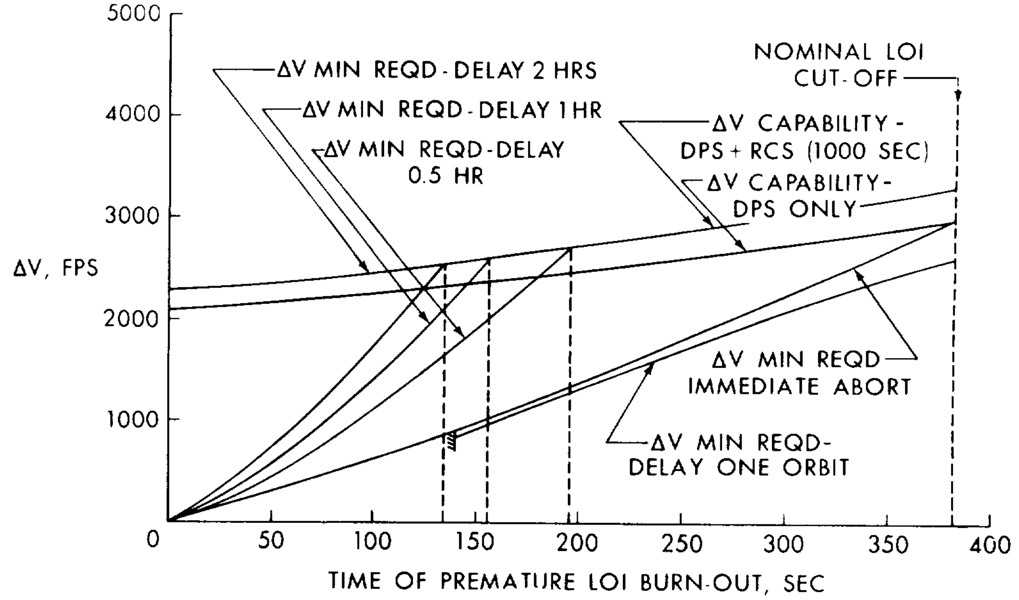


Fig. 41

NASA-S-66-6458 JUN

ABORTS TO SPECIFIC SITES FROM A PRE-ABORT ELLIPSE ABOUT THE MOON

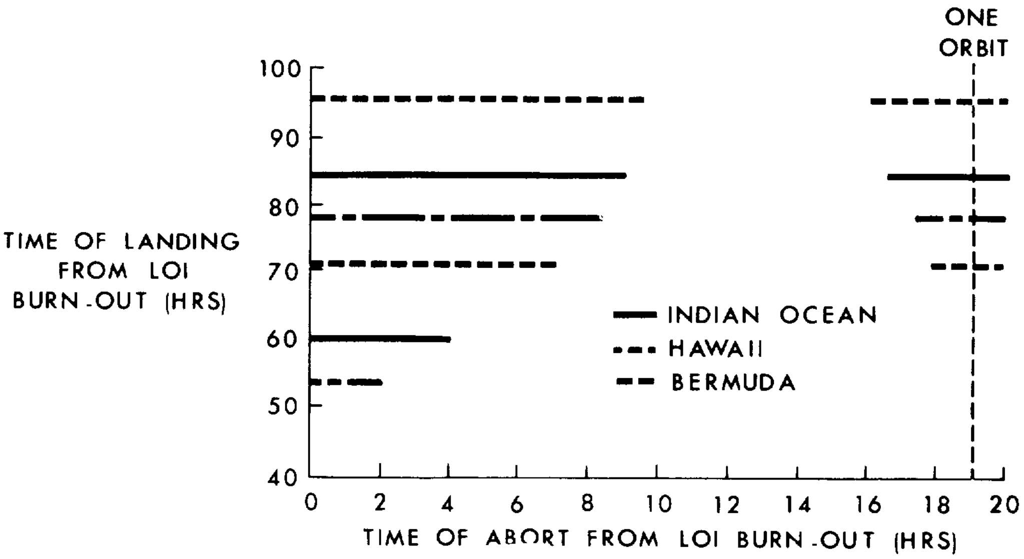


Fig. 42

NASA-S-66-5158 JUN 8

LUNAR ORBIT COAST ABORT MODES

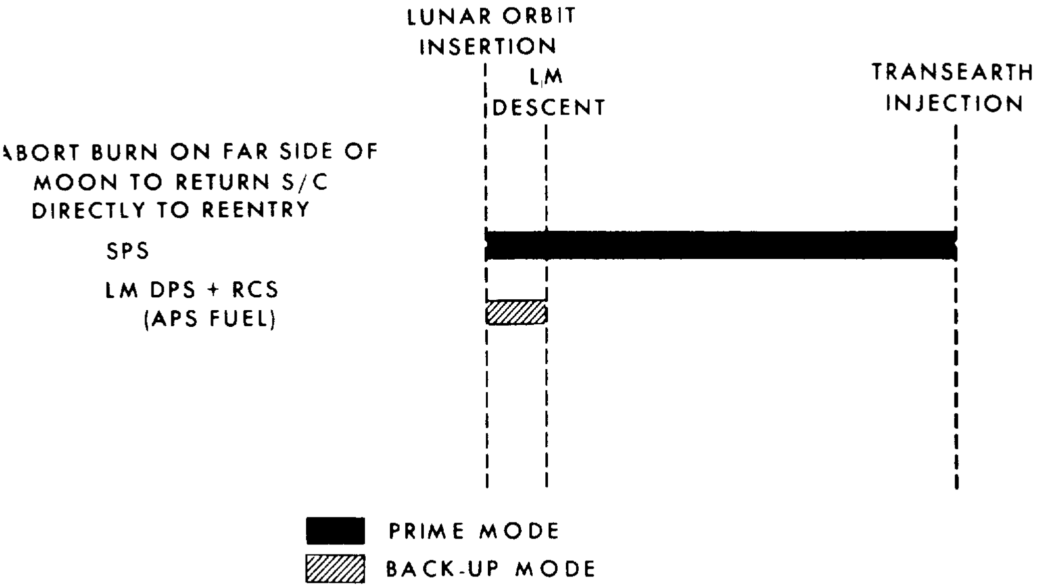


Fig. 43

NASA-S-66-4978 MAY 25

ABORT FROM LUNAR ORBIT COAST

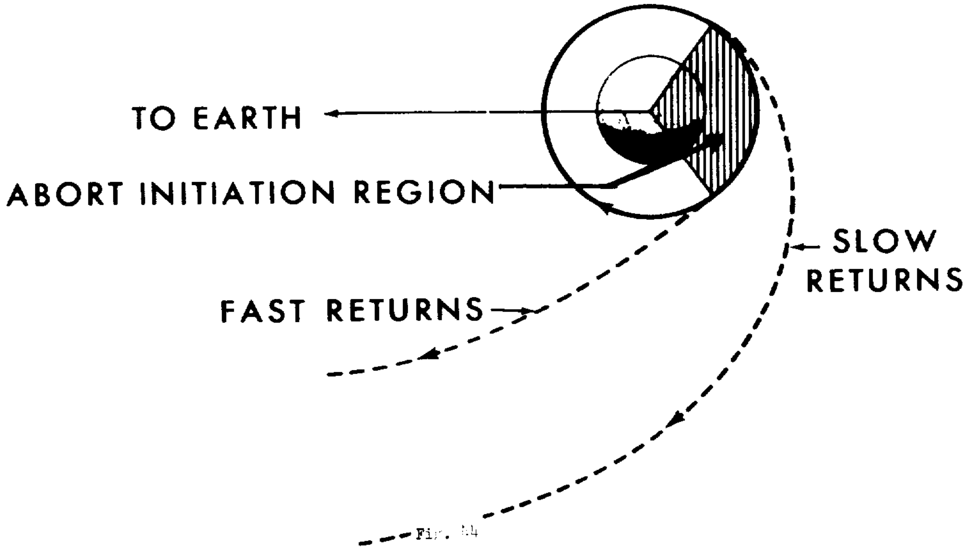

TO EARTH
ABORT INITIATION REGION
SLOW RETURNS
FAST RETURNS

Fig. 44

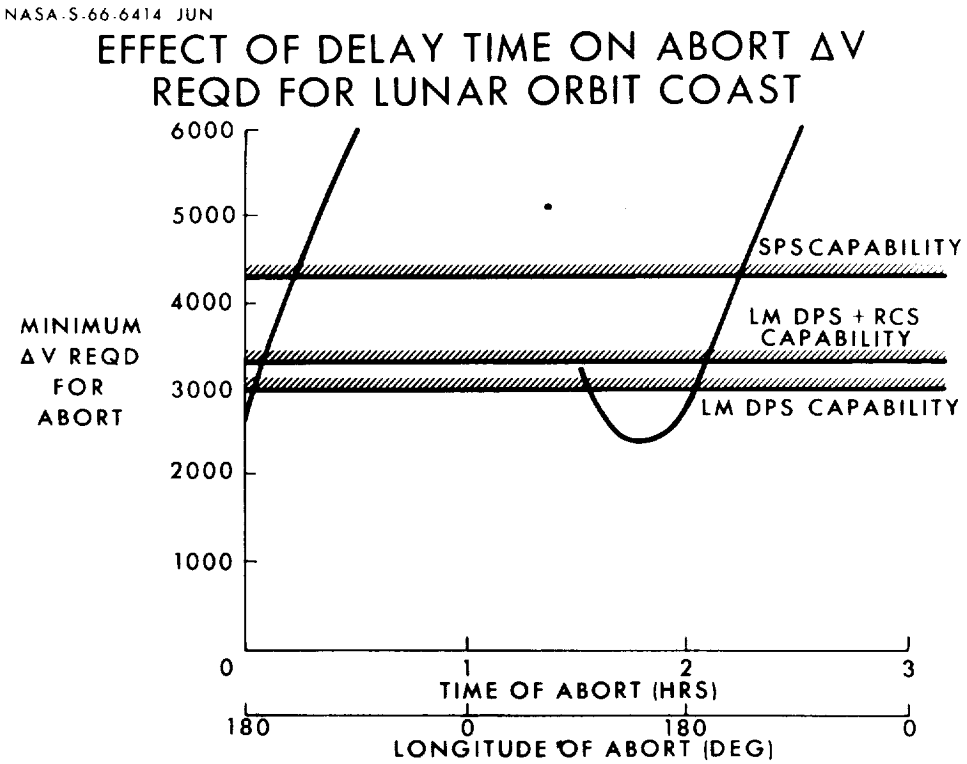

NASA-S-66-6414 JUN
EFFECT OF DELAY TIME ON ABORT ΔV REQD FOR LUNAR ORBIT COAST
MINIMUM ΔV REQD FOR ABORT
6000
5000
4000
3000
2000
1000
0
SPS CAPABILITY
LM DPS + RCS CAPABILITY
LM DPS CAPABILITY
1
2
3
TIME OF ABORT (HRS)
180
0
180
0
LONGITUDE OF ABORT (DEG)

Fig. 45

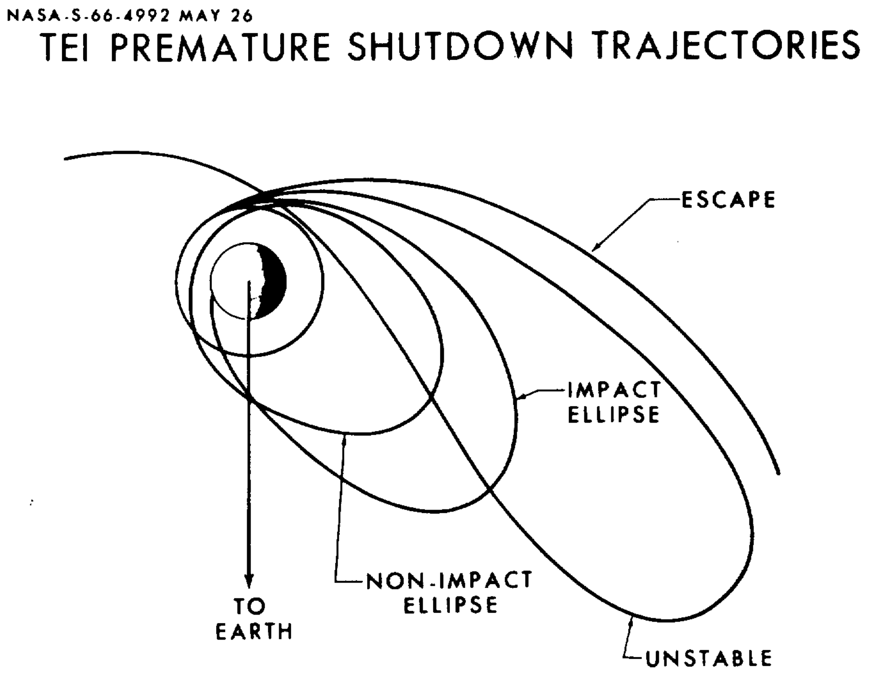

NASA-S-66-4992 MAY 26
TEI PREMATURE SHUTDOWN TRAJECTORIES
ESCAPE
IMPACT ELLIPSE
NON-IMPACT ELLIPSE
TO EARTH
UNSTABLE

Fig. 46

NASA-S-66-6401 JUN

PRE-ABORT TRAJECTORY FAMILIES DURING TRANSEARTH INJECTION

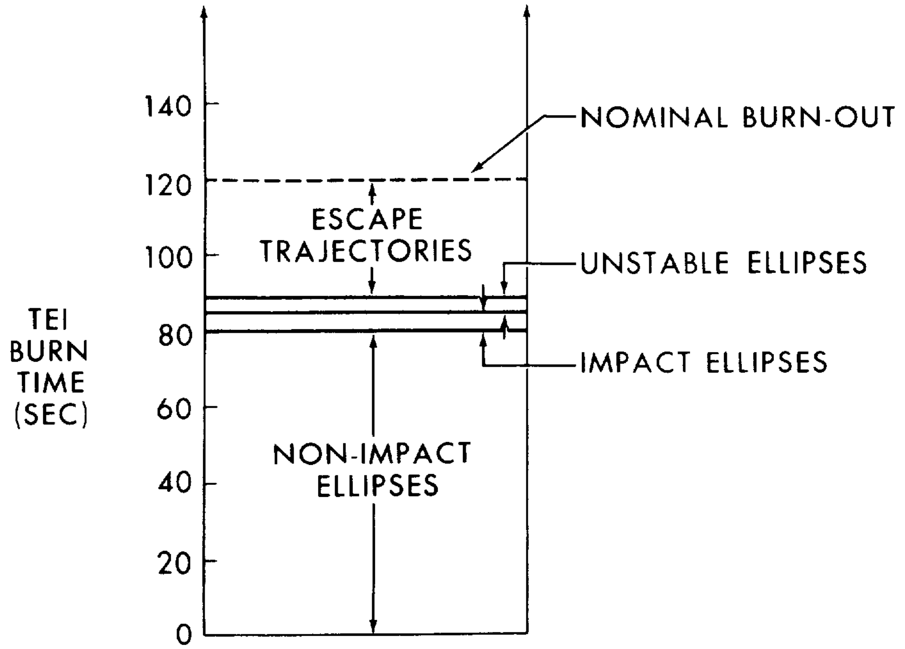


Fig. 47

NASA-S-66-5157 JUN 8

TRANSEARTH INJECTION ABORT MODES

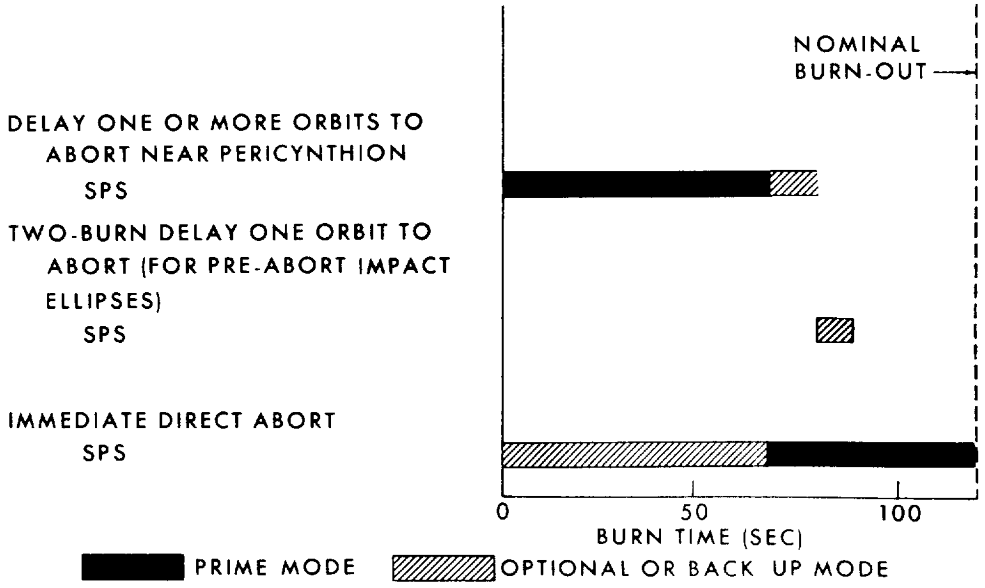


Fig. 48

NASA-S-66-6416 JUN

SPS ABORT CAPABILITIES DURING TRANSEARTH INJECTION PHASE

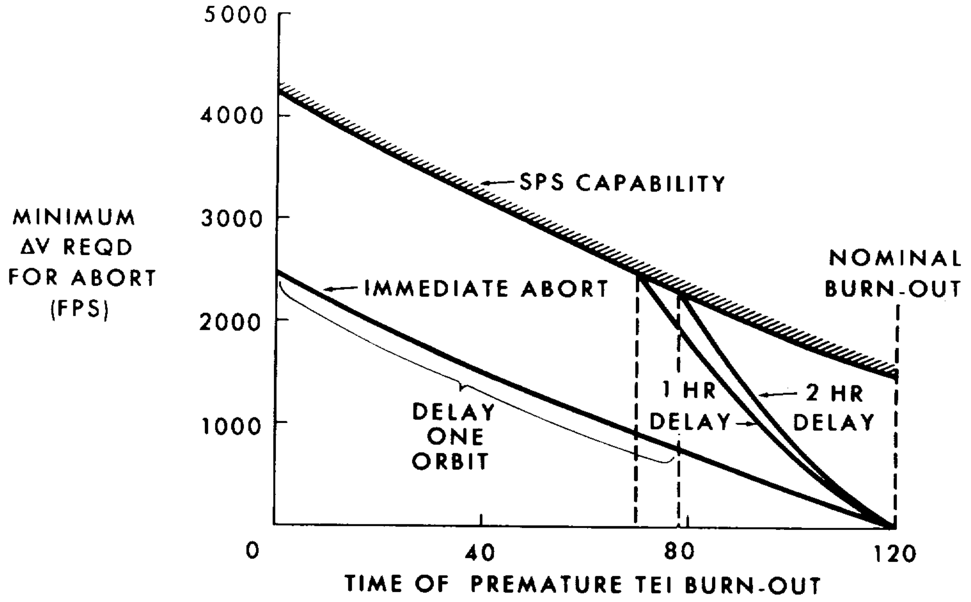
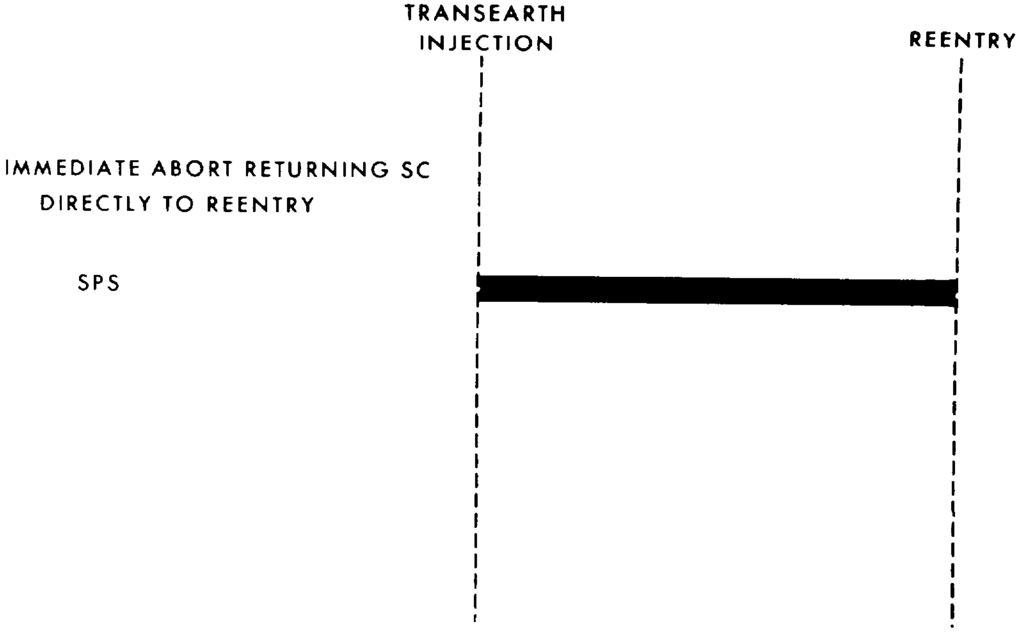


Fig. 49

NASA-S-66-5159 JUN 8

TRANSEARTH COAST ABORT MODES



TRANSEARTH INJECTION
REENTRY
IMMEDIATE ABORT RETURNING SC DIRECTLY TO REENTRY
SPS

Fig. 50

NASA-S-66-4981 MAY 25

ABORT FROM TE COAST

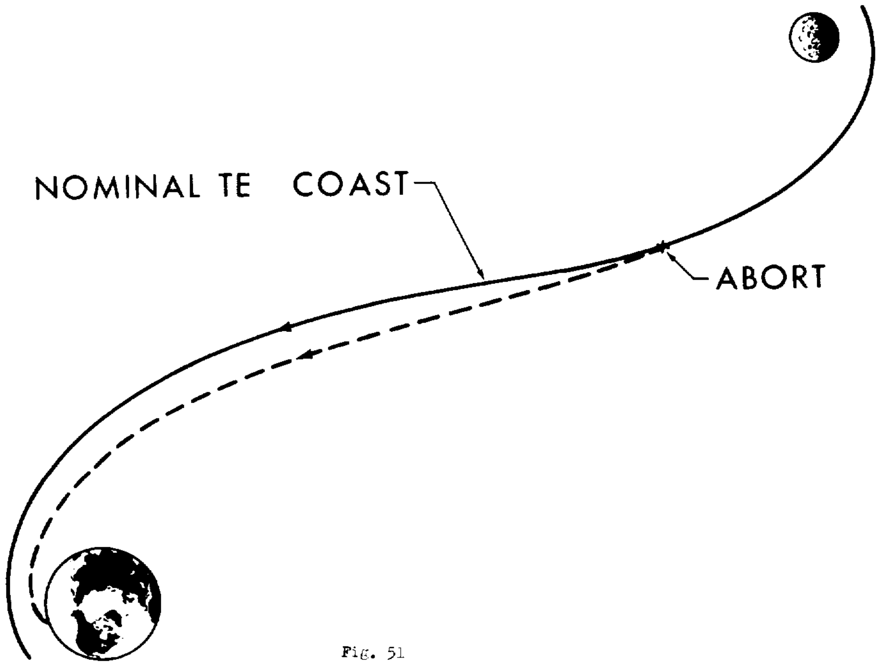

NOMINAL TE COAST
ABORT

Fig. 51

NASA-S-66-6408 JUN

ABORTS TO SPECIFIC SITES DURING TRANSEARTH COAST

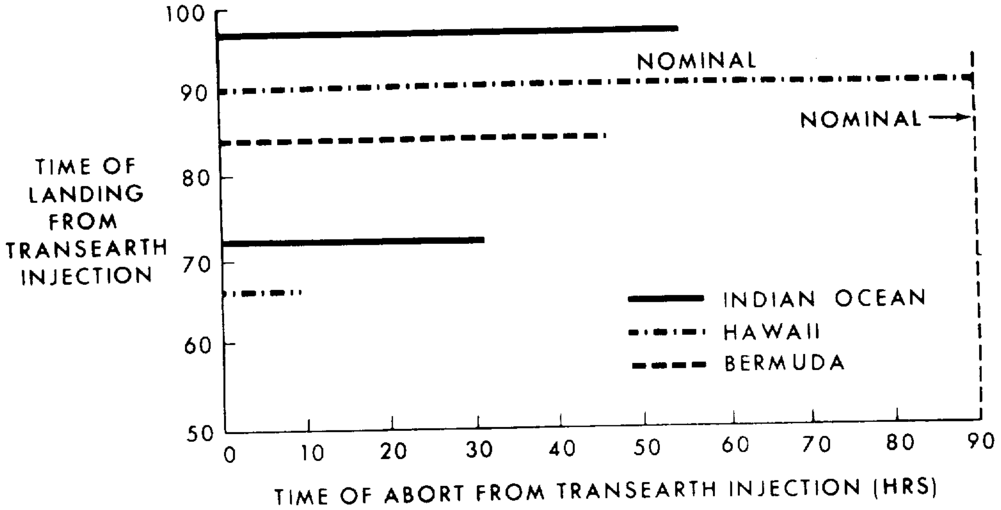


Fig. 52

NASA S 66 6477 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORT DURING EARTH PARKING ORBIT PHASE

PROPULSION SYSTEM	Δ V CAPABILITY
SPS (PRIME)	10,000fps
LM DPS	2000fps
LM DPS + RCS (APS FUEL)	2250fps
S-IVB	10,000fps +

Table
I

NASA-S-66-6436 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORT DURING TLI AND TL COAST PHASES

PROPULSION SYSTEM	ΔV CAPABILITY
SPS (PRIME)	10 000fps LM JETTISON 5300fps WITH LM
SM/RCS (RCS FUEL ONLY)	120fps LM JETTISON 90fps WITH LM
LM DPS	2000fps FULL SM
LM DPS + RCS (APS FUEL)	2250*fps FULL SM

*RCS BURN TIME LIMITED TO 1000 SEC
ASSUMES FOUR RCS THRUSTERS USED

Table
II

NASA-S-66-6475 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING LOI PHASE

PROPULSION SYSTEM	ΔV AVAILABLE
SPS (PRIME)	10 000 → 4300fps (LM JETTISON) 5300 → 2000fps (WITH LM)
LM DPS	2000 → 3000fps
*LM DPS + RCS (APS FUEL)	2250 → 3300fps

*ASSUMES DESCENT STAGING AFTER DPS BURN AND ALSO ASSUMES 1000 SEC LIMIT ON RCS THRUSTERS

Table III

NASA-S-66-6437 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING LO COAST PHASE

PROPULSION SYSTEM	ΔV CAPABILITY
SPS (PRIME)	4300 fps
LM DPS	3000 fps
*LM DPS + RCS(APS FUEL)	3300 fps

*ASSUMES DESCENT STAGING AND 1000 SEC LIMIT ON RCS THRUSTERS

Table IV

NASA-S-66-6496 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING TEI PHASE

PROPULSION SYSTEM	Δ V CAPABILITY
SPS (PRIME)	4300 → 1500fps

Table V

NASA-S-66-6444 JUN

PROPULSION SYSTEMS AVAILABLE FOR ABORTS DURING TE COAST PHASE

PROPULSION SYSTEM	Δ V CAPABILITY
SPS (PRIME)	1500fps

Table VI

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

by

Morris V. Jenkins

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

The term software implies the programs essential to the onboard and ground computers necessary for guidance and navigation; it also takes into account the input and output interfaces, such as the mode of entry into the computer and the outputs driving the displays.

The points to be discussed in this paper are outlined below.

SOFTWARE CONSIDERATIONS (Figure 1)

The main consideration is to provide software adequacy in all mission phases to insure crew safety. The next consideration is to select programs which significantly enhance the probability of mission success.

In general, onboard adequacy insures crew safety, and in particular, attention is paid to an onboard independent capability to return.

Programs have been selected which significantly contribute to the probability of mission success. Two obvious examples are the programs which allow the spacecraft G&N system to provide an essential contribution to the capability to perform the translunar injection in the event of a Saturn inertial platform failure and another example is the program formulation to accept ground updates.

It is found in some cases that ground capability is essential.

All the time the ground capability is contributing to mission success with its active and monitoring guidance and navigation capability, together with the necessary provision of peripheral data.

Combined ground and onboard solutions further advance crew safety, provide an additional measure of optimization such as to produce the maximum probability of mission success.

The greatest factor contributing to our present day status of software capability can be attributed to the decision made in the initial stages of framing the G&N capability for Apollo. This decision was that of adopting the simple and reliable MIT targeting schemes. The principles of these schemes are being used, in many instances, both onboard and on the ground.

ADEQUACY OF ONBOARD SOFTWARE CAPABILITY (Figure 2)

The main features of the software which contribute to crew safety include the following:

1. An acceptable degree of independent capability to return.

2. The capability to accept ground updates either by the up-digital link in the case of the Command Module Computer (CMC), or by voice as in the case of the Lunar Module Guidance Computer (LGC) where the entry to the computer is made through crewman use of the Display Keyboard (DSKY).

3. Compatibility of software in the Prime Guidance and Navigation Control System (PGNCS) and the Abort Guidance System (AGS) and an obvious example of this is the equivalent targeting for the powered ascent.

4. The existence of a double check on the relative state between the Command Service Module (CSM) and the Lunar Module (LM) which is derived from the LM rendezvous radar observations of the CSM and the CSM sextant observations of the LM.

5. The LM can direct the CSM to do a concentric flight plan rendezvous, which is the preferred mode of rendezvous in non-time-critical situations. Briefly, this is achieved by the entry of the CSM state into the LGC, whereby the LGC will solve the concentric rendezvous solution for the CSM. Following this, the necessary information can be transmitted by voice to the CSM and the necessary targeting parameters are entered into the CSM DSKY by the crewman.

6. The software logic facilitates meaningful crew checks and this is achieved by the way the programs are framed such that data may be input in a form such that the crewman has an understanding of what he is doing and, in addition, the computer driven displays are also in meaningful form to the crewman. An example of this is where the crewman is allowed to input data referred to the local horizontal and the trajectory plane and, in addition, displays can be referred to the same reference.

The obvious examples of programs selected to promote mission success are:

1. The steering processors which constitute the main components of the "thrust programs."

2. The programs which allow the translunar injection to be accomplished in the event of a Saturn inertial platform failure. At a minimum, this capability will exist if the takeover decision is made prior to the burn.

3. The capability to accept alternative or modified targeting from the ground will facilitate the opportunity to use ground optimized solutions.

A summary of what makes these capabilities possible appears in the next section.

BASIC ONBOARD G&N SOFTWARE CAPABILITY (Figure 3)

The main factor in the capability of a manned vehicle to achieve its destination is the crew G&N management. This management is possible through the monitoring of computer driven displays. A familiar pattern of G&N management questions may still be recognized. In their basic form they are:

1. Where are we?

2. Where will we be?

3. What change of course is needed?

4. Is the prime G&N system controlling the propulsion correctly?

The corresponding respective activities are commonly recognized as:

1. Navigation

2. Dead reckoning

3. Change of course determination

4. Maneuver control

In current software terms, these same activities are respectively recognized as:

1. Trajectory determination processors

2. Trajectory prediction processors
3. Pre-thrust programs
4. Thrust programs

The function of the "trajectory determination processors" is to accept observations such as the range and range rate of the rendezvous radar, and to formulate a smoothed solution wherein the current position and velocity is determined. This is essentially achieved by the equivalent of fitting a trajectory in a least squares manner such as to minimize the residuals.

The function of the "trajectory prediction processors" is to accept a state vector from the trajectory determination process and with the ephemeris information available as an input. The trajectory prediction processors will produce the best estimate of position and velocity at any future required time, provided that free coast conditions exist. The ephemeris data includes oblateness terms and the position and magnitude of other perturbing gravitational sources, such as the moon if we are in a near earth phase.

The main function of the "pre-thrust programs" is to determine the targeting parameters for the "thrust programs."

The function of the "thrust programs," the main components of which are the steering equations, is of course to achieve desired cutoff conditions corresponding to a required trajectory condition.

Some amplification of this discussion occurs in the next section.

AMPLIFICATION OF THE BASIC PROGRAMS (Figure 4)

The main component of the "trajectory determination processors" is the "Kalman filter." The Kalman filter is essentially an efficient way of writing least squares fit equations. It is particularly adaptable to the processing of one observation at a time. In contrast, on the ground, data is processed in batches. This is for the convenience of manual editing and the detection and exclusion of a bad data source. In the case of the ground, the Bayes least squares method has been adopted.

The "trajectory prediction processors" have as their main component the Encke method. The stratagem that this method uses is to use the equations of motion in incremental form. The incremental form is achieved by subtracting the basic two body equation, i.e., the point mass - particle solution, away from the total equation of motion. Because the equations of motion have been reduced to incremental form in this manner, the integration routine can take larger steps for the same accuracy that would be accomplished by an integration of total quantities. Prolonged, extensive tests were carried out at Manned Spacecraft Center to verify this assertion.

The function of the "pre-thrust programs" is threefold: (1) to determine the targeting and put it in a form which the "thrust programs" will accept; (2) to determine the initial thrust command attitude; and (3) to determine the preferred platform alignment.

The function of the "thrust programs" is to provide state input, i.e., position and velocity data, to the steering laws which in turn will determine what the appropriate command quantities are. In general, the command quantities will determine the attitude and duration of the thrust. The steering laws normally strive to null out the discrepancy between command and response. The commonest form of steering used by the onboard programs is "X product steering." This is an old MIT steering law in which the velocity to be gained is implicitly constrained to be inertial during a continuing thrusting process, while the scalar value of it is shrunk to acceptable limits, approximately zero. "E guidance" is used for LM descent and ascent where additional control characteristics are required to fit the event peculiar to these special maneuvers. Another form of thrust which will be formed is where the thrust line is controlled in direction relative to "local vertical coordinates."

TARGETING

As was mentioned in the section on Software Considerations, the greatest factor contributing to our present day status of software capability can be attributed to the decision made in the initial stages of framing the G&N capability for Apollo. This decision was that of adopting the simple and reliable MIT targeting schemes.

In order that an insight into the potential of adopting these principles of targeting, together with a realization of the flexibility that exists for the current stage of program formulation, all the major types of targeting are discussed, at least conceptually. It should not be assumed at this time, in terms of specifics, that all of these targeting forms will be employed on the lunar landing mission. However, minor modifications of the stratagems used will be employed for the lunar landing mission, both for onboard and ground G&N systems.

TARGETING - TYPE 1 - Translunar Injection (Figure 5)

The essential ingredients of this type of targeting are: (1) utilization of current $\vec{r}$; (2) specification of target vector; and (3) specification of energy. A sensitive parameter in the control of lunar trajectories is energy. Therefore, it is not surprising that this form of targeting was selected for "translunar injection."

Corresponding to the optimized lunar trajectory flight plan, there is a corresponding two body solution generated by the same $\vec{r}$ and $\vec{V}$ vector at the instance of translunar injection. The equivalent two body solution assuming a point mass gravity field only is commonly called the osculating conic, in this case an ellipse. The steering law is framed such that it is only cognizant of the two body situation. An $\bar{r}_t$ is selected on the osculating ellipse, a specified energy is stipulated for the ellipse, and this is achieved by specifying the semi major axis a. This, together with the current burn $\vec{r}$ instantaneously specifies the two body trajectory. This is done in a continuum of instances during a continuous burn wherein the velocity required and its derivative with time is continuously known such that cross product steering may be applied. The target $\bar{r}_t$ on the osculating ellipse is sometimes known as the phantom target. The additional perturbation gravity effects will cause the spacecraft to drift on to the actual trajectory which will go around the moon.

TARGETING - TYPE 1 - Lunar Orbit Insertion and Repeat for Plane Change (Figure 6)

In this instance the orbit is being controlled around the moon and the period of it is a parameter to control, but this is equivalent to saying the energy must be controlled, i.e., the semi major axis a will be specified. As will be guessed by the fact that we are classifying this scheme again as Targeting - Type 1, a target vector $\bar{r}_t$ must be specified. However, the latter is done in a special way in order to insure circular conditions. The scalar value, $|r_T|$, of the vector $\bar{r}_t$ is specified to be the same length as the current burn $|r|$ and the semi major axis a is continuously commanded to be the same value as $|r_T|$ which in turn is the value of the current burn $|r|$.

TARGETING - TYPE 2 - First Midcourse Correction and Initiation of Circumlunar Abort (Figure 7)

The principles used in this type of targeting are as follows:

1. Use an accurate trajectory predictor to determine what the state vector will be corresponding to some particular event in the future. The particular event could be specified by time or when pericynthion occurs, for example.

2. A two body conic trajectory solution is found which will satisfy the current state position vector and the future state. Then another body conic trajectory solution is found to satisfy the current state position vector and the future state.

3. The velocity vectorial difference between the two two body conic solutions is noted.

4. The velocity vectorial difference of the conics is assumed to be the precise velocity to be gained vector. It would be if the conic solutions were perfect simulations of trajectories going from the current state position to the future states and indeed because the differential of the conics is taken, the inaccuracy due to using imperfect simulations is almost completely nulled.

5. The vector velocity differential of the conics is added vectorially to the precise current state as determined by the "trajectory determination process" and the resultant is accepted as the precise vectorial velocity required.

This is the basis of the determination of Targeting - Type 2.

It is appropriate now to consider how these principles are applied to the "first midcourse correction." The original preflight optimized flight plan has on it a point where the translunar trajectory pierces the sphere of influence and the time (T_2) of the event is known. Let it be supposed that the orbit determination process indicates a dispersed state subsequent to translunar injection and transposition and docking. At time T_1 it is decided to apply the first midcourse correction. The desired future state position at time T_2, as has already been mentioned, is known. The trajectory determination processors are used to find out where the spacecraft will actually be by continuing free coast until time T_2. Now by referring to Figure 7, it will be seen how the above described principles of Type 2 can now be applied to the circumstances at hand.

The first midcourse correction capability may also be used for the "initiation of circumlunar abort."

TARGETING - TYPE 2 - Second and Third Midcourse Corrections and Circumlunar Abort (Figure 8

It is now convenient to examine the circumstances for the second and third midcourse corrections. They are assumed to occur within the lunar sphere of influence and, hence, it is advisable to reference position and velocity to a lunar set of coordinates. It may be assumed that the stratagems used for the second midcourse correction will be used for the third midcourse correction. Let it be supposed that the trajectory position processors have determined that our current state within the sphere of influence will propagate to an unacceptable dispersion prior to lunar orbit insertion. More precisely, the activity would be as follows:

1. The trajectory determination processor predicts precisely ahead to find out where the "propagated pericynthion" will be. For simplification, let it be said that for fuel economy reasons, that the lunar orbit insertion must occur at a pericynthion point and therefore it is mandatory that this pericynthion point must be in the predetermined "desired orbit plane" which has been selected to be consistent with an overall optimization of flight plans including LM landing and subsequent rendezvous. However, when the current "propagated pericynthion" is determined, it is noted that it is not at the required altitude and furthermore, it is not in the "desired orbit plane."

2. The first action is to shrink the scalar value of the "propagated pericynthion" vector.

3. The next action is to rotate the shrunken "propagated pericynthion" vector into the "desired orbit plane." This is done by rotating about the current state position vector $\vec{r}$ holding the angle between this vector and the "propagated pericynthion" vector constant. In this way the "desired pericynthion" is formed.

Hence, the situation in which Targeting - Type 2 can be applied may be recognized. The original "propagated pericynthion" constitutes the future state generated by the "predicted actual" and the newly formed "desired pericynthion" constitutes the future desired state and, hence, two, two body hyperbolic conic solutions can be applied to find the required differential and thus the principles of Targeting - Type 2 are being applied. All of the above can be fairly easily followed by making reference to Figure 8.

The completion of the translunar leg is essential to a circumlunar abort capability and the principles of the "second and third midcourse corrections" can also be applied to "circumlunar abort."

TARGETING - TYPE 3 - Transearth Injection, Transearth Abort, Transearth Midcourse (Figure 9)

Targeting - Type 3 is the targeting scheme found in the MIT "return to earth" program. A modified form of this program will be used by the ground computers for determining maneuvers corresponding to the mission phases indicated in the title of this section.

Basically, the required solution is scanned by approximating the return to earth trajectory with a hyperbola joined to an ellipse at the sphere of influence. This is then followed by another iteration, this time in a precision mode.

For convenience, "Targeting - Type 3" can be identified with the initial conic iteration mode wherein there is a matching of conics at the sphere of influence.

Conic Iteration

The main steps of the conic iteration mode are as follows:

1. An initial guess of a suitable juncture point on the sphere of influence (S of I). Typically this is done by taking a point 45° center angle from the moon in the

current hyperbolic trajectory plane and also meeting the condition that it is on the S of I. Thus, the initial sphere of influence point (S of I point) is identified.

2. A parabolic trajectory is run from this point back to the earth maintaining the same plane at the S of I.

3. In general, it will be found that the earth has not conveniently rotated the recovery point into the plane of the parabolic trajectory.

4. A transit time is now selected corresponding to an entry speed at 400,000 ft earth altitude less than parabolic.

5. By successive selection of transit times, a transit time will be found when the recovery point has conveniently rotated into the plane of the trajectory. Up and downrange requirements for the recovery point are taken care of by the freedom of "reentry maneuver range."

6. Unfortunately, the required total solution has not been achieved even in conic form. Tentatively the near earth conic trajectory has been found; however, it must pass a subsequent step which is found in the next step.

7. At the S of I point, the earth dominated trajectory will have an energy appropriate to the potential and kinetic energy referred to the earth coordinate system. The joining hyperbola at the S of I point must have a compatible energy when the conditions at the join are referenced to the moon coordinate system. Furthermore, the terminal velocity vector of the hyperbola at the S of I point must, when earth referenced, be in the same straight line as the terminal velocity vector of the ellipse at the S of I point.

8. When the conditions of the previous step have been satisfied at the S of I point, then the backtracked hyperbolic solution from the S of I point is explored to determine whether it intersects the desired "departure vector."

9. If the backtracked hyperbolic solution does not intersect the desired "departure vector" then a simultaneous iteration proceeds, in which the S of I point is matched and other Δt's of the earth dominated eliptical trajectory are carried out in a rational manner to achieve convergence on

all required conditions for the "conic iteration." The required conditions may be summarized as: a correct matching of the conics at the finally selected S of I point; the recovery point must have rotated into the plane of the earth dominated elliptical trajectory at the time of entry; and the backtracked hyperbola must intersect the desired "departure vector." When this is done the conic iteration is at an end; however, this solution is not sufficiently accurate, and it is necessary to proceed to a "precision iteration."

Precision Iteration

In this mode, a precision trajectory found by numerical integration is determined which will go from the desired "departure vector" to the recovery point. The steps to accomplish this are:

1. A precision trajectory is back integrated from the last found conic reentry condition and the pierce point at the S of I is noted.

2. A precision forward integration from the "departure vector" is done in order to note the corresponding pierce point at the S of I.

3. In general, it will be noted that there is a mis-match at the S of I.

4. A simultaneous "precision iteration," using the derivatives found in the conic iteration, must now proceed to achieve a match. This is achieved by again adjusting the S of I point and transit time, Δt, of the earth dominated trajectory.

5. Finally, all the necessary end conditions indicated in the "conic iteration" description are achieved for the "precision iteration."

6. When this has been done, then the necessary osculating hyperbolic conic corresponding to the established "departure vector" and the determined Δ V vector for it is established for targeting purposes.

In detailed specifics, the above may not be exactly correct; however, all the above principles are employed in intended "return to earth" programs for both onboard and ground computers.

DESCENT AND ASCENT TARGETING (Figure 10)

In the descent and ascent targeting additional conditions are required beyond that which could be achieved with MIT cross product steering. So called E steering, also developed at MIT, is used to achieve conditions required during descent and ascent. Briefly, the command thrust vector is derived from explicit expressions involving the differentials of current and target conditions, and the gains in the steering laws are influenced by the ΔT_{GO}. The referenced diagram indicated the conditions required by descent and ascent targeting. No further comments will be made in this discussion, as this would incur a level of detail not intended for this discussion.

CONCENTRIC FLIGHT PLAN (Figure 11)

The targeting for the concentric sequence is discussed in this section. A prerequisite of getting exactly nominal flight plan conditions is that the standard launch routine is initiated at the correct time. The concentric sequence objectives per se are are not identical with achievement of absolutely nominal flight plan conditions. For example, it is not an objective of the concentric sequence to achieve precisely a 15 mi differential altitude at terminal phase initiation, which might well be an objective of a reference nominal flight plan. Late liftoff and subsequent dispersions will prohibit absolute achievement of the reference flight plan conditions. The concentric sequence retains two main objectives, one is the establishment of a Constant Differential Height coast (CDH) prior to the Terminal Phase Initiation (TPI). The other objective is that TPI should occur at a particular time. The latter is a requirement aimed at achieving TPI such that mission planning purposes may be satisfied, e.g., the line of sight relationship with the sun direction. The steps taken to achieve the concentric sequence objectives are as follows:

1. The first thing is to determine the Concentric Sequence Initiation (CSI). This is done on a basis of instantaneous impulses simulating the CSI and CDH burns.

2. The CSI is specified as a horizontal burn and the next steps will indicate how its scalar value is determined in order to achieve the objectives of the concentric sequence.

3. An iteration is initiated by selecting a tentative scalar value of the CSI maneuver.

4. The resultant orbit is examined to determine: the position of the apsis point; the transit time (Δt_1) to achieve it; and the differential height of the apsis point from that of the target vehicle orbit opposite the apsis point.

5. The phase lead of the target vehicle at the time of the CDH maneuver, which occurs at the apsis point, is noted. In addition, the phase lead of the phase lead required at TPI is determined simply on a linear proportional basis, i.e., phase lead $\phi = K_1.(\Delta h)$, where K_1 is a known constant. Thus the net central angle catch up, i.e., net phase angle catch up, is known. This is a property of the concentric flight plan. Furthermore, the catch up rate is determined on a linear proportional basis, i.e., $\dot{\phi} = K_2.(\Delta h)$, where K_2 is a known constant. This is another property of the concentric flight plan, which simplifies the iterative solution.

6. Now if the net $\Delta \phi$ between CDH and TPI is divided by $\dot{\phi}$, then the Δt_2 between CDH and TPI is established since $\Delta t_2 = \Delta \phi / \dot{\phi}$.

7. Now it must be determined if $\Delta t + \Delta t_2 =$ TPI time (T_2) - CSI time (T_0).

8. If the above condition is found not to be the case, then the whole process must be repeated and a new exploratory scalar value of the CSI maneuver is tried to initiate the second iteration. Following this iteration, a partial of ΔV of CSI with respect to time error at the desired time error referred to the desired TPI time (T_2), can be formed and used to converge with the third iteration which is initiated now with a calculated ΔV for CSI. It is just possible that even this third iteration has not achieved the objectives of the concentric sequence within acceptable limits; however, a highly convergent state exists and the objectives will surely be achieved with the next iteration.

With the required instantaneous ΔV established for CSI, the target orbit conditions for CSI are established such that a steering law can be applied to facilitate a finite burn. The final apsis point determination establishes the time at which the CDH maneuver will occur and in most cases, this will not be altered even with trajectory determination updating subsequent to the CSI maneuver.

The steps taken to establish the targeting of the CDH maneuver are as follows:

1. The value of $\dot{r}$ of the target vehicle orbit opposite the CDH apsis point is determined.

2. A requirement of the CDH maneuver cutoff is that it will have the same value of $\dot{r}$ as determined in the previous step.

3. In addition, the semi major axis a of the resultant orbit following the CDH cutoff is determined by subtracting (Δh) from the scalar value of r of the target vehicle orbit opposite the CDH apsis point. The last two steps describe the necessary cutoff conditions for the CDH maneuver and they will minimize the variation in differential height subsequent to the CDH maneuver. If the target vehicle orbit is circular, it is plain to see that the resultant orbit following the CDH maneuver will be concentric. It is from this property that the concentric flight plan derives its name.

It is certain that there will have been execution errors following the CSI maneuver and the CDH maneuver, and it is desirable that the resultant dispersions are absorbed in a manner to accomplish near minimum ΔV penalty while achieving rendezvous at approximately, but not constrained to, planned time. Another property of the concentric flight plan is that near optimum phase height relationships may be achieved by one single correlating parameter, and that is the elevation of the line of sight from the LM to the CSM. For example, let it be said that it is an elevation angle 26-1/2°. If this elevation angle is adhered to as a trigger for the TPI maneuver, then all subsequent dispersions will essentially be absorbed. The potential penalty is that TPI triggered in this manner will not occur at the selected time for mission planning purposes. However, this will not really be a problem in that the mission planning selected time for TPI will have been chosen knowing that dispersions of a known magnitude might occur and due allowance will have been made for it in the selection of TPI time. Therefore, although the occurrence of an elevation angle 26-1/2° may occur earlier corresponding to a lower altitude of CDH, there should not be any resultant problems.

The following steps are taken to achieve TPI:

1. The occurrence of an elevation angle of 26-1/2° for the line of sight is determined.

2. The position of the target vehicle following 140° of central angle travel subsequent to the newly determined TPI time is established. An equivalent of this is just to find the position of the target vehicle at a set elapsed time from the TPI time, supposedly corresponding to 140° travel of the target vehicle.

3. A Lambert solution for an intercept trajectory from the TPI LM vector to the intercept vector is now determined.

4. The orbit of the Lambert solution is the basis of the targeting to which cross product steering may be applied during a finite burn in which the Lambert solution will be repetively found in order to derive the velocity required.

The midcourse correction occurring at a fixed elapsed time after TPI will also be derived from the Lambert solution. Main braking may also be determined by solving Lambert's problem providing that a lsightly modified intercept point and associated time consistent with an approximately average closing rate which is specified from mission planning considerations. Subsequent braking will be carried out using the same principles until manual takeover occurs.

Before closing the section on concentric flight plan targeting, it should be mentioned that both the CSI and CDH maneuvers are constrained to burn parallel to the CSM orbit plane.

REENTRY TARGETING

Reentry targeting is achieved by creating a reference in real time such that subsequent to the reference gains on the lift control may be adjusted as functions of differentials between the reference and actual conditions. Hence, the discussion is centered on two main subjects, "creation of the reentry reference," and "real time iterative control."

CREATION OF THE REENTRY REFERENCE (Figure 12)

The reentry discussion which follows is to some extent conceptual; however, the description of the major principles employed is sufficient to give an insight to the onboard computer logic which will be formulated to provide control during reentry from a lunar mission.

The CSM will enter lift vector up. This is because the last midcourse correction will be aimed at a corridor height consistent with the following two conditions:

1. If the onset of g tests indicate a possibility of a near skip condition, then the situation may be safely corrected by rolling the lift vector down until g onset is satisfactory.

2. If g predicted is more than the preferred limit, say 8 g, then if the left vector control retains the lift vector upwards, then the 8 g will be exceeded, but not to an extent which will really trouble the crewmen and certainly the g reached will be well below that level which would give concern regarding structural integrity.

With this philosophy, vertical lift is maintained until the trigger for the "creation of the reentry reference" is activitated. This trigger is a radial rate of 700 ft/sec (see Figure 12), whereupon "constant drag control" is initiated, i.e., by controlling the roll angle of the lift vector, a control is exercised which strives to maintain constant drag.

A reentry reference is now calculated. This is preferable to using a preflight reentry reference, in that basic assumptions in any preflight nominal would be impossible to maintain. The reference trajectory created in real time attempts to take into account actual conditions being experienced such as the actual height, velocity, density relationships. The steps taken to find the reference in real time are briefly, conceptually described below:

1. The constant drag mode, together with simplified and linearized equations of motion referred to the trajectory plane, allow a prediction of the state conditions when the flight-path angle is zero, i.e., at the bottom of the pull up. Hence R_1 is known.

2. With an effective L/D = .2 from the bottom of the pull up, the state conditions when the acceleration falls to 6 ft/sec^2 may be predicted. When the acceleration, and to some the preferred term may be deceleration, has fallen to 6 ft/sec^2, than a space condition trajectory is assumed. However, at this point let us note that the state condition at this juncture is noted and, hence, R_2 is determined.

3. From this deceleration point until the acceleration has risen to 6 ft/sec^2, "two body space assumptions" are applied and, hence, R_3 may be calculated and the state

conditions when the acceleration has reached 6 ft/sec^2 may be determined. This corresponds to the definition of a second entry.

4. The second entry range is predicted by linearized assumptions assuming a constant effective L/D and, hence, the reentry range corresponding to second entry, R_4, is known.

5. R_1, R_2, R_3, R_4, are summed and compared with the desired range which corresponds to the downrange distance of the splash point relative to the initial entry point.

6. The error Δ R is noted and if it corresponds to an overshoot, the consequent logic is that the "constant drag control" which is being maintained during the "creation of the reentry reference" is extended for another cycle and if, after going through the whole process again, there is still an overshoot predicted, this time smaller because of the energy absorbtion, another cycle will be called for. If, however, there is an undershoot ΔR error, then the "constant drag control" mode will be terminated. If the predicted range is within a certain limit "real time iterative control" would commence immediately. If the undershoot is exceeding acceptable limits, then the "down control" of the "real time iterative control" would command lift vector up. In any case, the phase of iterating onto the reentry reference terminates.

Before leaving this section, it should be noted that if the velocity corresponding to 6 ft/sec^2 deceleration at the predicted end of the "exit" phase is more than circular velocity, then this in itself will call for a recycle while "constant drag control" is maintained, in order to absorb a sufficient amount of energy to reduce the exit velocity to below that of circular conditions.

REAL TIME ITERATIVE CONTROL (Figure 13)

When the mode in which the real time reference is being formulated terminates, a "down control" mode begins. The logical steps taken subsequent to the initiation of "down control" are conceptually described below:

1. The "down control" uses expressions formulated from simplified, linearized equations of motion. The differentials between the reference conditions and the corresponding actual values computed in real time and compared at the same velocity value as is being determined in real time

provide the commands for the "down control" (see Figure 13). This control attempts to achieve the same velocity magnitude at the bottom of the pull up, i.e., with flight-path angle (γ) = zero.

2. When it is determined that $\dot{r}$ = zero, i.e., $\gamma = 0$, then the logic will go over to employing "up control." In this mode again simplified, linearized equations of motion in the trajectory plane are used. With the equations in this form, V_{ref} and $\dot{r}_{ref}$ are derived for given drag values. The differentials between V_{ref} and actual V, $\dot{r}_{ref}$ and actual $\dot{r}$, determined for the same drag value which essentially is being measured, provide variable gains to increment effective L/D (see "up control" block on Figure 13). The "up control" mode will continue until the deceleration has fallen to 6 ft/sec^2.

3. The onboard computer, having sensed the deceleration has fallen to 6 ft/sec^2 will occur which by definition is the initiation of the "second entry phase."

4. During the "second entry phase," the second entry range is predicted empirically based on drag and velocity differentials referenced to preflight simulated data. Velocity is the correlating parameter establishing the differentials.

5. The range error is fed back and acts as a variable gain for effective L/D increment during the "second entry phase" (see "second entry phase" block on Figure 13). With this iterative control, downrange error is nulled.

6. During the "second entry phase," computer logic will also call for a prediction of max g and when the max g predicted is beyond the acceptable limit, then the normal "second entry phase" control logic is interrupted and lift vector up is commanded; however, the max g prediction continues and when the predicted max g gets within the acceptable limit, then normal "second entry phase" control logic continues to call for L/D increments which will null the downrange error.

In all atmospheric phases of the reentry control, lateral control is also exercised to null out crossrange errors. Essentially, it works on the following principles:

1. When max uplift is called for, the lift vector will retain a 15° roll angle in a direction such as to null out crossrange errors.

2. For all other roll angles used for controlling effective L/D in the plane of the trajectory, the roll angle is in the direction which will null out any existing predicted crossrange errors.

This concludes the conceptual description of the way in which downrange and crossrange errors are nulled out during reentry, together with an indication of the logic to avoid potential skip and unacceptable max g conditions.

ENTRY MONITORING SYSTEM LOGIC (Figure 14)

This section includes a brief reminder of the monitoring that goes on parallel to the operation of the prime guidance system. First of all, the onset of g is noted in the very initial phases of the first entry, and if the onset does not reach critical value within 10 sec, then lift vector down will be commanded. This is classified as "corridor verification" (see Figure 14).

"Corridor verification" is followed by "g, V" monitoring where the actual "g, V" that is being experienced is detected by backup integrating accelerometer and is shown as a trace on a display where rays are scribed and g boundaries corresponding to entry velocity converge on the asymptotic value of the acceptable g limit (see Figure 14) and the associated value of 9 g. If the slope of the trace is more negative than the appropriate g boundary corresponding to the actual entry velocity, then this is an indication that the g to be encountered will be unacceptable. If the prime guidance in this instance is not commanding lift vector up, then a serious excessive g situation is developing. If, during the "exit" phase of the first entry, the trace is parallel to the "skip" rays and the prime guidance is not commanding lift vector down, then a serious potential skip situation is developing. To insure safe reentry the crewman will probably resort to manual control of the lift vector; however, no longer will an accurate splashdown point be possible comparable with when the prime guidance is functioning properly.

Further development work is continuing as regards range control lines added to the "g, V" display. This latter development would reduce gross range errors during manual control in circumstances when the prime guidance was not functioning correctly or capable of providing accurate displays. Reentry ship tracking of the exit phase, especially where effectively direct altitude rate is being measured, will also rapidly detect gross overshoots or undershoots.

SOFTWARE CAPABILITY NOT ONBOARD (Figure 15)

The title implies a list of capabilities not onboard, but it should be explained at this juncture that the ground capability in every case, will provide the capability which is stated does not exist for the onbaord capability.

1. On the ground, the capability exists to run ahead of real time and simulate an intended maneuver such that all conditions subsequent to the burn may be evaluated such as tracking acquisition, look angles, etc., and an accurate prediction of coast subsequent to the burn may be evaluated. The onboard capability is restricted to free coast prediction.

2. The ground control has the capability of predicting the characteristics of the total flight plan assuming present targeting values are retained. The fuel reserves may be evaluated after all the maneuvers have been made, together with all the appropriate operational information required to insure a mission success.

3. Ground control has the capability to determine a dispersed condition for the spacecraft, and from this dispersed state, can re-optimize the entire flight plan. This incurs sending new targeting data to the spacecraft which the onboard capability has been formulated to accept.

4. If the LM has to take off at a time when the CSM is out of communication with it, e.g., when the CSM is behind the moon, then the consequent rendezvous is excessively lengthy for an entirely LM active rendezvous with the CSM remaining passive. The ground control can design maneuvers for the CSM subsequent to which the LM active rendezvous is accomplished in a considerably shorter time than would be the case with the CSM remaining passive. There are cases with the CSM, say 160° phase behind the LM at insertion, where if an out of plane situation exists, and the CSM remains entirely passive, then the LM capability to rendezvous within its lifetime and delta V limits are in jeopardy. With the ground control in the picture, targeting advice would be issued to the CSM such that the subsequent LM active rendezvous could be accomplished in less time and for less delta V.

5. The onboard capability, in many instances, has not got the capability, and this is by intent as will be indicated later, to calculate peripheral data. Examples of peripheral data are: look angles, prediction of optical conditions, prediction of lighting conditions for rendezvous, docking, prediction of contrast and shadow conditions for descent, trajectory information for operational facilities such as monitoring aircraft and recovery ships, and differential radiation prediction.

6. The ground has the capability of using observations from mixed sources to provide optimum trajectory determination solutions. An example of the mixed sources would be ground radar and onboard radar. The observations receive appropriate weighting consistent with a knowledge of the accuracy existing for the circumstances incurred.

In the next section, it is intended to discuss the ground control G&N functions, not in the sense that it is filling gaps in the onboard capability, but in the sense of what the ground control provides as a service when the onboard G&N systems are functioning normally.

GROUND SUPPLEMENTARY AND BACKUP G&N FUNCTIONS (Figure 16)

1. In many circumstances during the lunar landing mission, the MSFN determination of position velocity will be more accurate than that which would be determined onboard. Furthermore, for mission planning purposes, the number of observations is being restricted in order to economize on RCS fuel, i.e., a change of attitude would, in all probability, be required prior to a sequence of onboard sightings. Consequently, it is the ground control which provides updated or modified targeting data.

2. The ground control provides monitoring information for manual maneuvers. Examples of this are launch abort, manual insertion into orbit using Stabilization Control System (SCS) mode, when the terminal phase initiation is performed manually, then the fround would provide monitoring information prior and subsequent to it. There is the distinct possibility that prior to the first braking of the terminal phase maneuvers when the AGS is being used to accomplish the rendezvous that the MSFN will supply the targeting information.

3. The ground control, with its extensive computer capability available to it, together with the presence of specialists in the particular area in which a problem may occur, has a capability for a more extensive exploration of the possibilities, especially in contingency situations.

4. The MSFN capability will, on occasions, perform an umpiring function. For example, the LM has two onboard guidance systems which will provide data. If a discrepancy occurs, then the MSFN will provide a significant clue as to which system is functioning correctly. When long tracking passes are possible, then the MSFN will provide a determination of the orbital elements or of the relative state between the two modules, when coasting conditions prevail. During short tracking passes, the MSFN can be relied upon to give extremely accurate data along the direction of the line of sight from the ground radar dish to the spacecraft. This is because the MSFN is relying on Doppler rate information. Even during LM launch, this particular measurement by the MSFN will provide an immediate clue as to which onboard guidance system is functioning correctly, should there be a serious deterioration in the performance of either one of the onboard systems.

In summary then, the MSFN will provide some assistance as regards "ascent switchover monitoring," "navigation", and determination of targeting.

AUSTERITY IN ONBOARD COMPUTER

During the current stage of the Apollo project, it has become increasingly clear that restraint must be used in the selection of programs to be formulated for the onboard computers. There are two important reasons for this and they are: (1) to avoid overflowing the capacity of the computer, and (2) maintenance of schedule.

1. Approximately 1-1½ years before launch, it is important that the summation of the estimates of the word capacity for each program leaves something of the order of a 15% margin relative to the ultimate capacity of the computers. There are two good reasons for this. The first one is that when a program is only partially formulated, then there is a degree of uncertainty as to the final capacity allocation needed for that program and also a nominal 15% margin at this stage is desirable because secondly, when the overall margin of the computer capacity is in jeopardy, inherently each program has to be formulated with additional care in order to restrict capacity demands. This means that there is a tendency for all programs to be formulated somewhat more slowly than they normally would than when it is known that the margin is not in jeopardy. This facet, in itself, causes a delay in schedule.

2. It is important that the first draft of each program is formulated at the earliest time possible. This is because it is important that each program is fitted into an

integrated formulation such that controlling and sequence selector programs can be tested in conjunction with the individual programs. Furthermore, a period is necessary for checking, testing, and verifying each program and the integrated formulation of all programs.

3. Recent surveys of the selection of programs for the onboard computers have revealed that further austerity in onboard computer capability is needed. It is now apparent that some main programs previously selected with the best intentions and there were particular reasons why each one in itself was desirable, however, not mandatory for onboard computer capability. These main omissions are:

(1) Self checking restricted: It was intended that there should be automatic testing for malfunction of the computer with diagnostic warnings displayed.

(2) Logic to avoid gimbal lock not included: This is a feature of the altitude change programs to insure that the rotations were such that gimbal lock was not incurred. Crew monitoring of the 8 Ball now becomes a more critical function to insure the same objective.

(3) CMS: No concentric sequence logic: In all non time critical situations, a concentric rendezvous flight plan is the preferred mode of rendezvous, however, the capability for this has had to be excluded from the CMC state to be entered into the LGC, which is then called upon to work out the concentric sequence for the CSM, and this is advised by voice to the CSM crewman. He in turn enters the required targeting through the CSM DSKY.

(4) Absence of flight plan prediction: Trajectory determination has always been, and is restricted to free coast prediction.

(5) Point return: This capability has been deleted from the "return to earth" program, however, the time critical mode may be manually iterated through the DSKY by entering delta V capabilities less than the maximum. This effectively produces longitudinal control of the splash point, in that convergence is easy enough if the variation of the longitude of the splash point is noted in its relationship with the "delta V capability" input to the DSKY. The iteration logic is described in Targeting - Type 3. When point return capability is deleted, and with this proviso it is not necessary that the nominal recovery point is contained within the return trajectory plane, then the iteration logic may be considerably simplified.

(6) MSFC iterative steering: In order to insure a lack of serious transients in the event of the spacecraft having to take over guidance of the Saturn, it was considered necessary that the CMC computer contained the MSFC iterative steering logic. This requirement has been deleted because: (1) it was not considered that crew safety was seriously impaired, in that safe aborts are still possible. This entails a CSM separation from the Saturn and therefore, in this case, the objectives would no longer be possible, however the probability is low. (2) The probability of detection of trouble with the Saturn platform during the translunar burn associated with a nondetection during parking orbit, is considered to be extremely low. If a Saturn platform failure is detected during the parking orbit and it must be remembered that it is only the Saturn platform which is being backed up, i.e., the Saturn computer is an essential component when the spacecraft has taken over, the spacecraft can use a simpler scheme without fear of transients. This scheme is described in Targeting - Type 1, and the simpler MIT cross product steering is applied. It must be remembered that the spacecraft only takes over in contingency circumstances.

TIME HISTORY OF AN ONBOARD PROGRAM (Figure 18)

Figure 18 indicates a typical time history of an onboard program. The initial functional "requirements" must be issued something like $18\frac{1}{2}$ months before launch. Manned Spacecraft Center issues the requirements to the contractor, in the case of the prime guidance systems, MIT. Subsequent to the issuance of the requirements, program modular formulation begins which is followed by unit checking and testing in each case. With the complexity of modern onboard integrated programs, this process of modular formulation and unit checking and testing will take on the order of $10\frac{1}{2}$ months. At the end of this time the "first draft of integrated formulation" is achieved. Now bit by bit testing of the whole integrated program begins, together with "hybrid simulations" involving the checking of warning lights, etc. These processes will take on the order of 3 months by the end of which time the "flight program release" may be issued to the contractor responsible for the manufacture of ropes. From the "flight program release" to the "delivery of flight ropes at Cape", will incur an additional $1\frac{1}{2}$ months. Subsequent to the delivery of the flight ropes at the Cape, a period of intense computer interface testing with other systems dependent upon the onboard computer, ensues. With one or more tests occurring daily, a busy schedule lasting an additional $3\frac{1}{2}$ months, is unavoidable. When the interface testing has been achieved then the onboard computers are ready for launch activity.

One unescapable milestone should be recognized and that is that the entire flight program must have been verified at least 5 months before launch. This is the permanent logic in the computer. Of course, the quantitative values of guidance constants, etc., can be inserted into the erasable memory at a much later time before launch, facilitating alternative missions, etc. However, again it is significant that the permanent logic must be ready and verified 5 months prior to launch.

SUMMARY (Figure 19)

1. The onboard capability has been framed such that:

    (1) The selected programs provide a capability that insures crew safety.

    (2) The selected programs provide a capability which significantly enhances mission success, for example the spacecraft guidance capability for translunar injection, which would be used in contingency circumstances.

    (3) The onboard programs have been framed such that they can efficiently use ground information.

2. The ground capability is essential in some cases, particularly is this the case when real time flight planning is required involving both the CSM and LM. It will be remembered that this is incurred with rendezvous cases with abnormal initial phasing. In general, the ground capability contributes to mission success as has been noted in the manned space projects so far and the ground capability provides necessary useful peripheral data.

3. It is found that the combined capability further advances crew safety. This is evident in launch and abnormal rendezvous situations. The combined capability insures more optimum trajectories due in some cases to the increased accuracy and in others, to the increased flexibility of capability, and in general, it is found that the combined onboard and ground capability provides maximum probability of mission success.

Questions and Answers

SOFTWARE COMPATIBILITY WITH LUNAR MISSION OBJECTIVES

Speaker: Morris V. Jenkins

1. Dr. Mueller - Is there a budget for the CSM and LM computers?

   ANSWER - Yes. There is now about a 3,000 word pad in the CSM and a 5,000 word pad in the LM.

2. Mr. North - Would you ever use remaining SPS propellant to reduce entry velocity?

   ANSWER - No.

NASA-S-66-6847 JUN

SOFTWARE CONSIDERATIONS

- ONBOARD ADEQUACY INSURES CREW SAFETY
  - INDEPENDENT CAPABILITY TO RETURN
- MISSION SUCCESS CONTRIBUTIONS
  - TLI CAPABILITY
  - ACCEPTS GROUND UPDATES
- GROUND CAPABILITY ESSENTIAL IN SOME CASES
- GROUND CONTRIBUTES TO MISSION SUCCESS AND PROVIDES NECESSARY PERIPHERAL DATA
- COMBINED SOLUTIONS:
  - ADVANCES CREW SAFETY
  - OPTIMIZED SOLUTIONS
  - MAXIMUM PROBABILITY OF MISSION SUCCESS
- ADOPTION OF SIMPLE MIT TARGETING SCHEMES HAS CONTRIBUTED

Fig. 1

NASA-S-66-6553 JUN

ADEQUACY OF ONBOARD SOFTWARE CAPABILITY

- CREW SAFETY
  - INDEPENDENT CAPABILITY TO RETURN
  - CAPABILITY TO ACCEPT GROUND UPDATES
  - COMPATIBILITY OF SOFTWARE IN PGNCS AND AGS
  - DOUBLE CHECK ON RELATIVE STATE
  - LM CAN DIRECT CMS ACTIVE RENDEZVOUS
  - SOFTWARE LOGIC FACILITATES MEANINGFUL CREW CHECKS
- MISSION SUCCESS
  - NECCESSARY STEERING
  - TRANSLUNAR INJECTION
  - ACCEPTANCE OF ALTERNATIVE TARGETING FROM THE GROUND

Fig.

NASA-S-66-6542 JUN

BASIC ONBOARD G AND N SOFTWARE CAPABILITY

CREW G AND N MANAGEMENT →				DISPLAYS
WHERE ARE WE ?	NAVIGATION	OBSERVATIONS →	TRAJECTORY DETERMINATION PROCESSORS →	CURRENT SMOOTHED POSITION AND VEL DETERMINATION
WHERE WILLWE BE ?	DEAD RECKONING	EPHEMERIS →	TRAJECTORY PREDICTION PROCESSORS →	BEST ESTIMATE OF FUTURE POSITION AND VELOCITY
WHAT CHANGE OF COURSE IS NEEDED ?	CHANGE OF COURSE DETERMINATION	CHANGE OF COURSE DETERMINATION PROGRAMS →	PRE-THRUST PROGRAMS →	TARGET OBJECTIVES FOR THRUST
IS THE PRIME G AND N SYSTEM CONTROLLING THE PROPULSION CORRECTLY ?	MANEUVER CONTROL	CREW INITIATION OR PROCEED CONTROL →	THRUST PROGRAMS STEERING EQUATIONS →	ACHIEVEMENT OF DESIRED CUTOFF CONDITIONS

Fig. 3

NASA-S-66-6555 JUN

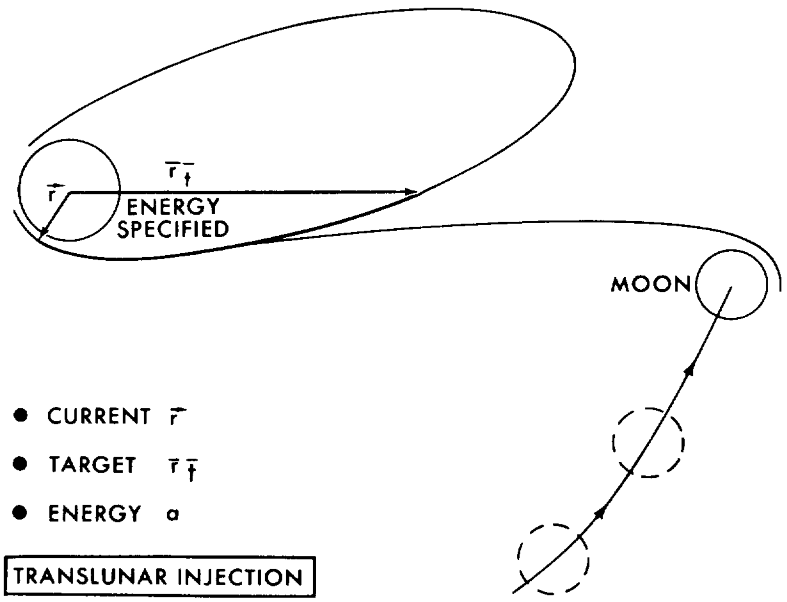
AMPLIFICATION OF THE BASIC PROGRAMS

TRAJECTORY DETERMINATION PROCESSORS	BASIC COMPONENT: KALMAN FILTER
TRAJECTORY PREDICTION PROCESSORS	● ENCKE METHOD ● INCREMENTS INTEGRATED ● LARGER STEPS FOR THE SAME ACCURACY
PRE-THRUST PROGRAMS	● SET UP THE TARGETING ● THRUST COMMAND ATTITUDE ● PREFERRED PLATFORM ALIGNMENT
THRUST PROGRAMS	● X PRODUCT STEERING ● E GUIDANCE ● LOCAL VERTICAL COORDINATES

Fig.

NASA-S-66-6551 JUN

TARGETING - TYPE 1


$\bar{r}_t$
$\bar{r}$
ENERGY
SPECIFIED
MOON

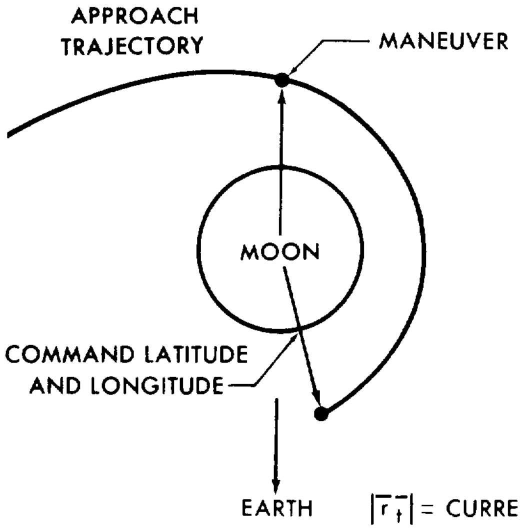
- CURRENT $\bar{r}$
- TARGET $\bar{r}_t$
- ENERGY a

TRANSLUNAR INJECTION

Fig. 5

NASA-S-66-6547 JUN

TARGETING TYPE 1


APPROACH
TRAJECTORY
MANEUVER
MOON
COMMAND LATITUDE
AND LONGITUDE
EARTH

- CURRENT $\vec{r}$
- TARGET $\bar{r}_t$
- ENERGY a

LUNAR ORBIT INSERTION

REPEAT FOR PLANE CHANGE

$|\bar{r}_t| = \text{CURRENT BURN} \, |r|$

$a = |\bar{r}_t|$

Fig. 6

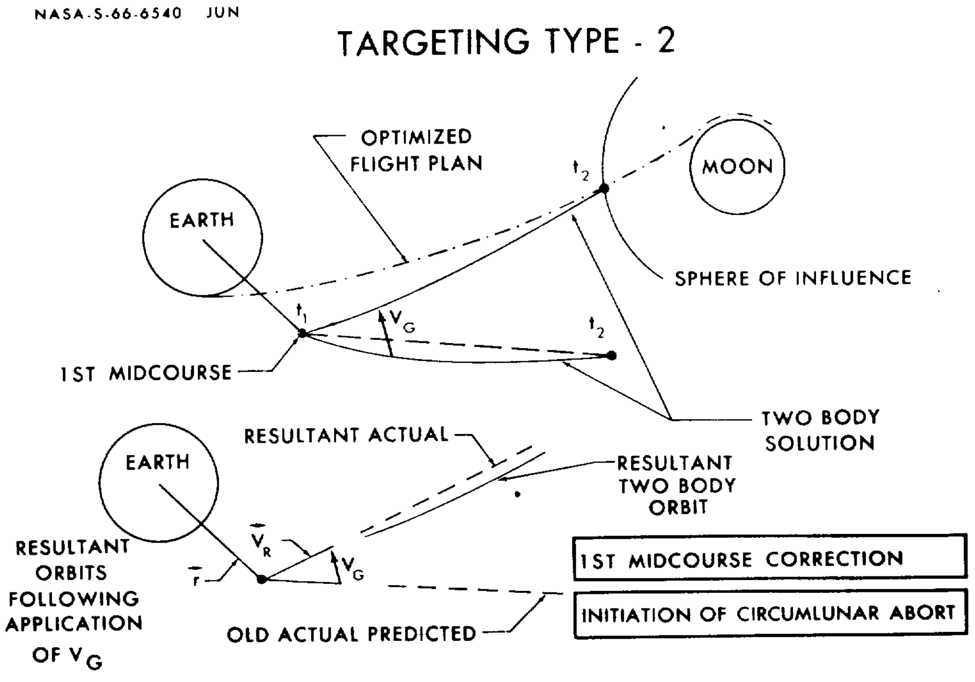

NASA-S-66-6540 JUN
TARGETING TYPE - 2
OPTIMIZED FLIGHT PLAN
EARTH
MOON
t_2
SPHERE OF INFLUENCE
t_1
V_G
t_2
1ST MIDCOURSE
TWO BODY SOLUTION
RESULTANT ACTUAL
EARTH
RESULTANT TWO BODY ORBIT
RESULTANT ORBITS FOLLOWING APPLICATION OF V_G
$\vec{V}_R$
$\vec{r}$
V_G
1ST MIDCOURSE CORRECTION
INITIATION OF CIRCUMLUNAR ABORT
OLD ACTUAL PREDICTED

Fig. 7

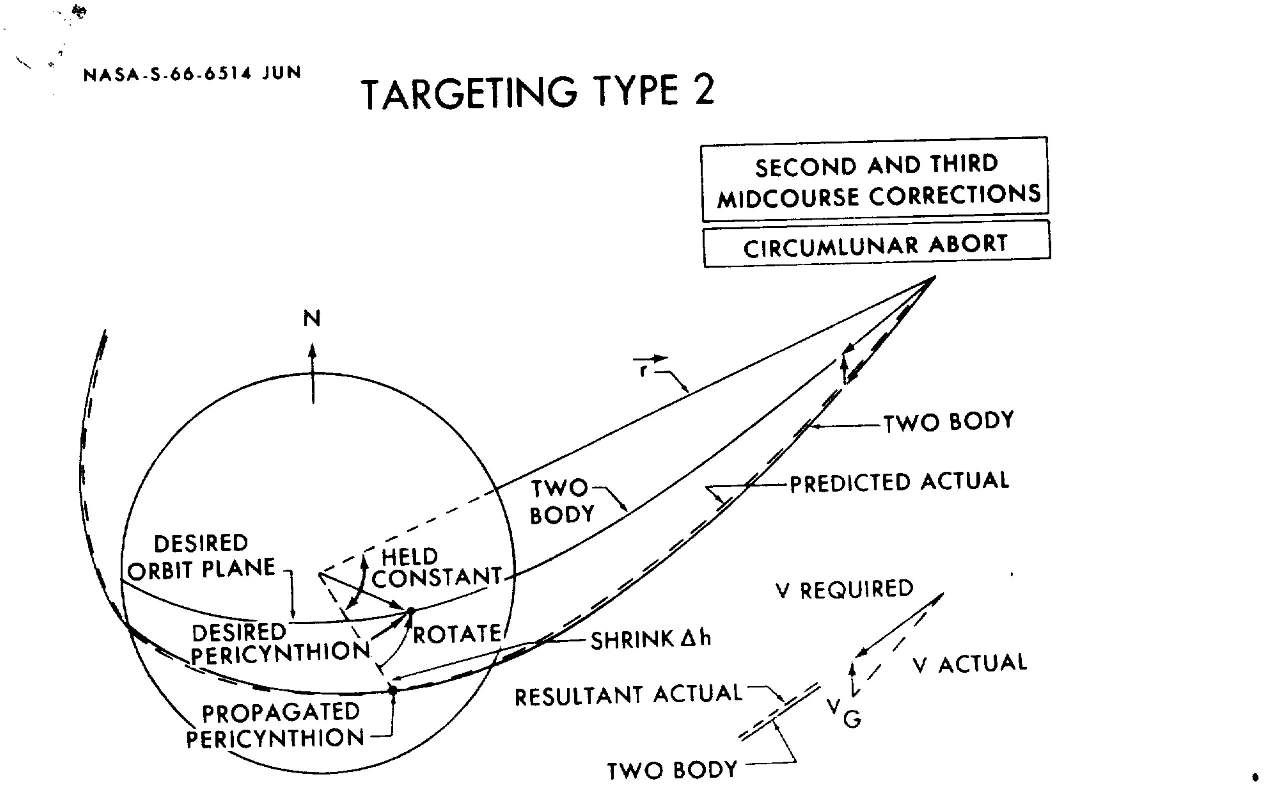
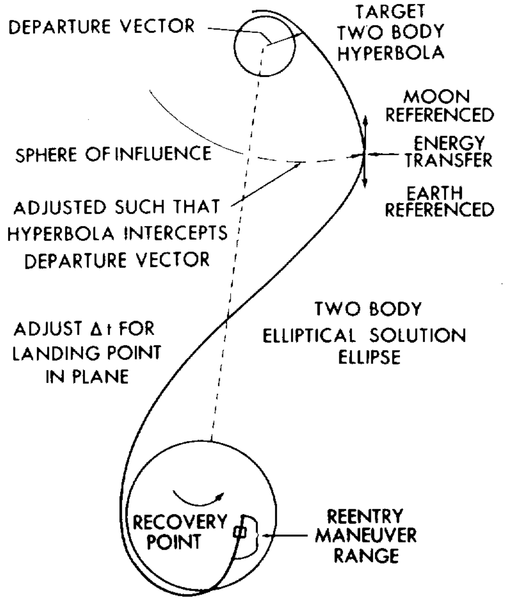

NASA-S-66-6514 JUN
TARGETING TYPE 2
SECOND AND THIRD MIDCOURSE CORRECTIONS
CIRCUMLUNAR ABORT
N
$\vec{r}$
TWO BODY
TWO BODY
PREDICTED ACTUAL
DESIRED ORBIT PLANE
HELD CONSTANT
DESIRED PERICYNTHION
ROTATE
SHRINK Δh
V REQUIRED
V ACTUAL
RESULTANT ACTUAL
V_G
PROPAGATED PERICYNTHION
TWO BODY

Fig. 8

NASA-S-66-6565 JUN

TARGETING TYPE 3


DEPARTURE VECTOR
TARGET TWO BODY HYPERBOLA
MOON REFERENCED
SPHERE OF INFLUENCE
ENERGY TRANSFER
EARTH REFERENCED
ADJUSTED SUCH THAT HYPERBOLA INTERCEPTS DEPARTURE VECTOR
TWO BODY ELLIPTICAL SOLUTION ELLIPSE
ADJUST Δt FOR LANDING POINT IN PLANE
RECOVERY POINT
REENTRY MANEUVER RANGE

CONIC ITERATION

Δt
S OF I
POINT

- REENTRY CONDITIONS
- RECOVERY POINT IN PLANE
- STATE MATCH AT S OF I
- BACK TRACKED HYPERBOLA INTERCEPTS CURRENT STATE

PRECISION ITERATION

- BACK INTEGRATION FROM EARTH
- FORWARD INTEGRATION FROM DEPARTURE VECTOR
- ITERATE TO MATCH AT S OF I

TRANSEARTH INJECTION

TRANSEARTH ABORT

TRANSEARTH MIDCOURSE

Fig. 9

NASA-S-66-6531 JUN

DESCENT AND ASCENT TARGETING

DESCENT

- **PROFILE SHAPING TARGETING**
  - **INTERMEDIATE POSITION TARGETING**
  - **WITH AN ASSOCIATED VELOCITY VECTOR**
  - **TOGETHER WITH AN ASSOCIATED ACCELERATION VECTOR**

ASCENT

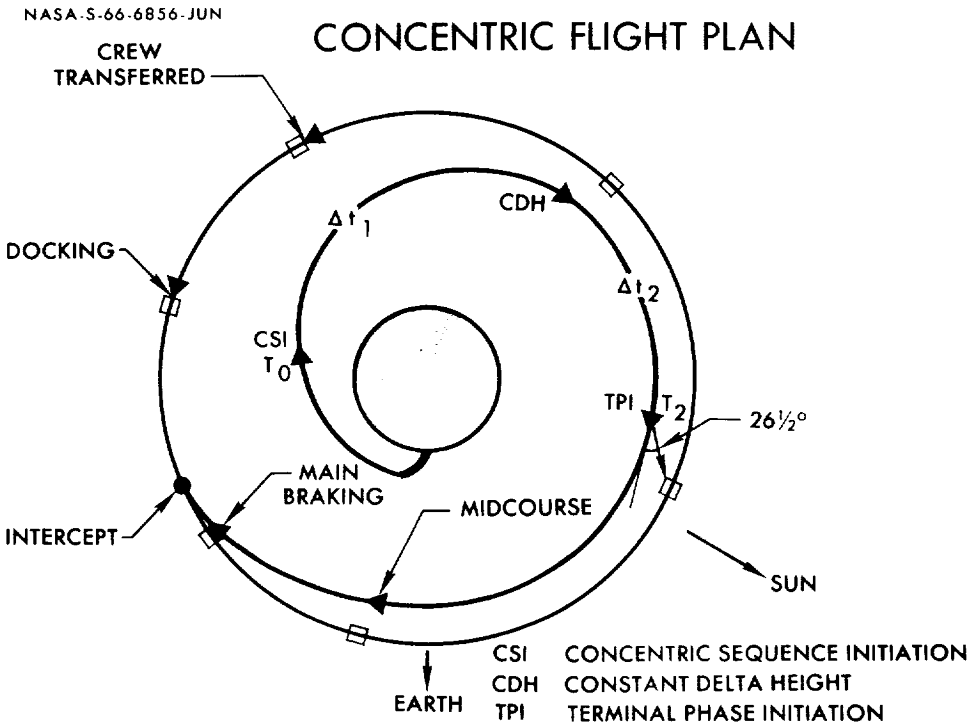
- **ORBITAL TARGETING**
  - **CUTOFF POSITION AND VELOCITY VECTOR IN CSM PLANE**
  - **τ SPECIFIED**
  - **$|V|$**
  - **CUTOFF $|h|$ SPECIFIED**
  - **DOWNRANGE FREEDOM IN CUTOFF POSITION WITH SOME RESULTANT FREEDOM IN DIRECTION OF LINE OF APSIDES**

PRINCIPLES OF STEERING COMMON TO BOTH: COMMAND THRUST VECTOR DERIVED FROM EXPLICIT EXPRESSIONS INVOLVING CURRENT ACCELERATION VECTOR, AND DIFFERENTIALS OF CURRENT AND TARGET CONDITIONS, i.e., ΔT_{GO}

$\underline{\Delta V}$

$\underline{\Delta r}$

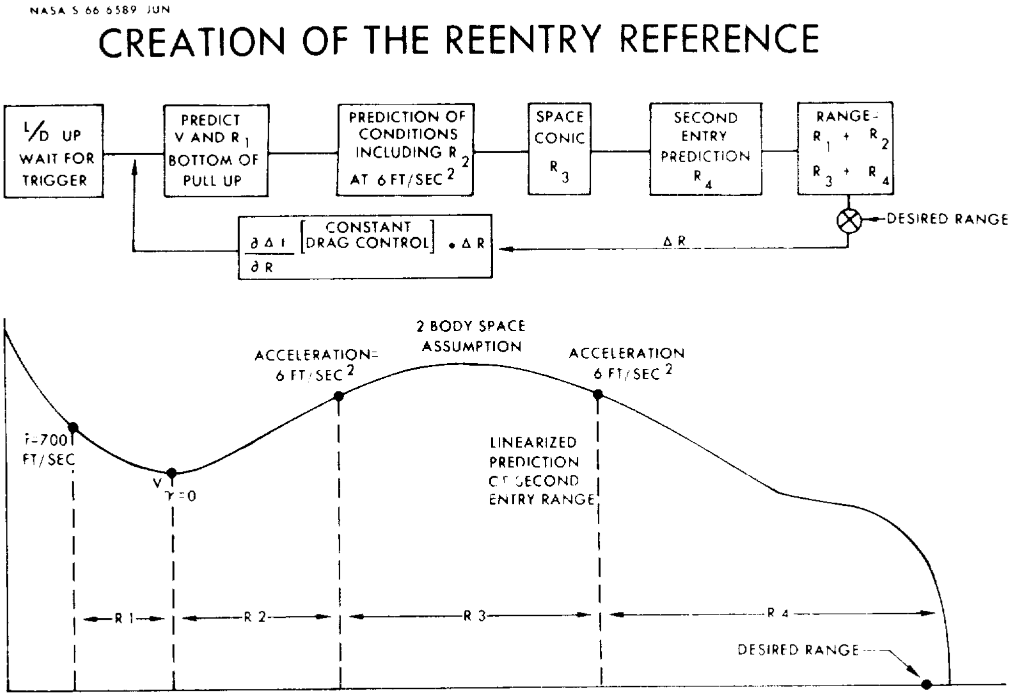
Fig. 10


NASA-S-66-6856-JUN

CONCENTRIC FLIGHT PLAN

CREW TRANSFERRED
DOCKING
INTERCEPT
MAIN BRAKING
MIDCOURSE
CSI
T0
Δt1
CDH
Δt2
TPI
T2
26½°
SUN
EARTH
CSI CONCENTRIC SEQUENCE INITIATION
CDH CONSTANT DELTA HEIGHT
TPI TERMINAL PHASE INITIATION

Fig. [illegible]


NASA S 66 6589 JUN

CREATION OF THE REENTRY REFERENCE

L/D UP WAIT FOR TRIGGER
PREDICT V AND R1 BOTTOM OF PULL UP
PREDICTION OF CONDITIONS INCLUDING R2 AT 6 FT/SEC2
SPACE CONIC R3
SECOND ENTRY PREDICTION R4
RANGE = R1 + R2 + R3 + R4
DESIRED RANGE
ΔR
∂Δt/∂R [CONSTANT DRAG CONTROL] • ΔR
2 BODY SPACE ASSUMPTION
ACCELERATION= 6 FT/SEC2
ACCELERATION 6 FT/SEC2
Ṙ=700 FT/SEC
V γ=0
LINEARIZED PREDICTION OF SECOND ENTRY RANGE
R 1
R 2
R 3
R 4
DESIRED RANGE

Fig. [illegible]

NASA S 66 6588 JUN

REAL TIME ITERATIVE CONTROL

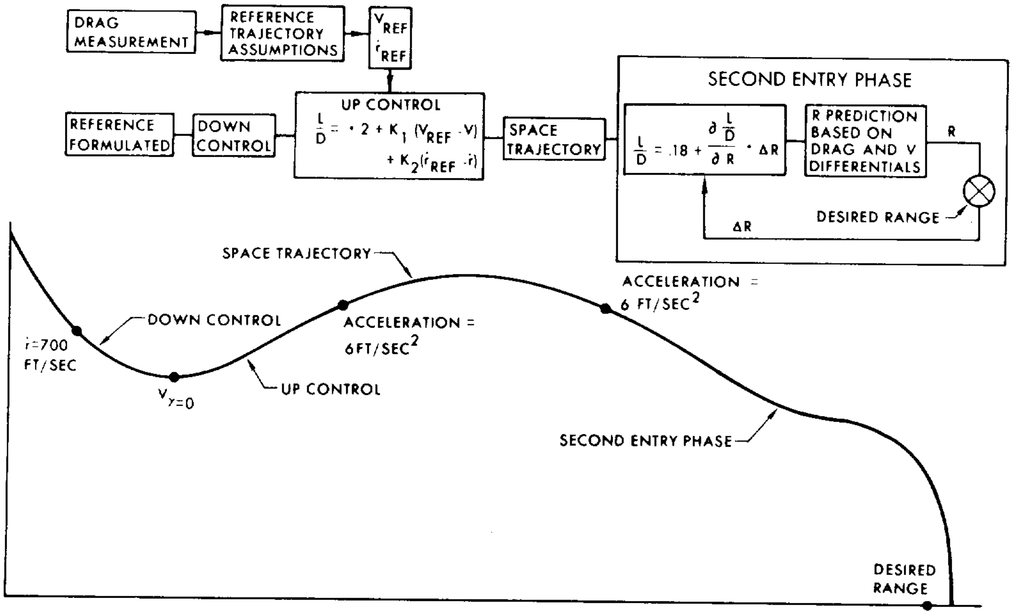

DRAG MEASUREMENT
REFERENCE TRAJECTORY ASSUMPTIONS
V_{REF}
$\dot{r}_{REF}$
REFERENCE FORMULATED
DOWN CONTROL
UP CONTROL
$\frac{L}{D} = .2 + K_1(V_{REF} - V) + K_2(\dot{r}_{REF} - \dot{r})$
SPACE TRAJECTORY
SECOND ENTRY PHASE
$\frac{L}{D} = .18 + \frac{\partial \frac{L}{D}}{\partial R} \cdot \Delta R$
R PREDICTION BASED ON DRAG AND V DIFFERENTIALS
R
DESIRED RANGE
ΔR
SPACE TRAJECTORY
DOWN CONTROL
$\dot{r}$=700 FT/SEC
$V_{\dot{r}=0}$
UP CONTROL
ACCELERATION = 6FT/SEC^2
ACCELERATION = 6 FT/SEC^2
SECOND ENTRY PHASE
DESIRED RANGE

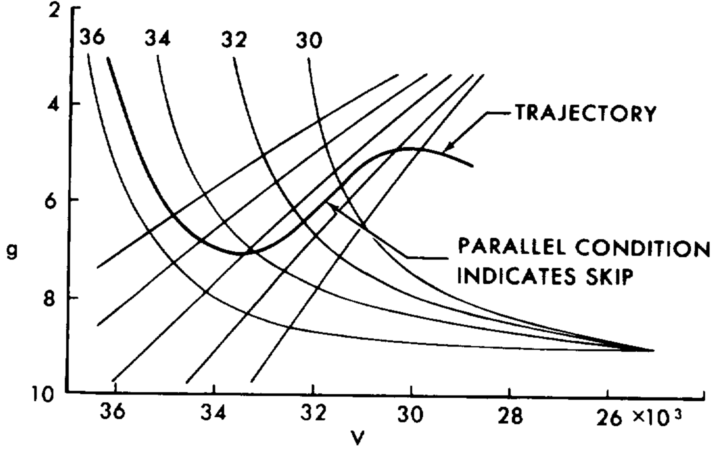
Fig. 13

NASA-S-66-6550 JUN

ENTRY MONITORING SYSTEM LOGIC

- ● CORRIDOR VERIFICATION
- ●(g, V) MONITORING

0.05 g–10 SECOND DELAY–g
CRITICAL VALUE >(g —0.05)



- ● FURTHER DEVELOPMENT - RANGE CONTROL LINES

Fig. 14

NASA-S-66-6530 JUN

SOFTWARE CAPABILITY NOT ONBOARD

- PRE-MANEUVER SIMULATION
- REAL TIME TOTAL FLIGHT PLAN EVALUATION
  - PREDICTION OF SEQUENCE EVENTS AND FUEL RESERVES
- CAPABILITY TO RE-OPTIMIZE THE WHOLE FLIGHT PLAN
- COMBINED MANEUVERS FOR ANY TIME LIFT-OFF
- PERIPHERAL INFORMATION
  - LOOK ANGLES/SIGHTING DATA
  - LIGHTING CONDITIONS FOR RENDEZVOUS CONTRAST SHADOW CONDITIONS FOR DESCENT
  - TRAJECTORY INFORMATION FOR OPS FACILITIES
  - DIFFERENTIAL RADIATION PREDICTION
- COMBINED O D SOLUTIONS FROM HYBRID OBSERVATIONS WITH APPROPRIATE WEIGHTING

Fig. 15

NASA-S-66-6512 JUN

GROUND SUPPLEMENTARY AND BACKUP G AND N FUNCTIONS

- TARGET DETERMINATION AND SUPPLY
- PILOT MONITORING INFORMATION FOR MANUAL MANEUVERS
- MORE EXTENSIVE EXPLORATION OF THE POSSIBILITIES ESPECIALLY IN CONTINGENCY SITUATIONS
- UMPIRING FUNCTION: PGNCS AND AGS
  MSFN $\dot{r}$ INFORMATION TO ASSIST IN LM
  - ASCENT SWITCHOVER MONITORING
  - NAVIGATION
  - DEAD RECKONING

Fig. 16

NASA-S-66-6544 JUN

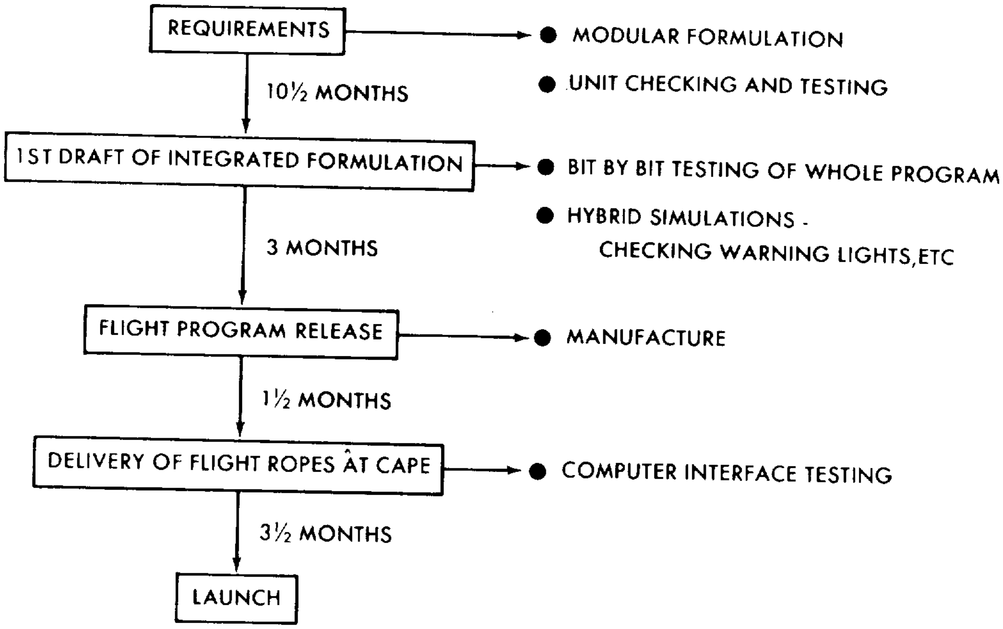
AUSTERITY IN ONBOARD COMPUTER

- CAPACITY
- SCHEDULE
  - PROGRAM FORMULATION
  - CHECKING
  - TESTING
  - VERIFICATION
- MAIN OMISSIONS
  - SELF CHECKING RESTRICTED
  - LOGIC TO AVOID GIMBAL LOCK NOT INCLUDED
  - CMC: NO CONCENTRIC SEQUENCE LOGIC
  - ABSENCE OF FLIGHT PLAN PREDICTION
  - POINT RETURN
  - MSFC ITERATIVE STEERING

Fig. 17

NASA-S-66-6552 JUN

TIME HISTORY OF AN ONBOARD PROGRAM



REQUIREMENTS

- MODULAR FORMULATION
- UNIT CHECKING AND TESTING

10½ MONTHS

1ST DRAFT OF INTEGRATED FORMULATION

- BIT BY BIT TESTING OF WHOLE PROGRAM
- HYBRID SIMULATIONS - CHECKING WARNING LIGHTS, ETC

3 MONTHS

FLIGHT PROGRAM RELEASE

- MANUFACTURE

1½ MONTHS

DELIVERY OF FLIGHT ROPES AT CAPE

- COMPUTER INTERFACE TESTING

3½ MONTHS

LAUNCH

Fig. [illegible]

NASA-S-66-6518 JUN

SUMMARY

- ONBOARD
  - CAPABILITY INSURES CREW SAFETY
  - CAPABILITY IS EFFICIENTLY USED TO ENHANCE MISSION SUCCESS
  - CAN EFFICIENTLY TAKE GROUND INFORMATION
- GROUND
  - CAPABILITY IS ESSENTIAL IN SOME CASES
  - CAPABILITY CONTRIBUTES TO MISSION SUCCESS AND PROVIDES NECESSARY USEFUL PERIPHERAL DATA
- COMBINED ONBOARD AND GROUND
  - CREW SAFETY FURTHER ADVANCED
  - MORE OPTIMUM TRAJECTORIES
  - PROVIDES MAXIMUM PROBABILITY OF MISSION SUCCESS

Fig. 19

COMMUNICATIONS FUNCTIONS

By

Douglas R. Broome, Jr.

COMMUNICATIONS FUNCTIONS

A summary of the spacecraft and launch vehicle functional capabilities is shown in Figures 1 and 2. As can be seen in Figure 1, CSM and LM both provide the S-Band voice telemetry, at both the high and low bit rates, ranging, tracking, and television. One difference in these capabilities is that the LM cannot transmit television except from the lunar surface after the erection of the Lunar Surface Erectable Antenna. In addition to these capabilities, the CSM provides the capabilities of updata, recorded data and voice playback, and scientific data transmission. At VHF, both the CSM and LM have voice communications capabilities and the LM has the capability of transmitting low bit rate telemetry from the LM to the CSM during descent and ascent. The CM has both VHF and HF two-way voice and beacon transmission capability for recovery phases.

The launch vehicle Instrument Unit (IU) provides the capabilities of telemetry, tracking, ranging, and updata at S-Band, as well as telemetry at VHF and tracking at C-Band. The three launch vehicle propulsion stages have VHF telemetry capabilities and range safety destruct.

Figures 3 and 4 show the Manned Space Flight Net (MSFN) support capabilities to support the spacecraft and launch vehicle communications functions. Figure 3 illustrates the ground based station capabilities and Figure 4 illustrates the ship and aircraft capabilities. It should be noted that the ground based stations have identical capabilities except that the Madrid, Canberra, and Goldstone stations have the ability to recieve television from lunar distance. It should also be noted that not all stations can track the IU C-Band transponder. The reentry ships do not have updata capabilities and they utilize 12 foot parabolas rather than 30 foot parabolas. The recovery forces have, in addition to VHF and HF voice and beacon capabilities, S-Band direction finding capability.

Figure 5 represents a ground track for the first two orbits of the nominal lunar mission. The circles shown represent the station coverages for a one hundred nautical mile orbit. Not shown on this map is the fact that for altitudes above 8,000 nautical miles, the Madrid, Goldstone, and Canberra coverages overlap, thus providing continuous, unbroken communications with the spacecraft.

The Radio Frequency Systems have been broken down into several figures in order to make it easier to understand their functions in meeting their respective Module requirements. Figures 6 and 7 represent the data sources for the two Modules. As shown in Figure 6, the Central Timing Equipment, PCM Telemeter, Signal Conditioner, and the Pre-Modulation Processor all contain redundant circuit elements for critical circuits such as power supplies, voice modulators, telemetry modulators, and digital counting blocks. The PCM telemetry, in particular, contains both circuit and block redundant elements. Also, the CSM has three sets of microphones and amplifiers, one set for each astronaut; each of these sets is redundant.

Figure 7 represents the data sources for the LM. Again, circuit redundancy is provided. The LM voice recorder has no playback capabilities. It will be brought back to earth, however.

Figures 8 and 9 are block diagrams of the two respective Modules' S-Band systems. The LM S-Band system is comprised of a coherent transponder, a power amplifier, a diplexer, and an antenna system consisting of two Omni elements, an In-Flight Steerable High-Gain antenna and a Lunar Surface Erectable Antenna. The transponder and power amplifier are completely redundant and there is by-pass switching around the power amplifier to allow for prime power conservation when high bit rate telemetry is not required. The Steerable Antenna is a 2 foot diameter parabola providing a transmit gain of 20.3 db and a receive gain of 16.5 db with a beam width of 13 degrees. The Erectable Antenna, in conjunction with the 750 milliwatt transmitter, allows transmission of voice and low bit rate telemetry to the MSFN without the use of the power amplifier, thus resulting in considerable prime power conservation. This antenna, in conjunction with the power amplifier, also allows transmission of high-quality real-time television pictures to the MSFN from the lunar surface.

Figure 9 presents the CSM S-Band system. This system is essentially the same as that of the LM with the addition of a separate S-Band FM transmitter. In the CSM, as in the LM, the coherent transponder is redundant. The power amplifiers are not shown as being redundant in that, at times, they are used in parallel; that is, power amplifier number 1 being driven by the FM transmitter and power amplifier number 2 by the PM transmitter. However, switching is such that the number 1 power amplifier can be switched to back up the number 2 power amplifier.

The CSM antenna system consists of four Omni antenna elements and a High-Gain antenna. This High-Gain antenna is made up of four 31-inch diameter parabolas and a crossed dipole array. The transmission system has three operating modes--wide, medium, and narrow, with attendant beam widths of 45 degrees, 12.3 degrees, and 4.4 degrees at the half power points. The receiving modes are coarse and fine--the beam widths being 45.5 degrees and 5.5

degrees at the half power points. If the transmit mode switch is in the "Wide" position, the receiving antenna mode will remain in the "Coarse" position upon acquisition of an RF signal. However, if the transmit mode switch is in the "Medium" or "Narrow" position, the receiving mode will automatically switch from "Coarse" to "Fine" upon signal acquisition. Use of the High-Gain antenna begins at an altitude of approximately 2,500 nautical miles after the transposition but prior to docking.

The three beam widths are derived as follows: The wide beam width is provided by using a crossed dipole array; the medium beam width by using one of the 31-inch parabolas; and the narrow beam width by driving the four 31-inch parabolas in parallel.

Figures 10 and 11 represent the VHF systems for the CSM and LM. These two systems are essentially identical in that both utilize a transmitter and receiver at 296.8 MC, and a transmitter and receiver at 259.7 MC. Usage of these equipments will be explained later. The CSM VHF Omni system consists of two scimitar elements located on the Ascent Stage. Note that the LM has an additional VHF antenna which is used for Extra Vehicular Crewmen (EVC) communications. This antenna is a discone mounted on an erectible boom located on the Ascent Stage. EVC signals are picked up by the 259.7 MC receiver, de-modulated, and then used to modulate the S-Band link to earth.

Figure 12 presents the equipments used on board for recovery phase communications. These equipments consist of an HF transceiver/beacon, a VHF beacon, and a VHF survival transceiver/beacon. This latter equipment is provided as part of the astronaut's survival equipment package. In addition, the 296.8 MC transmitter and receiver is used for two-way voice communications during the recovery phase. These equipments are used with three antenna systems which are erected during the recovery phase. The HF recovery antenna is a 16-foot spring steel leafed antenna which is erected following attainment of spacecraft Flotation 1 position. The two VHF recovery antennas are approximately 14 inches long, are made of spring steel such as is used in measuring tapes, and are erected immediately following main parachute deployment in order to allow maximum range communications with recovery forces. One of the recovery antennas has a manual connection provision which will allow use of the astronaut's survival transceiver/beacon with the spacecraft antenna.

Figures 13 and 14 present the pertinent parameters of the two S-Band systems. The CSM utilizes a coherent transponder operating at the frequencies shown on Figure 13 and a separate S-Band transmitter operating at 2272.5 MC. These radio equipments operate with either S-Band Omni antenna system or the High-Gain antenna. Three RF power transmission modes are available: 250 milliwatts, 5 watts, and 20 watts. Power level is selected as a function of range and data rate.

The LM S-Band system utilizes a coherent transponder, operating at the frequencies shown on Figure 14; no separate transmitter is provided. This system is used in conjunction with one of the three LM antenna systems: the Omni, In-Flight High-Gain, or the Lunar Surface Erectible Antenna. Two RF power levels are provided: 750 milliwatts and 20 watts. Power level selection for the LM is based on data rate and bandwidth requirements. The erectible antenna is used for prime power conservation and also to allow the transmission of high quality real-time television pictures from the lunar surface.

Figures 15 and 16 present the characteristics of the two VHF systems. Power output for all transmitters is 5 watts each. Each in-flight antenna system consists of two radiating elements which allow essentially omnidirectional coverage. The CM also has two VHF recovery antennas and the LM has one VHF antenna for EVC operations.

Figure 17 presents the characteristics of the HF recovery equipment. The transmit and receive frequencies are 10.006 MC for both voice and beacon. Transmission power and peak-envelope power is 20 watts in the single-sideband mode and 5 watts in the double-sideband mode. The antenna used is the 16-foot erectible whip.

The Launch Vehicle RF requirements are presented in Figures 18 and 19. The VHF telemetry requirements are met by fifteen transmitters located by stage as shown. Power output for each transmitter is 18 watts through an Omni antenna system.

The C-Band tracking requirements for the launch vehicle are met using a pulse-type radar transponder and an Azusa transponder. The launch vehicle S-Band system is much like that of the CSM in that it consists of a coherent transponder and a separate FM transmitter operating at the frequencies shown here. Note that the frequencies of the coherent transponder are the same as those of the LM. However, no frequency interference problems are expected due to the physical separation of the launch vehicle and LM when the LM system is activated. Both launch vehicle S-Band transmitters operate at 20 watts through Omni antenna systems.

Figure 20 describes the Lunar Surface Experiment Package (LSEP). The uplink frequency of this package is 2119 MC, and three downlinks are shown. However, a single LSEP will only utilize one downlink at a time. Since the lifetime of these packages is one year, it is expected that during a year multiple LSEP's will be operating simultaneously. Power output for each package is one watt through a helical antenna. Design life of the packages is one year with 100 per cent utilization. All experiments are multiplexed on one PCM downlink channel at 1060 bits per second to a 30-foot MSFN antenna. The uplink command system provides for system on-off, experiment activation, experiment mode changing, and additional command capabilities not yet allocated.

The communications capabilities provided by the combined space vehicle can be summarized as follows:

1. Earth-Spacecraft two-way voice.
2. Earth-Spacecraft-EVC voice relay and conference.
3. Telemetry from launch vehicle, spacecraft modules, and LSEP to earth.
4. Updata to launch vehicle, CSM, and LSEP from earth.
5. Tracking information enabling utilization of ground-based navigation capabilities.
6. Television from the CSM and the lunar surface to earth.
7. Recorded data and voice playback from CSM to earth.
8. Recovery location aids.

Figures 21 and 22 present the primary and secondary means of providing voice communication between the Spacecraft Modules, EVC's and the MSFN. The primary communication link between the Modules and the MSFN in-flight are the two S-Band coherent systems. In addition, in-flight, the CSM has the ability to dump via the S-Band FM transmitter any recorded data that it has acquired. For recovery, CSM has additional voice capabilities at 243.0 MC and 10.006 MC. The backup voice capabilities between the modules and earth are provided by the redundant S-Band systems in each Module. In addition, the CSM 296.8 MC equipment can be used as a second voice backup during earth orbital phases of the lunar mission.

The CSM-LM primary voice communications are provided with the 296.8 MC equipments in the Modules. The backup to this system is provided by using the remaining VHF equipments as required depending on the nature of the failure experienced. Voice communications between the modules and an EVC are provided by transmitting to the EVC at 296.8 MC and receiving at 259.7 MC. The backup to this system results in a reversal of these two frequencies. This frequency reversal was found necessary in order to reduce the physical size and complexity of the suit equipment.

Figure 23 illustrates the voice relay and conference capabilities provided by the Spacecraft-EVC-MSFN equipments. The simplest way to describe these capabilities is to say that, given the condition of one or two EVC's on the lunar surface and the CSM in line-of-sight of the LM, any party in the loop can talk to any other party in the loop by relaying through the modules and ground stations as required. Any party can interrupt a conversation which is going on merely by speaking up. It is impossible for any party to lock up the system and prevent another party from entering it, such as has happened in previous manned space flights.

The space vehicle telemetry capabilities are presented in Figure 24. Primary telemetry capability is provided for the launch vehicle during the launch and orbital phases using the VHF/PCM equipment, and during injection and post-injection using S-Band/PCM equipment. During launch, there is no backup to the VHF capability. In orbit, the S-Band FM transmitter provides a backup to the VHF equipment. During the injection and post-injection phases, the coherent S-Band system in the launch vehicle provides a backup to the S-Band transmitter.

Primary telemetry transmission for the CSM and LM are provided using the coherent S-Band equipments. In addition, the CSM can dump recorded data via its separate FM transmitter. The LM transmits low bit rate telemetry from the LM and the CSM during descent and ascent for recording and subsequent to playback to earth. The backup to the two S-Band systems are the redundant equipments; also, the two Module telemeters utilize both block and circuit redundancy. There is no backup to the LM-to-CSM low bit rate capability.

Telemetry is received in the primary mode for the EVC over the 259.7 MC carrier and relayed by the appropriate Module to the earth at S-Band. There is no backup to EVC telemetry capability.

The primary launch vehicle tracking aids during launch and orbit phases are the C-Band and Azusa transponders as shown in Figure 25. During the injection and post-injection phases, the CSM S-Band system is the primary tracking system for the launch vehicle. Backups for the launch vehicle for tracking are as shown here. Primary tracking for the CSM and the LM is provided by the S-Band coherent equipments. These equipments allow the MSFN to two-way doppler tracking and ranging on the spacecraft Modules. The range ambiguity inherent in two-way doppler tracking is resolved by transmitting a particular type of pseudo-random binary code, called a Ranging Code, to the vehicle-borne transponder where it is turned around and retransmitted to the MSFN. The code length is greater than the round-trip transit time to the moon. Comparison of the in-coming code at the MSFN station with a stored model of the code allows range ambiguity resolution. The backup for these primary equipments for tracking are the redundant equipments on each Module.

Primary equipment used by the launch vehicle for updata is the S-Band coherent system and a digital decoder, and backup capability is provided by its inherent redundancy. The primary system for updata in the CSM is provided by the coherent S-Band receiver and a Digital Command Decoder. Capabilities provided by this system are as shown in Figure 26. The backup to the RF equipment supporting this capability is the redundant S-Band receiver; there is no backup to the decoder. The voice link will provide the necessary redundancy for computer update and real time commands.

There is no updata capability provided for the LM.

The real time ground command functions to the CSM for control of the data system are as shown in Figure 27. Note that the reset switch command resets all switches to the position that they were in prior to the transmission of any real time commands.

The next several figures present a review of the usage of the RF systems by mission phase. Figure 28 illustrates the use of the coherent S-Band system and a separate S-Band transmitter in conjunction with the S-Band Omni antenna system during the ascent and earth orbital phases of the mission. Figure 29 illustrates the usage of these same equipments for the translunar and transearth coast phases, but note that during these phases these equipments are normally operated in conjunction with the High-Gain antenna on the CSM, rather than the Omni. Failure in the High-Gain antenna would require use of the Omni antenna resulting in the necessity of changing to the low telemetry bit rate. Figure 30 illustrates the usage of equipments when the CSM and LM are separated but with the LM not yet on the lunar surface. The significant difference here is that the LM communicates directly with the CSM by two-way voice and VHF telemetry transmission during descent and ascent phases as well as with the MSFN at S-Band.

Also shown on this figure are the frequencies used by the CSM and an EVC. The next two figures, 31 and 32, illustrate communications between the earth and the two spacecraft modules with the LM on the lunar surface and one EVC out and with two EVC's out. The frequency utilization shown here is the same as that previously described. However, should LM receiver capture or interference problems occur with two EVC's out, the EVC not transmitting telemetry can manually switch to his reverse frequency equipments, which will allow him to carry on duplex voice communications with the other EVC.

The next group of figures present the percentage availability of functional capabilities by mission phases. Referring to Figures 33 and 34, during launch, injection and translunar insertion, the primary functions of voice, high bit rate telemetry, ranging, tracking, and updata are available 100 per cent of the time. No playback will be utilized during these phases. The 26 per cent figures shown for parking orbit are based on the coverage capability for the first two orbits of the nominal mission. The ground-tracking capability is also shown as being 26 per cent. The 74 per cent figure shown for low bit rate telemetry was derived by assuming continuous low bit rate recording when the CSM is in line-of-sight of a ground station. The 20 per cent figure for playback is slightly less than the 26 per cent figure for the other functions because playback to a MSFN station will not begin until lock up of the primary system has been achieved. In Figure 34 note that voice, tracking and updata capabilities are available 100 per cent of the time for the three mission phases shown. The percentages shown for telemetry transmission are based on normal transmission of low bit rate telemetry with short periodic transmissions of high bit rate telemetry in order to conserve prime power during the coast phases. The one per cent figure shown for ranging on both of these charts is representative of the fact that very little actual ranging is required in order to maintain a valid range tally. The two per cent figure shown for TV is a guesstimate of actual TV usage which has not yet been determined, and is not caused by any limitation of equipment or power availability.

The next four figures, Figures 35 through 38, present the lunar phase communications usage. These figures are self-explanatory once it is understood that the 60 per cent figures are based on the fact that for 40 per cent of a lunar orbit the modules are behind the moon. The numbers shown for television usage, both in lunar orbit and on the lunar surface, are again guesstimates and again do not represent either equipment or power limitations. The normal operation for LM telemetry transmission during descent and ascent is for the LM to transmit low bit rate telemetry to the CSM at any time that the LM is not in line-of-sight of the MSFN. Whenever LM-MSFN signal lock-up is achieved, the LM will immediately switch to high bit rate for transmission directly to the MSFN via S-Band.

Figure 39 presents capability utilization for transearth coast and entry phases. The same comments as applied to Figure 34 apply here. Figures 40 and 41 illustrate the utilization of equipment during the recovery phase. The HF equipment is used for long range direction finding; the VHF equipments are used for short range direction finding and communications with recovery aircraft and ships. Finally, a swimmer hardline is available to allow direct voice communications between the astronauts in the command module and frogmen in the water. In Figure 41, the 40 per cent utilization of the VHF beacon is achieved by a power programmer which automatically cuts the power on for two seconds and off for three seconds continuously for 48 hours. Manual programming is employed to enable utilization of the HF equipment in the following manner: six minutes of carrier transmission, two minutes of single sideband voice transmission, and 52 minutes listen only, repeated hourly for 48 hours. This power programming is required in order to provide 48 hour post-landing recovery communications capability.

The last figure presents the status of the Unified S-Band Compatibility Test Program being conducted at the Manned Spacecraft Center with the cooperation of Goddard Space Flight Center, Marshall Space Flight Center, Jet Propulsion Laboratory, and contractor personnel. The Block I Engineering Model, Block II Engineering Model, Launch Vehicle Command and Communication System, and Block I Production Model compatibility tests (using an acutal MSFN receiving system) have been completed, and post-test reports have all been issued except for the latter test report which is due to be published in early July. The Block II CSM-MSFN compatibility test began upon receipt of the Block II S-Band communications system. The gap in the Block II test program results from lending Goddard Space Flight Center the Block II equipment for fly-by tests.

Questions and Answers

COMMUNICATIONS

Speaker: Douglas R. Broome

1. Can the transmission from the lunar surface experiments package be received during the ascent phase and during conclusion of mission?

   ANSWER - Yes.

2. Will second lunar landing affect the operation of the first lunar surface experiments package?

   ANSWER - No. A difference in transmitting frequency has been made.

3. Why is TV used only 5% of the time?

   ANSWER - Usage is based on power limitations.

NASA-S-66-6039 MAY

SPACECRAFT FUNCTIONS

	CSM	LM
S-BAND		
● VOICE	×	×
● TELEMETRY		
● HIGH BIT RATE (51.2 KBS)	×	×
● LOW BIT RATE (1.6 KBS)	×	×
● RANGING	×	×
● TRACKING	×	×
● UP DATA	×	
● VOICE/DATA PLAYBACK	×	
● TELEVISION	×	×
● SCIENTIFIC	×	
VHF		
● VOICE	×	×
● TELEMETRY (LM TO CSM)		×
● RECOVERY-VOICE/BEACON	×	
HF		
● RECOVERY-VOICE/BEACON	×	

Figure 1

NASA-S-66-5029 JUNE 1

LAUNCH VEHICLE FUNCTIONS

	S-IC	SII	SIVB	IU
S-BAND				
TELEMETRY				X
TRACKING				X
RANGING				X
UPDATE				X
VHF				
TELEMETRY	X	X	X	X
RANGE SAFETY	X	X	X	
C-BAND				
TRACKING				X

Figure 2

NASA-S-66-5999 JUL 5

MSFN CAPABILITIES

		S-BAND							VHF		HF	C BAND
CALL SIGN	STATION	VOICE	SC, SIVB & ALSEP TLM	TRACKING & RANGING	UP DATA	TV	ANT. DISH (FT)		VOICE	SIVB TLM	VOICE/ BEACON	SIVB TRACKING
							30	85				
MLA/CNV	CAPE KENNEDY	X	X	X	X		X		X	X		X
BDA	BERMUDA	X	X	X	X		X		X	X		X
GBM	GRAND BAHAMA	X	X	X	X		X		X	X		X
GTI	GRAND TURK								X	X		X
ANG	ANTIGUA	X	X	X	X		X		X	X		X
ACN	ASCENSION	X	X	X	X		X		X	X		X
CYI	GRAND CANARY	X	X	X	X		X		X	X		X
MAD	MADRID	X	X	X	X	X		X				
CRO	CARNARVON	X	X	X	X		X		X	X		
CNB	CANBERRA	X	X	X	X	X		X				
GWM	GUAM	X	X	X	X		X		X	X		
HAW	HAWAII	X	X	X	X		X		X	X		X
CAL	PT. ARGUELLO								X			X
GDS	GOLDSTONE	X	X	X	X	X		X				
GYM	GUAYMAS	X	X	X	X		X		X	X		
TEX	CORPUS CHRISTI	X	X	X	X		X		X	X		

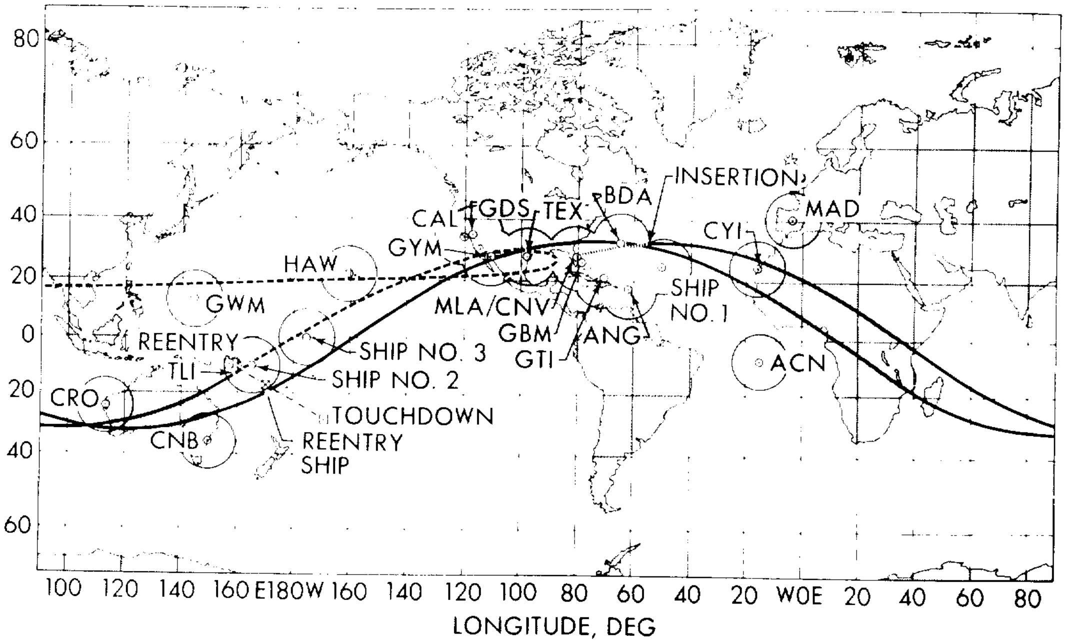
NASA-S-66-6631 JUL 6

MSFN CAPABILITIES(CONT)

		S-BAND							VHF		HF	C BAND
CALL SIGN	STATION	VOICE	S/C, SIVB & ALSEP TLM	TRACKING & RANGING	UP DATA	TV	ANT DISH (FT)		VOICE	SIVB TLM	VOICE/ BEACON	SIVB TRACKING
							30	85				
S-1	INSERTION-SHIP NO. 2	X	X	X	X		X		X		X	X
S-2	INJECTION-SHIP NO. 2	X	X	X			12'		X		X	X
S-3	INJECTION-SHIP NO. 3	X	X	X	X		X		X		X	X
S-4	REENTRY-SHIP NO. 4	X	X	X			2'		X		X	X
S-5	REENTRY-SHIP NO. 5	X	X	X			1		X		X	X
A/C	INSTRUMENTATION A/C (8)	X	X	X					X	X		
RECOVERY FORCES				X					X		X	

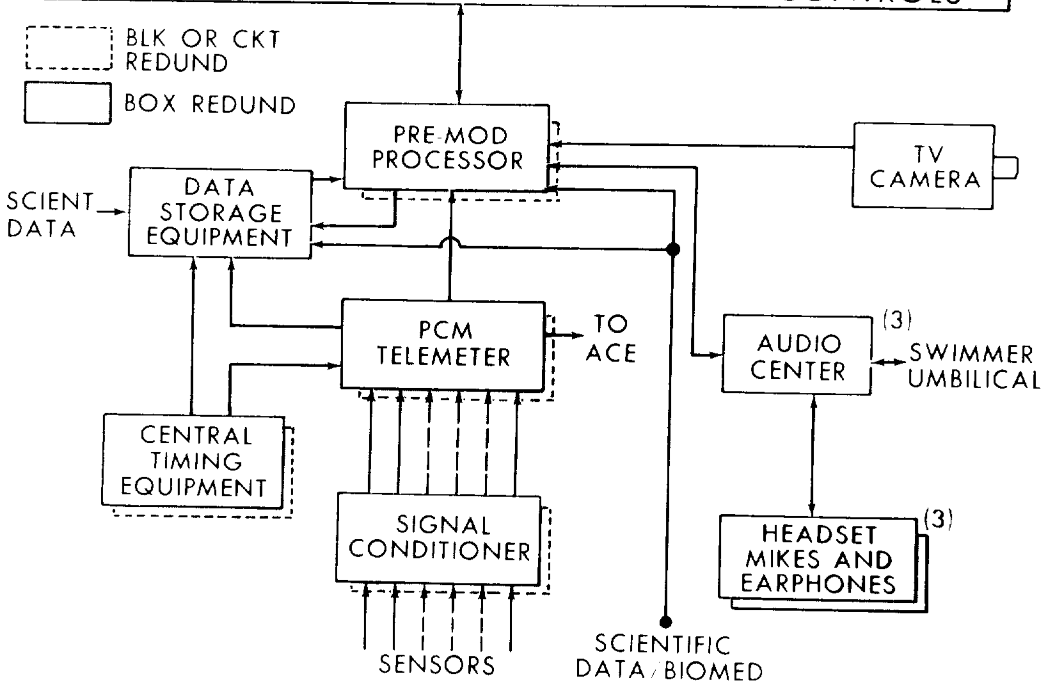
NASA S 66 5569 JUL 5

MANNED SPACE FLIGHT NETWORK COVERAGE


80
60
40
20
0
20
40
60
INSERTION
BDA
TEX
GDS
CAL
GYM
HAW
GWM
MLA/CNV
GBM
GTI
ANG
SHIP NO. 1
CYI
MAD
ACN
SHIP NO. 3
SHIP NO. 2
REENTRY
TLI
CRO
CNB
TOUCHDOWN
REENTRY SHIP
100 120 140 160 E180W 160 140 120 100 80 60 40 20 W0E 20 40 60 80
LONGITUDE, DEG

NASA S 66 6839 JUN

CSM TELECOMMUNICATIONS SUBSYSTEM CONTROLS


BLK OR CKT REDUND
BOX REDUND
PRE-MOD PROCESSOR
TV CAMERA
SCIENT DATA
DATA STORAGE EQUIPMENT
PCM TELEMETER
TO ACE
AUDIO CENTER
(3)
SWIMMER UMBILICAL
CENTRAL TIMING EQUIPMENT
SIGNAL CONDITIONER
HEADSET MIKES AND EARPHONES
(3)
SENSORS
SCIENTIFIC DATA/BIOMED

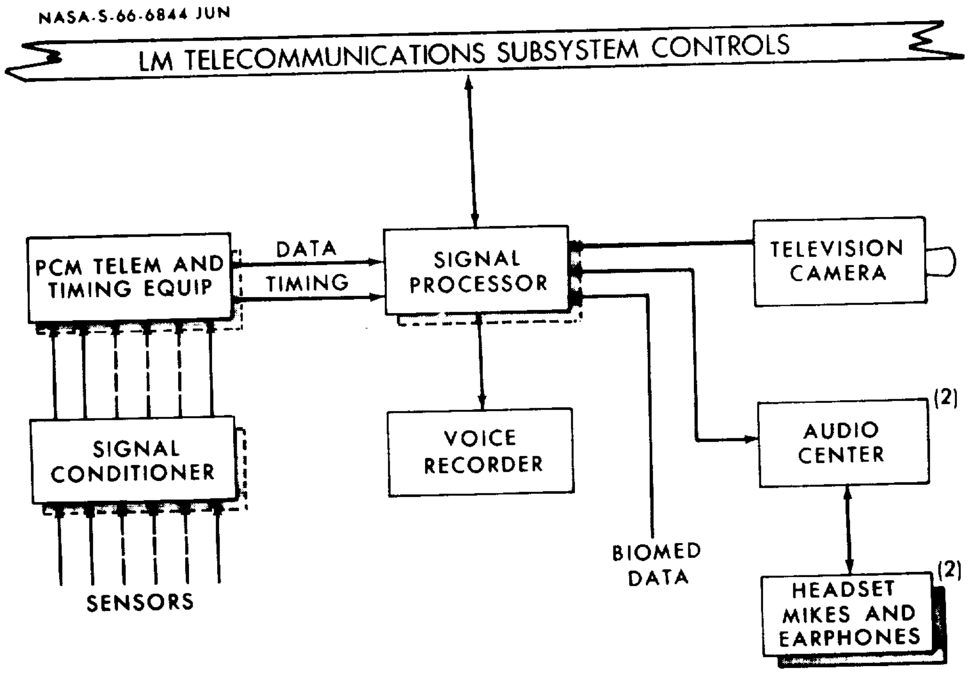

NASA-S-66-6844 JUN
LM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
PCM TELEM AND TIMING EQUIP
DATA
TIMING
SIGNAL PROCESSOR
TELEVISION CAMERA
SIGNAL CONDITIONER
VOICE RECORDER
AUDIO CENTER
(2)
BIOMED DATA
HEADSET MIKES AND EARPHONES
(2)
SENSORS

Figure 7

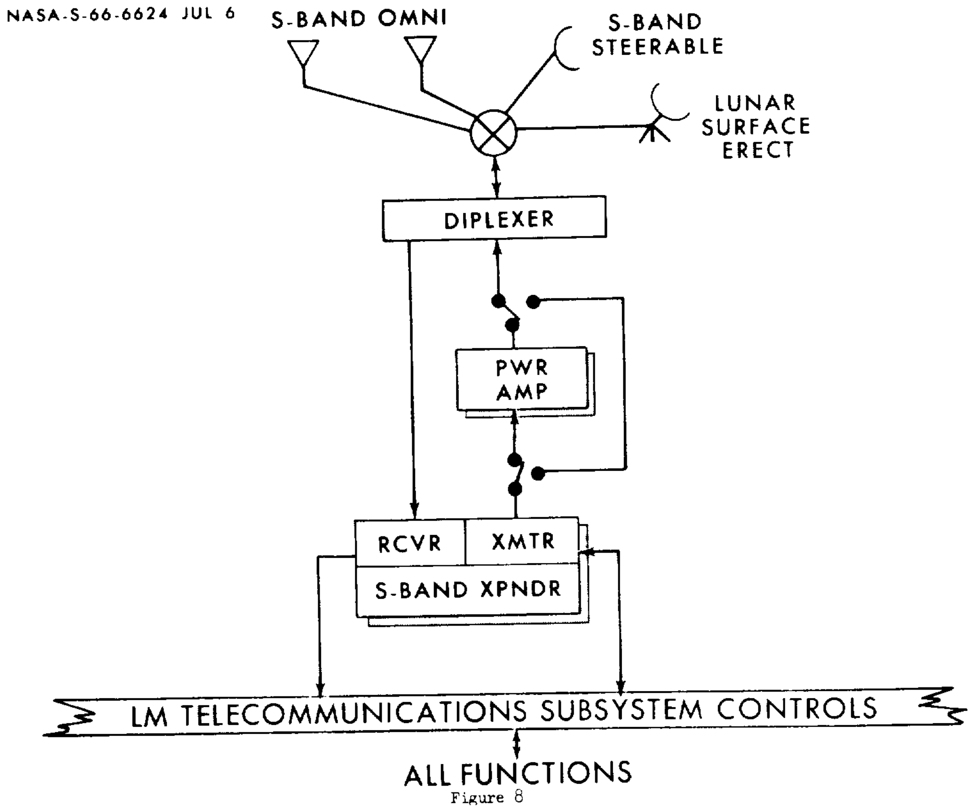

NASA-S-66-6624 JUL 6
S-BAND OMNI
S-BAND STEERABLE
LUNAR SURFACE ERECT
DIPLEXER
PWR AMP
RCVR
XMTR
S-BAND XPNDR
LM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
ALL FUNCTIONS

Figure 8

NASA-S-66-6632 JUL 6

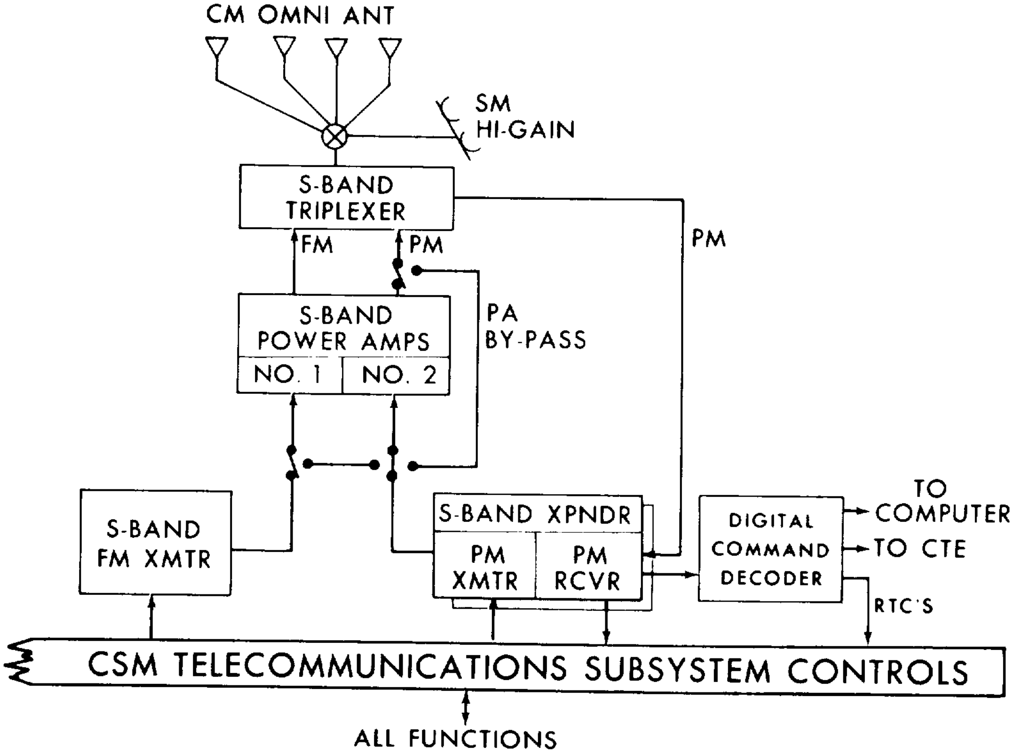

CM OMNI ANT
SM HI-GAIN
S-BAND TRIPLEXER
FM
PM
PM
S-BAND POWER AMPS
NO. 1
NO. 2
PA BY-PASS
S-BAND FM XMTR
S-BAND XPNDR
PM XMTR
PM RCVR
DIGITAL COMMAND DECODER
TO COMPUTER
TO CTE
RTC'S
CSM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
ALL FUNCTIONS

Figure 9

NASA-S-66-6840 JUN

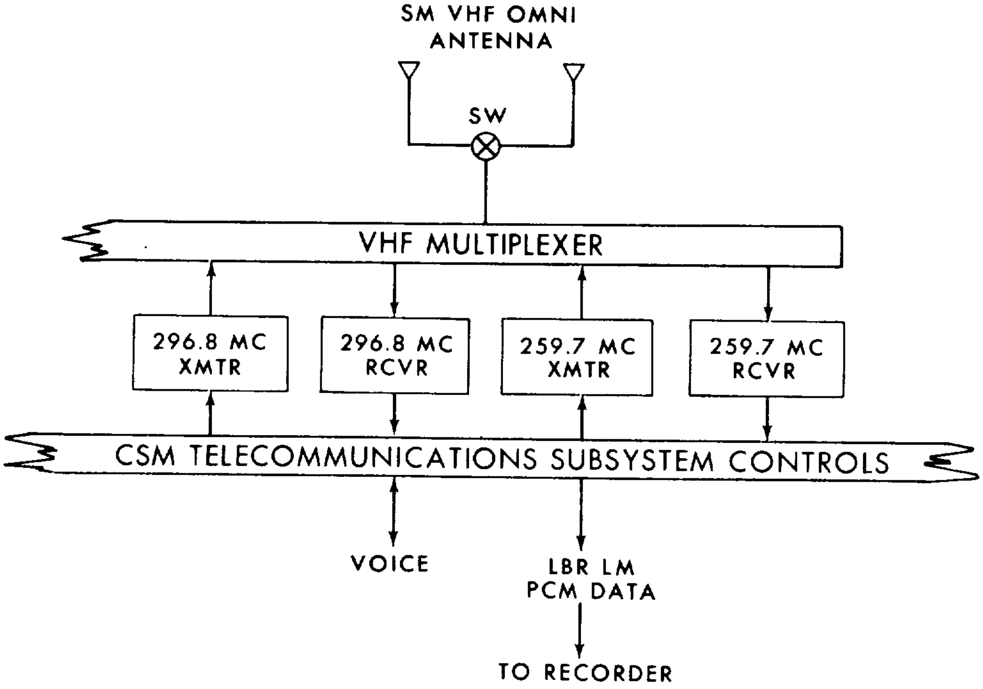

SM VHF OMNI ANTENNA
SW
VHF MULTIPLEXER
296.8 MC XMTR
296.8 MC RCVR
259.7 MC XMTR
259.7 MC RCVR
CSM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
VOICE
LBR LM PCM DATA
TO RECORDER

Figure 10

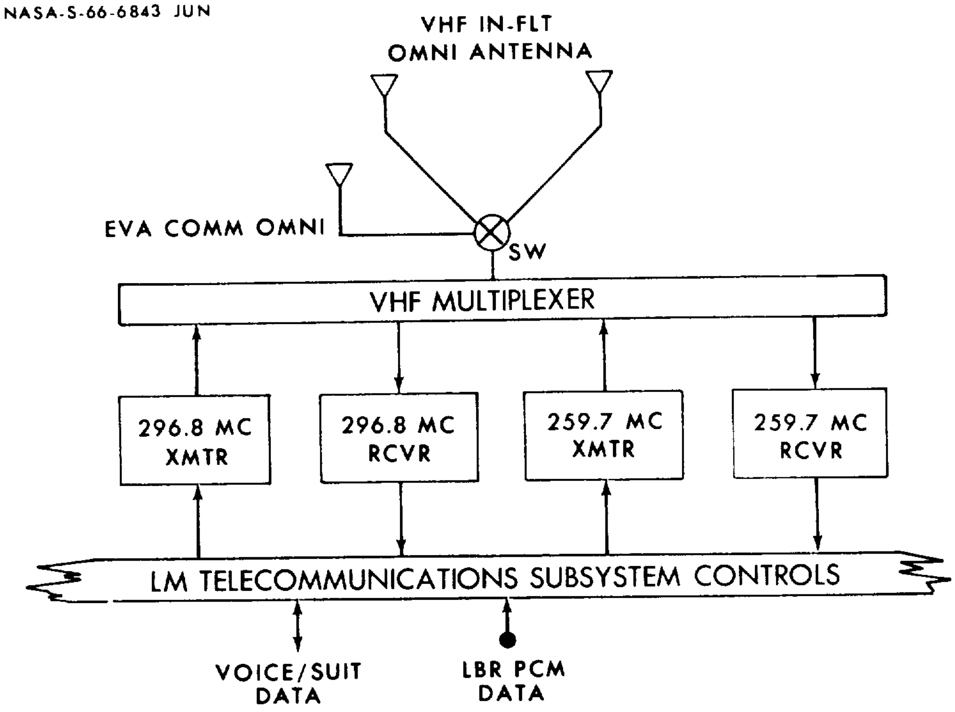

NASA-S-66-6843 JUN
VHF IN-FLT
OMNI ANTENNA
EVA COMM OMNI
SW
VHF MULTIPLEXER
296.8 MC
XMTR
296.8 MC
RCVR
259.7 MC
XMTR
259.7 MC
RCVR
LM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
VOICE/SUIT
DATA
LBR PCM
DATA

Figure 11

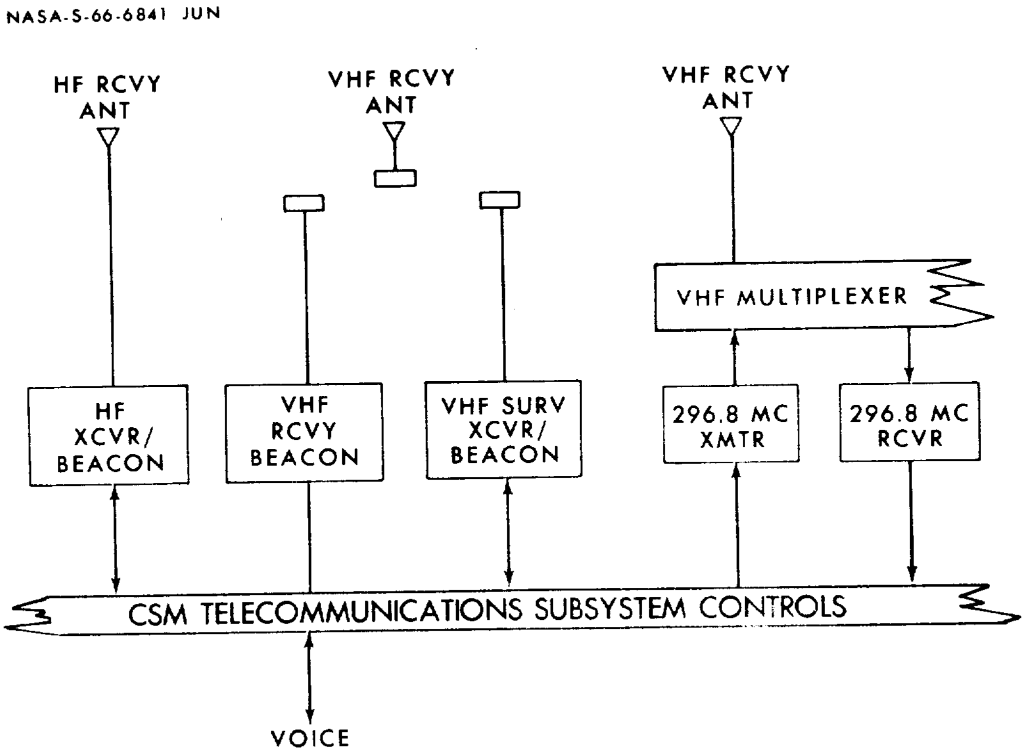

NASA-S-66-6841 JUN
HF RCVY
ANT
VHF RCVY
ANT
VHF RCVY
ANT
VHF MULTIPLEXER
HF
XCVR/
BEACON
VHF
RCVY
BEACON
VHF SURV
XCVR/
BEACON
296.8 MC
XMTR
296.8 MC
RCVR
CSM TELECOMMUNICATIONS SUBSYSTEM CONTROLS
VOICE

Figure 12

NASA-S-66-6036 MAY

S-BAND–CSM

- FREQUENCIES:
  - UP LINK–2106.4 MC
  - DOWNLINK–2287.5 MC

  COHERENT PAIR
  - DOWNLINK–2272.5 MC
- ANTENNAS:
  - 4 OMNI
  - 1 HIGH GAIN (WITH RF TRACKING)
- POWER:
  - LOW ¼ WATT
  - MED 5 WATTS
  - HIGH 20 WATTS

POWER LEVEL SELECTED AS A FUNCTION
OF RANGE AND DATA RATE

Figure 13

NASA-S-66-6027 MAY

S-BAND-LM

- FREQUENCIES:
  - UP LINK 2101.8 MC
  - DOWNLINK 2282.5 MC

  COHERENT PAIR
- ANTENNAS:
  - 2 OMNI
  - 1 HIGH GAIN (WITH RF TRACKING)
  - 1 HIGH GAIN ERECTABLE (SURFACE)
- POWER:
  - LOW ¾ WATT
  - HIGH 20 WATTS

POWER LEVEL SELECTED AS A FUNCTION
OF DATA RATE. ERECTABLE ANTENNA
USED FOR POWER CONSERVATION.

Figure 14

NASA-S-66-6038 MAY

VHF-CSM

- FREQUENCY:
  - 296.8 MC
  - 259.7 MC

  DUPLEX
  - 243.0 MC – RECOVERY (VOICE/BEACON)
- POWER:
  - 5 WATTS
- ANTENNAS:
  - 2 FLIGHT (ON SM)
  - 2 RECOVERY (ON CM)

Figure 15

NASA-S-66-6037 MAY

VHF-LM

- FREQUENCY:
  - 296.8 MC
  - 259.7 MC

  DUPLEX
- POWER:
  - 5 WATTS
- ANTENNAS:
  - 2 OMNI-FLIGHT
  - 1 OMNI-EVA

Figure 16

NASA-S-66-5060 JUNE 1

HF - CSM

- FREQUENCY 10.006 MC RECOVERY (VOICE/BEACON)
- POWER 20 WATTS IN SSB
  5 WATTS IN DSB
- ANTENNA 1 OMNI

Figure 17

NASA S 66 5028 JUNE 1

VHF - LAUNCH VEHICLE

- FREQUENCY

231.9 MC	S-IC
252.4 MC	
240.2 MC	
235.0 MC	
256.2 MC	
244.3 MC	
241.5 MC	S-II
234.0 MC	
299.9 MC	
248.6 MC	
236.2 MC	
253.8 MC	S-IVB
258.5 MC	
250.7 MC	IU
245.3 MC	

- POWER
  18 WATTS
- ANTENNA
  OMNI

Figure 18

NASA-S-66-5027 JUNE 1

LAUNCH VEHICLE (CONT)

- C-BAND TRACKING

FREQUENCY	5765 MC	RADAR
	5000 MC	AZUSA

- S-BAND (UP DATA, RANGING & TRACKING)

FREQUENCY	2282.5 MC	TRANSMIT
	2101.8 MC	RECEIVE
POWER	20 WATTS	
ANTENNA	OMNI	

- S-BAND TELEMETRY

FREQUENCY	2277.5 MC
POWER	20 WATTS
ANTENNAS	OMNI

Figure 19

NASA-S-66-5062 JUNE 1

LUNAR SURFACE EXPERIMENTS PACKAGE

● FREQUENCY	UPLINK 2119 MC DOWNLINK 2275.5, 2276.5 & 2278.5
● POWER	1 WATT
● ANTENNA	HELICAL
● DESIGN LIFE	1 YEAR
● DESIGN USAGE	100%

- ONE PCM DOWNLINK CHANNEL WITH ALL EXPERIMENTS MULTIPLEXED
- TRANSMITS AT NORMAL BIT RATE OF 1060 BPS TO A 30 FOOT ANTENNA
- UPLINK COMMAND CAPABILITY
  - SYSTEM ON-OFF
  - ACTIVATE EXPERIMENTS
  - CHANGE EXPERIMENT MODES
  - (COMMAND LIST TO BE DETERMINED BY EXPERIMENT REQUIREMENTS)

Figure 20

NASA-S-66-6628 JUL 6

VOICE - PRIMARY

CSM	LM
CSM-EARTH	LM-EARTH
S-BAND DUPLEX 243.0 MC RCVY 10.006 MC RCVY 296.8 MC RCVY S-BAND TAPE DUMP	S-BAND DUPLEX
CSM-LM	LM-CSM
296.8 MC SIMPLEX	296.8 MC SIMPLEX
CSM-EVA	LM-EVA
296.8 MC TRANSMIT 259.7 MC RECEIVE	296.8 MC TRANSMIT 259.7 MC RECEIVE

Figure 21

NASA-S-66-5069 JUN

VOICE - BACKUP

CSM	LM
CSM-EARTH	LM-EARTH
REDUNDANT S-BAND XPNDRS AND PWR AMPS	REDUNDANT S-BAND XPNDRS AND PWR AMPS
296.8 MC NEAR-EARTH ONLY	
CSM-LM	LM-CSM
ALTERNATE VHF EQUIPMENTS	ALTERNATE VHF EQUIPMENT
CSM-EVA	LM-EVA
259.7 MC TRANSMIT 296.8 MC RECEIVE	259.7 MC TRANSMIT 296.8 MC RECEIVE

Figure 22

NASA-S-66-5068 JUNE 1

VOICE RELAY AND CONFERENCE

- RELAY
  - EVA → CSM → EARTH (ONE EVA OUT)
  - EVA → CSM → EARTH (TWO EVA'S OUT)
  - EVA → LM → EARTH (ONE EVA OUT)
  - EVA → LM → EARTH (TWO EVA'S OUT)
  - EVA → LM → EVA
  - LM → EARTH → CSM
  - CSM → EARTH → LM
  - EVA → LM → EARTH → CSM
- CONFERENCE
  - LM ↔ EARTH ↔ CSM
  - EVA ↔ LM ↔ EARTH ↔ CSM
  - EVA ↔ CSM ↔ EARTH

Figure [illegible]

NASA-S-66-5026 JUNE 1

TELEMETRY

SYS	L/V	CSM	LM	EVA
PRIMARY	LAUNCH VHF/PCM ORBITAL VHF/PCM POST INJECTION S-BAND/PCM	S-BAND CARRIER (REAL-TIME XMSN) S-BAND CARRIER (TAPE DUMP)	S-BAND CARRIER (REAL-TIME ONLY) VHF XMSN OF LBR TLM DATA TO CSM FOR RECORD AND PLAY B TO EARTH	259.7 MC CARRIER, RECEIVED IN LM OR CSM AND RELAYED TO EARTH VIA S-BAND OVER THE REAL-TIME VOICE SUBCARRIER
SECONDARY	LAUNCH NO BACK-UP ORBITAL S-BAND/PCM POST INJECTION S-BAND/COMMAND & COMMUNICATION SYSTEM	REDUNDANT S-BAND EQUIPMENTS REDUNDANT PCM BLOCK CIRCUITRY	AS FOR BLOCK II CSM (NO BACKUP FOR LM-TO-CSM TLM XMSN)	NONE

Figure [illegible]

NASA-S-66-5025 JUNE 1

TRACKING

SYS	L/V	CSM	LM
PRIMARY	LAUNCH C-BAND AZUSA SYSTEM ORBITAL C-BAND RADAR POST INJECTION SC S-BAND	LUNAR MISSIONS S-BAND XPNDR (2-WAY COHERENT DOPPLER) PLUS RANGING)	SAME AS FOR CSM
SECONDARY	LAUNCH C-BAND RADAR ORBITAL SC S-BAND POST INJECTION C-BAND RADAR	REDUNDANT S-BAND EQUIPMENTS	SAME AS FOR CSM

Figure 25

NASA-S-66-5024 JUNE

UP-DATA

SYS	L/V	CSM	LM
PRIMARY	S-BAND COMMAND AND COMMUNICATION SYSTEM ALL PHASES	S-BAND DIGITAL COMMAND SYSTEM • RTC'S • CTE UPDATE • COMPUTER UPDATE	NO UPDATA CAPABILITY
SECONDARY	BUILT IN REDUNDANCY	REDUNDANT S-BAND RCVR NO REDUNDANT DECODER (VOICE LINK IS REDUNDANT FOR COMPUTER UPDATE & RTC'S)	N/A

Figure 26

NASA-S-66-5023 JUNE 1

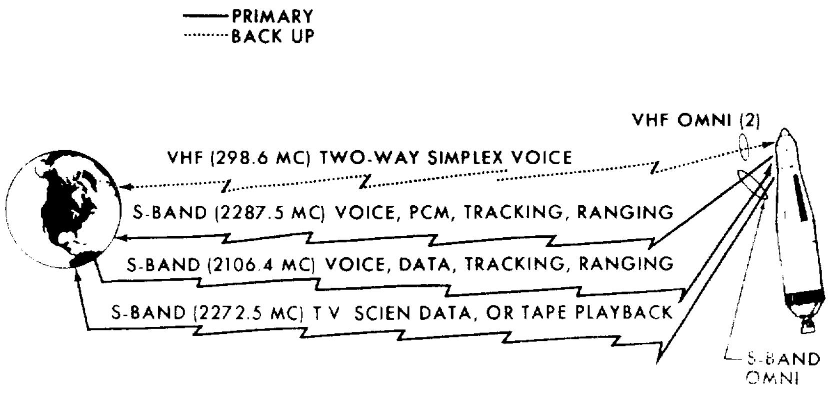
REAL TIME GROUND COMMAND FUNCTIONS TO CSM

- FLIGHT CREW ALARM SIGNAL
- PCM DATA RATE
- S-BAND POWER
- S-BAND RANGING
- S-BAND TAPE
- S-BAND PCM ON-OFF
- TAPE RECORDER
- RESET SWITCH
- ABORT REQUEST LIGHT
- SWITCH ANTENNA (HIGH GAIN TO OPPOSITE OMNI)

Figure 27

NASA-S-66-6001 MAY

ASCENT & EARTH ORBIT COMMUNICATIONS


——PRIMARY
··········BACK UP
VHF OMNI (2)
VHF (298.6 MC) TWO-WAY SIMPLEX VOICE
S-BAND (2287.5 MC) VOICE, PCM, TRACKING, RANGING
S-BAND (2106.4 MC) VOICE, DATA, TRACKING, RANGING
S-BAND (2272.5 MC) T V SCIEN DATA, OR TAPE PLAYBACK
S-BAND OMNI

TAPE PLAYBACK: RECORDED CSM VOICE, PCM,&SCIEN DATA

Figure 28

NASA-S-66-6629 JUL 6

TRANSLUNAR AND TRANSEARTH COAST COMMUNICATIONS

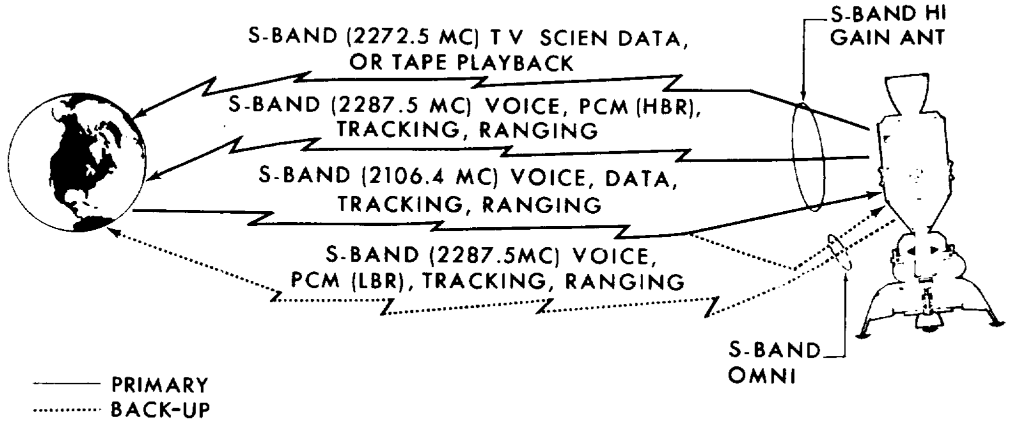

S-BAND (2272.5 MC) T V SCIEN DATA, OR TAPE PLAYBACK
S-BAND (2287.5 MC) VOICE, PCM (HBR), TRACKING, RANGING
S-BAND (2106.4 MC) VOICE, DATA, TRACKING, RANGING
S-BAND (2287.5MC) VOICE, PCM (LBR), TRACKING, RANGING
S-BAND HI GAIN ANT
S-BAND OMNI
PRIMARY
BACK-UP

Figure 29

NASA-S-66-6018 MAY

CSM-LM EARTH COMMUNICATIONS LINKS

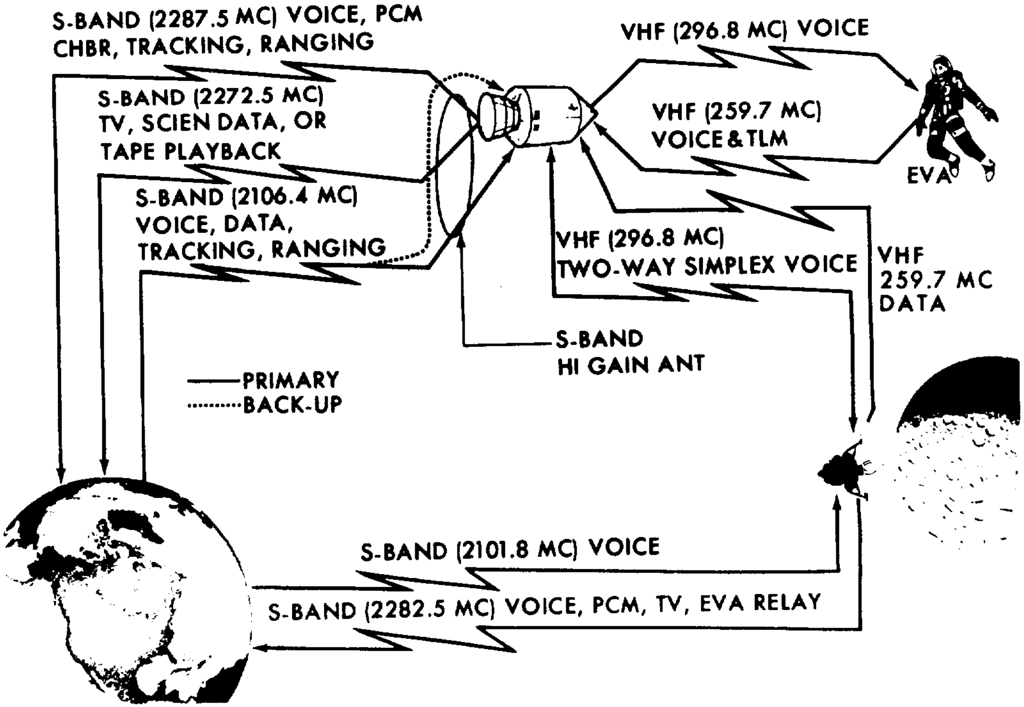

S-BAND (2287.5 MC) VOICE, PCM CHBR, TRACKING, RANGING
S-BAND (2272.5 MC) TV, SCIEN DATA, OR TAPE PLAYBACK
S-BAND (2106.4 MC) VOICE, DATA, TRACKING, RANGING
VHF (296.8 MC) VOICE
VHF (259.7 MC) VOICE&TLM
EVA
VHF (296.8 MC) TWO-WAY SIMPLEX VOICE
VHF 259.7 MC DATA
S-BAND HI GAIN ANT
PRIMARY
BACK-UP
S-BAND (2101.8 MC) VOICE
S-BAND (2282.5 MC) VOICE, PCM, TV, EVA RELAY

Figure 30

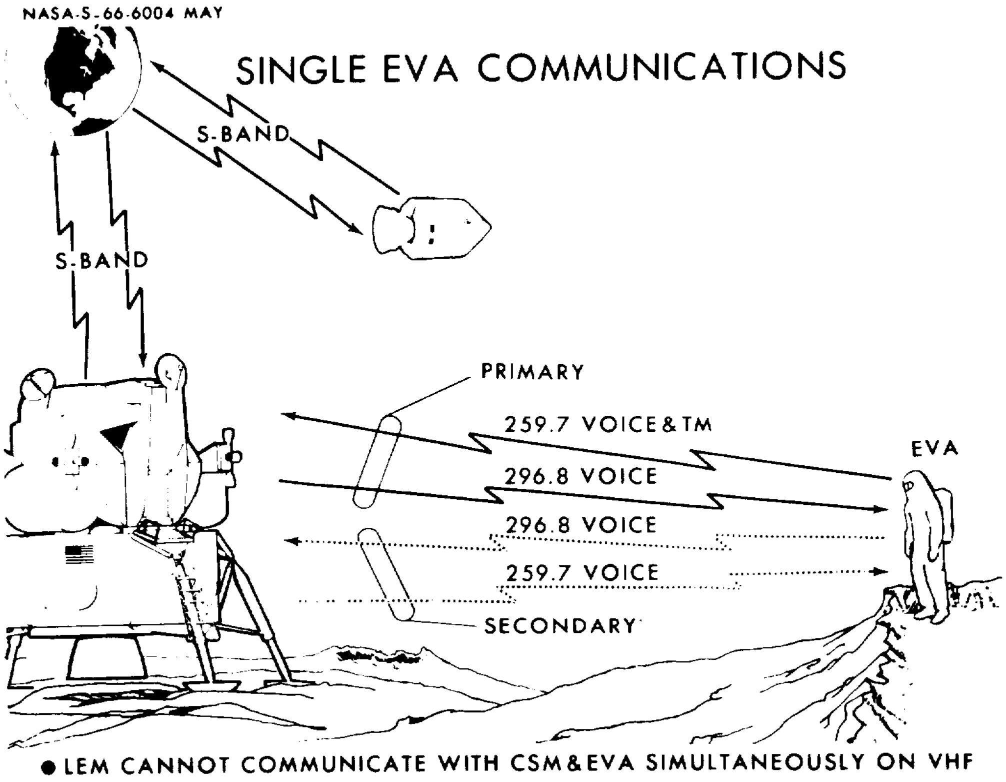

NASA-S-66-6004 MAY
SINGLE EVA COMMUNICATIONS
S-BAND
S-BAND
PRIMARY
259.7 VOICE & TM
EVA
296.8 VOICE
296.8 VOICE
259.7 VOICE
SECONDARY
● LEM CANNOT COMMUNICATE WITH CSM & EVA SIMULTANEOUSLY ON VHF

Figure 31

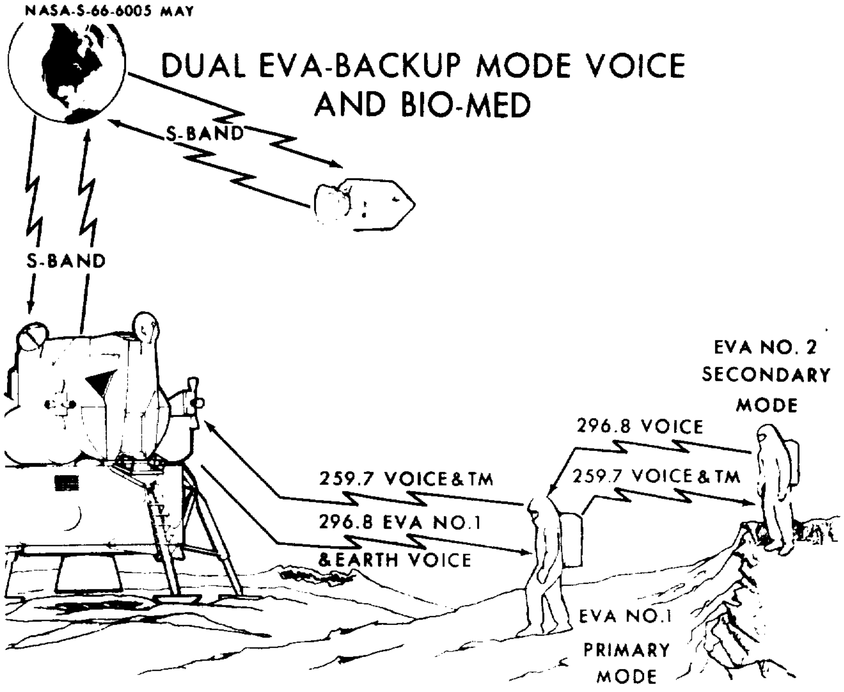

NASA-S-66-6005 MAY
DUAL EVA-BACKUP MODE VOICE AND BIO-MED
S-BAND
S-BAND
EVA NO. 2
SECONDARY
MODE
296.8 VOICE
259.7 VOICE & TM
259.7 VOICE & TM
296.8 EVA NO.1
& EARTH VOICE
EVA NO.1
PRIMARY
MODE

Figure [illegible]

NASA-S-66-5067 JUNE 1

ASCENT AND EARTH ORBIT COMMUNICATIONS

	LAUNCH TO INJEC-TION	EARTH PARKING ORBIT	TRANS-LUNAR INJEC-TION
S-BAND - VOICE	100%	26%	100%
S-BAND - HIGH BIT RATE TELEMETRY	100%	26%	100%
S-BAND - LOW BIT RATE TELEMETRY	N/A	74%	N/A
S-BAND - RANGING	100%	1%	100%
S-BAND - TRACKING	100%	26%	100%
S-BAND - UP DATA	100%	26%	100%
S-BAND - VOICE/DATA PLAYBACK	0%	20%	0%
VHF - VOICE	100%	26%	100%

Figure 33

NASA-S-66-6621 JUL 6

TRANSLUNAR COAST COMMUNICATIONS

	TRANS-LUNAR COAST	MID COURSE CORREC-TION	LUNAR ORBIT INSERTION*
S-BAND - VOICE	100%	100%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	100%	100%
S-BAND - LOW BIT RATE TELEMETRY	90%	N/A	N/A
S-BAND - RANGING	1%	100%	100%
S-BAND - TRACKING	100%	100%	100%
S-BAND - UP DATA	100%	100%	100%
S-BAND - VOICE/DATA PLAYBACK	AS REQ'D	AS REQ'D	AS REQ'D
S-BAND - TELEVISION	2%	N/A	N/A

*WHILE IN LINE OF SIGHT OF MSFN

Figure 34

NASA-S-66-6627 JUL 6

LUNAR PHASE COMMUNICATIONS

	DUAL LUNAR ORBIT COAST	SOLO COAST		DESCENT		LUNAR STAY (LM)
		CSM	LM	CSM	LM	
S-BAND - VOICE	60%	60%	60%	60%	60%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	10%	10%	60%	60%	5%
S-BAND - LOW BIT RATE TELEMETRY	50%	50%	50%	0%	0%	95%
S-BAND - TRACKING	60%	60%	5%	60%	60%	100%
S-BAND - RANGING	1%	1%	1%	1%	1%	1%
S-BAND - UP DATA	100%	100%	N/A	100%	N/A	N/A
S-BAND - VOICE/DATA PLAYBACK	10%	10%	N/A	10%	N/A	N/A
S-BAND - TELEVISION	2%	N/A	N/A	N/A	N/A	5%

Figure 35

NASA-S-66-6619 JUL 6

LUNAR PHASE COMMUNICATIONS

	DUAL LUNAR ORBIT COAST	SOLO COAST		DESCENT		LUNAR STAY (LM)
		CSM	LM	CSM	LM	
VHF - VOICE	-	10%	10%	100%	100%	10%
VHF - TELEMETRY (LM TO CSM)	-	-		N/A	40-60%	N/A
LUNAL SURFACE EXPERIMENT PACKAGE						*90%

*LSEP DESIGNED FOR CONTINUOUS OPERATION ON LUNAR SURFACE FOR ONE YEAR

Figure 36

NASA-S-66-5081 JUN 3

LUNAR PHASE COMMUNICATIONS

	ASCENT		RENDEZVOUS & DOCKING		TRANS-EARTH INJECTION
	CSM	LM	CSM	LM	
S-BAND - VOICE	60%	60%	0%	0%	100%
S-BAND - HIGH BIT RATE TELEMETRY	60%	60%	0%	0%	100%
S-BAND - LOW BIT RATE TELEMETRY	0%	0%	100%	100%	N/A
S-BAND - TRACKING	60%	60%	0%	0%	100%
S-BAND - RANGING	1%	1%	0%	0%	100%
S-BAND - UP DATA	0%	N/A	0%	N/A	100%
S-BAND - VOICE/DATA PLAYBACK	10%	N/A	10%	N/A	N/A
S-BAND - TELEVISION	N/A	N/A	N/A	N/A	N/A

Figure 37

NASA-S-66-6623 JUL 6

LUNAR PHASE COMMUNICATIONS

	ASCENT		RENDEZVOUS & DOCKING		TRANS-EARTH INJECTION
	CSM	LM	CSM	LM	
VHF - VOICE	100%	100%	100%	100%	N/A
VHF - TELEMETRY (LM TO CSM)	N/A	40-60%	N/A	100%	N/A
LUNAR SURFACE EXPERIMENT PACKAGE	100%	100%	100%	100%	100%

Figure 38

NASA-S-66-6622 JUL 6

TRANSEARTH COAST AND ENTRY COMMUNICATIONS

	TRANS-EARTH COAST	MID COURSE CORREC-TION	ENTRY TO BLACK OUT
S-BAND - VOICE	100%	100%	100%
S-BAND - HIGH BIT RATE TELEMETRY	10%	100%	100%
S-BAND - LOW BIT RATE TELEMETRY	90%	N/A	N/A
S-BAND - RANGING	1%	100%	100%
S-BAND - TRACKING	100%	100%	100%
S-BAND - UP DATA	100%	100%	100%
S-BAND - VOICE/DATA PLAYBACK	AS REQ'D	AS REQ'D	N/A
S-BAND - TELEVISION	2%	N/A	N/A

Figure 39

NASA-S-66-6009 MAY

RECOVERY COMMUNICATIONS

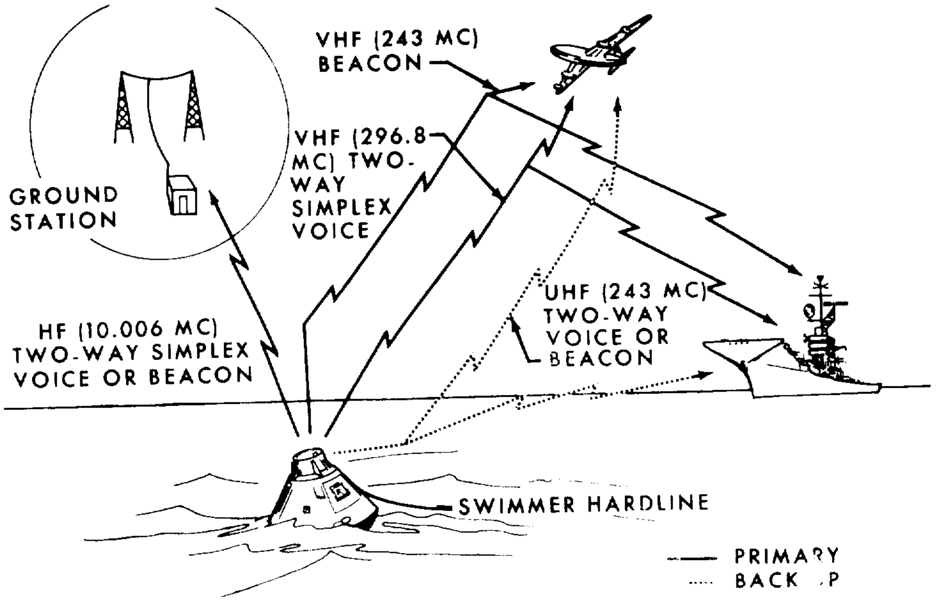

VHF (243 MC) BEACON
VHF (296.8 MC) TWO-WAY SIMPLEX VOICE
GROUND STATION
HF (10.006 MC) TWO-WAY SIMPLEX VOICE OR BEACON
UHF (243 MC) TWO-WAY VOICE OR BEACON
SWIMMER HARDLINE
PRIMARY
BACK UP

Figure 40

NASA-S-66-6620 JUL 6

POST LANDING COMMUNICATIONS

	POST LANDING
VHF-RECOVERY BEACON	40%
VHF - VOICE	3%
HF - VOICE	10%
HF - BEACON	3%

NOTE:

POSTLANDING SYSTEMS REQUIREMENTS ARE BASED ON A MAXIMUM OF 48 HOURS

Figure 41

NASA-S-66-6625 JUL 6

ELECTRONIC SYSTEMS TEST PROGRAM

SPACECRAFT-USB-MSFN
TEST SCHEDULE

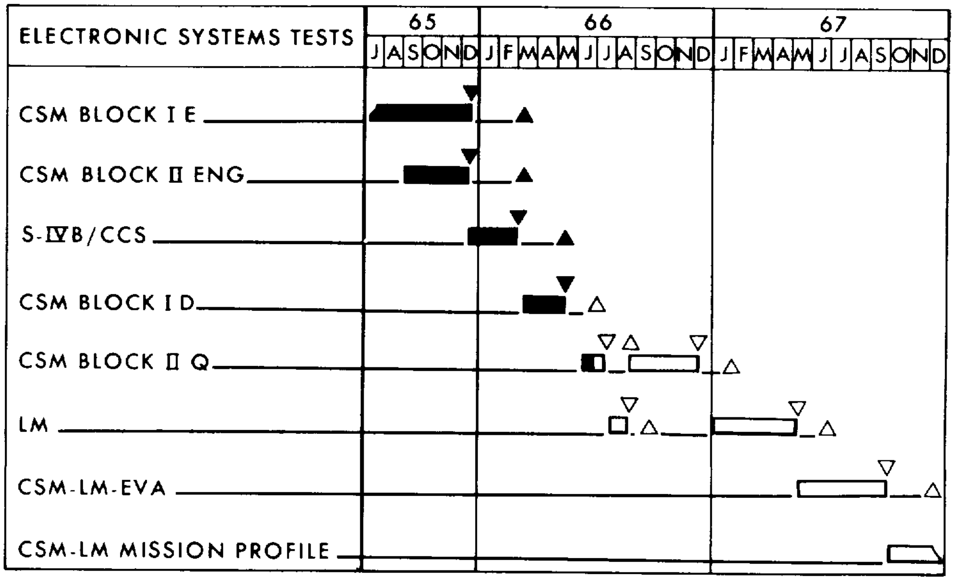


Figure 42

CREW TASKS AND TRAINING

by

J. P. Loftus
R. L. Bond

CREW TASKS AND TRAINING

The object of this paper shall be to provide an appreciation of the crew activities during the lunar landing mission. The emphasis is pictorial and qualitative rather than parametric and quantitative as in the papers presented to this point. We shall discuss examples of typical crew activity without emphasis upon detail because such detail does not lend itself to a presentation of this type and is often misleading when taken out of context. The training facilities will be discussed briefly not only in their customary role as training devices but as development tools for operating procedures and strategies. Crew activities shall be reviewed by mission phase.

NOTE: A motion picture film was used to furnish the nine sequences noted below and was an integral part of this paper. The film is available through the NASA-MSC film library and carries the NASA identification number S-66-411.

The sequences are: (1) Abort Studies, a brief sequence showing pilot reaction to system anomalies in a launch simulation; (2) Landmark Sighting, an animated view of a typical landmark sighting through the scanning telescope optics; (3) CM Active Docking, a pilot's-eye view of the lunar module during a command module active docking sequence; (4) Star Landmark, an animated view of what the navigator sees through the guidance optics during a star-landmark sighting; (5) Zero "G" Sequence, a short sequence filmed in the zero "G" aircraft showing transfer tunnel activity; (6) Lunar Landing Research Vehicle, a simulated lunar module descent trajectory flown by the LLRV at Edwards AFB; (7) Lunar Module Activity, a series of sequences showing typical lunar module cabin activity by the two crewmen subsequent to lunar module touchdown on the lunar surface; (8) Lunar Module Active Docking, a profile view showing the lunar module as the active vehicle in the docking sequence; and (9) Earth Approach, a film produced by inputting a computer with a moon-to-earth trajectory and letting the output drive an earth image on a cathode ray tube. One film frame represents four minutes of mission elapsed time. The resulting sketches indicate the apparent growth of the earth as seen by the homeward bound crew.

CREW REQUIREMENTS AND ORGANIZATION

The requirements placed upon the crew emphasize the role of the crew as a sensor, as a control element, and as a logic source. (Figure 1)

The crew participate as the abort initiation system in many parts of the launch. There are some failures requiring automatic abort in the early phase of the launch where failures can be abrupt and not suitable for crew action. A great number of the failures requiring abort can be detected more positively by the crew onboard the vehicle because of their capability, as a direct sensor, to confirm instrument indications by assessment of motion, vibration, and noise. The ground can observe some guidance failures which the crew might not perceive so readily and can request abort action.

In flight control functions the crew always acts in the sense of a programmer, selecting the suitable mode of the system for a given operation. The crew acts as a sensor in certain modes and as a control element in the manual and direct control modes.

The subsystem management role allows wide ranges of variation in the rate and manner of use of the consumables and in the programming of the systems. Crew action also provides a complex logic for use of the alternate modes and many forms of redundancy provided in each of the systems.

In navigation activities as in flight control functions the crew serves as a control element and as a sensing element.

The onboard mission management functions are primarily those where time is critical, where the crew by virtue of their presence in the vehicle have better data, and during those time periods in the mission during which they are out of communication with the ground complex.

The crew organization is an authority structure with duties being primarily assigned as a function of operating station rather than directly associated with an individual. (Figure 2) The crew is cross-trained and is trained as a team in the interest of enhancing reliability and to maximize the capacity of the crew to accept high work loads in certain portions of the mission.

TRAINING

The types of training are those with which we are familiar from Gemini and Mercury. (Figure 3) Systems training consists of briefings, textual materials, and actual participation in a number of spacecraft tests and engineering reviews. There are a number of specific task training elements related to particular manual skills, such as docking, extravehicular activities, and landing. Mission segment training is that type of training which is defined by the mission phase. Characteristic of this type of training are rendezvous, transposition docking, and entry. Specific mission training relates to those training activities determined by the trajectories and objectives of a particular mission. Illustrative of this class might be the lunar landmark briefings which would precede a particular mission to a particular site. Finally, there is experimental activity training which is both mission specific and general.

Training for normal modes of crew activity occupies perhaps the smallest portion of the total training time. (Figure 4) Training in malfunction recognition and response and emergency mode activity occupies the largest portion of crew time in training. The requirement for both prompt action and for high reliability establishes the requirement; the complexity of the system makes it time consuming. We have again emphasized the training of the crew as a team not only amongst themselves but in a number of exercises in conjunction with the flight controllers.

A summary listing of the major subsystems with which the crew training deals is presented in Figure 5. There is cross-summarization between the command module (CM), the launch vehicle, and the lunar module (LM) in this listing. The extravehicular mobility unit (EMU) is often overlooked, but there is significant training involved in the effective use of this complex.

Training facilities represent rather a large array of equipments. (Figure 6) This is due to the dimensions of the problem, to the limits of particular devices, and to the requirement to be able to concurrently train several crews.

The systems trainers are animated schematics wherein the spacecraft system is laid out on free-standing vertical panels. (Figure 7) The spacecraft controls and displays related to these subsystems are shown on the upper portion of the left panel. Located on the lower portion is a panel where malfunctions and anomalies can be created in the system so that the character of the malfunction and the symptoms associated can be studied.

There exists a family of devices known as part task trainers. We have mockups of both of the vehicles involved - the CM and the LM. (Figures 8 and 9) These devices are used in training in the many stowage and housekeeping functions, extravehicular activity, and other geometry defined activities. The dynamic crew procedures trainer, currently in the Gemini configuration, can accept a CM configuration. (Figure 10) This device is used to provide the dynamic environment associated with various launch conditions and entry conditions and it is primarily devoted to training in recognition of failures requiring abort. It provides sound, motion, and visual cues suitable for this environment since these are a significant part of the development of crew responses for such emergency conditions. This device can also accept a LM configuration for training in the dynamics of the LM motion in landing and in ascent. (Figure 11) The transposition and docking trainer is shown in the Gemini configuration in Figure 12, but it can be configured with the LM or the CM as the active element in the docking configuration. The lunar landing research vehicle (LLRV) shown in Figure 13 is illustrative of the lunar landing training vehicle which can provide a flight environment simulation for the terminal portions of the LM landing. We will discuss this device more at a later time. The egress trainer is used for training the crew in the post landing and recovery phases of the mission and integration of their activities with those of the recovery forces. (Figure 14)

The Apollo Mission Simulator (AMS) and the Lunar Module Mission Simulator (LMS) represent the major complexes in the training equipment array. (Figures 15 and 16) The AMS is controlled by the computer assembly shown in the background. There is a model house for the generation of the docking images, and an instructor's console where the device can be programmed and malfunctions can be inserted. The CM is nested among a large array of infinity image optical systems which are reflective optical transfer systems for providing images to the windows of the vehicle. There is also an assembly for providing the images to the guidance optics. The magnitude of this device can perhaps be appreciated when one considers that it has a 184 thousand word memory system with four central processors in parallel. The processors have $2\frac{1}{2}$ microsecond add times, effectively giving a nanosecond computer capability. There is a fully buffered channel capacity of one million words per second. The LMS is only slightly smaller. The computer complex contains three computers with a 98,000 word memory capacity. It has an image generation system for the CM docking portion of the mission and for the lunar touchdown portion of the mission. Film provides the approach image generation, again through the infinity image window systems. The window on the right side is shown in Figure 16 and the left window is shadowed in order that you can see the orientation of the crew within the vehicle. There is an image system for presentation to the overhead window which is used for observation of the horizon during powered descent and through which the docking sightings are made. These two devices can be operated in an integrated mode and can be integrated with the mission control center (MCC) to provide complete operational rehearsals of the mission. As a standard of comparison one may note that the BMEWS operational program required approximately 100 thousand instructions.

In addition to these devices, which are primarily oriented to the training and operational rehearsals, we have certain engineering devices which are used for special functions and tests as well as for certain specific types of training. The centrifuge shown in Figure 17 accepts a fixture which has three couches and all of the controls and displays suitable for launch, entry, and other high acceleration portions of the mission. There is an Air Force KC-135 which is used to simulate zero g and 1/6 g periods, again primarily for development testing but also for certain selected types of training primarily associated with extravehicular activities and with certain types of docking activity. (Figure 18)

Representative of another large class of engineering simulation devices is the lunar landing research facility (Figure 19) which is primarily a development device but which can be used for training in certain selected portions of the mission.

In addition to the devices shown we have a large number of engineering simulation facilities at the various contractors - at North American Aviation, Inc. (NAA), Grumman Aircraft Engineering Corporation (GAEC), and Massachusetts Institute of Technology (MIT).

CREW ACTIVITIES PROFILE

The mission for which these devices are used to prepare the crew involves a number of significant phases. In Figure 20 a typical profile of the mission shows two of the major constraints which govern the flight plan. The control events, shown as black diamonds are the launch, translunar injection, transposition docking, and midcourse corrections. The dark bars show the sleep periods which are allocated to the crew. The initial sleep period is obviously dictated by the time of the crew's waking and preparation for the launch. The schedule of activities must be arranged so that at lunar orbit insertion and the lunar descent portion of the mission the crew is suitably refreshed to enter a period of rather high density activity. The profile shown here reflects a constraint that sleep periods shall be at least six hours and that such periods shall occur at intervals not greater than 18 hours. It also presumes that three crewmen sleep at the same time. This takes advantage of the capacity of the ground to view the vehicle at all times and to wake the crew in the event that some anomaly is observed.

In the lunar portion of the mission an exploration period is scheduled immediately after the landing. The logic which dictates this schedule is that the portable life support systems (PLSS) are at this time fully charged and the vehicle is pressurized. If we perform an exploration at this time we can have an exterior check of the vehicle shortly after landing. The crew can donn the PLSS and the thermal meteoroid garments (TMG) in a pressurized cabin, depressurize the cabin, perform the exterior activities, return, pressurize the cabin, and then doff this gear and re-charge the PLSS in a pressurized configuration. If we had chosen to sleep first, an alternative which we can choose in real time, back-to-back exploration periods would require re-charging of the PLSS in an evacuated ascent stage. There are procedures for doing so, but in the early missions it seemed desirable to have the additional benefit of the pressurized cabin. This also allows a suitable period of rest prior to the activities of ascent into lunar orbit and transearth injection.

Figures 21 through 24 summarize the proportions of mission time - a total of 216 hours - and the proportions of the total crew man hours - 648 - which are devoted to various types of specific activity. The control tasks, as one would expect, occupy a relatively small proportion of the total mission time though they are obviously the most critical events. The relative proportions devoted to various activities are obviously a function of the particular details of the mission, though one may expect certain proportional relationships to remain constant for all missions. Notice that in such activities as monitoring, which is done primarily by one man on a watch configuration, there is some nominal accumulation of potential crew time as opposed to mission time. This can be more apparent than real. A number of tasks require activity by all three crewmen while others are performed primarily by one man at a given time. Figure 23 illustrates mechanical manipulations - the changing of stations within the vehicle, the donning and doffing of the suit, the manipulation of the docking equipment. The exploration period, effectively the payload, represents approximately 2% of the total mission time and of the total crew man hours. A third of the mission time in crew man hours is devoted to sleep. In this particular summary 31% of the mission hours or 46% of the crew man hours are not scheduled.

The validity of such numbers is only as good as the analysis of the times required to perform particular functions, and is highly sensitive to the details of the particular flight plan. Experience to date indicates that most activities take substantially longer in flight than during simulations, a factor of two or more being quite common.

There is a further artifact pertinent to the form of this summary. It does not distinguish usable free time from blocks of time of no value. Time, in the sense of crew man hours, may appear to be available but constraints such as vehicle arrangement, subsystem configuration, or location of crewman may vitiate its utility. It is also important to schedule some "free" time in the same fashion as planned "hold periods" in a countdown.

AVAILABLE WORK AREA

Figure 25 illustrates the available volumes in each of the vehicles. The geometry of the CM is dictated by the entry requirement, the offset center of mass providing the effective L/D, and by the sweep volume required by the couch in land landings. The crew compartment volume is defined as the lightly shaded area and the effective free volume is that area not occupied by such objects as the couch, the PLSS, and the various other stowed equipments. The LM geometry is defined by the requirement to counter balance the loads around the ascent engine. Here also there is a reduction in the total pressurized volume by the various equipments. Comparable figures for the Gemini vehicle are 80 cubic feet of pressurized volume and 50 cubic feet of effective free volume. We expect that the larger volume and its favorable arrangement will enhance the effectiveness of a number of types of crew activity.

CONTROLS AND DISPLAYS

The CM main display console shown in Figure 26 is color coded to indicate the various proportions of functional activities allocated display space. The exact proportions are not shown since large numbers of the displays are time shared. It is significant, however, to note the relatively large areas devoted to the sustaining systems. Since this vehicle must accomplish the long time operations of the mission a large number of levels of redundancy and cross-switching are available within the environmental and the electrical power generation and distribution systems. The sequential events and staging associated with launch and entry are also a major function of the CM.

The LM panel (Figure 51) is predominantly given to propulsion and flight control. This reflects the character of its portion of the mission and the simpler configuration of its sustaining systems. The use of batteries for electrical power, and bottled gas rather than cryogenic stores for the life support system make the sustaining systems simpler though less flexible.

MISSION OUTLINE

At this point it would be well to review the configuration of the vehicle at each of the phases of the mission and to examine the crew functions which characterize that phase of the mission.

Launch

During launch the crew is primarily concerned with monitoring the characteristics of the launch vehicle and its flight performance, maintaining communication, and monitoring the condition of the spacecraft. (Figure 27) The vehicle at this time is arranged with a number of soft goods stowed on the floor, the lithium hydroxide (LiOH) cannisters used in the environmental control system (ECS) are installed along the lower edge and a large number of equipments are stowed on the upper wall and in a number of compartments on both sides of the vehicle. (Figure 28) The crew is not shown in the launch configuration because they would obstruct the view but they are beneath the array of harnesses, oxygen umbilicals, and communications lines. It is difficult to describe, in a sense that can be appreciated, the monitoring activity in launch. However, Figure 29 outlines a typical sequence of events for one selected launch condition in which an abort would be necessary subsequent to 61 seconds of powered flight.

Once an abort decision is taken, either by the crew or on ground request, the crew has the capability to monitor a large number of the events which take place automatically. The timer would reset so that the time sequence characteristic of the abort could be monitored. There is a capability to note the acceleration of the CM by launch escape motor both by accelerometer and directly. If this does not occur the event can be commanded by the crew. Those items shown with question marks in the figure are items checked to confirm the functioning of the automatic sequencer; verifying that the CM reaction control system (RCS) has pressurized confirms that the relays have actuated which cut the tension ties, deadface the CM and service module (SM), and arm the CM batteries to provide electrical power in that configuration and pressurize the CM RCS for attitude control. Eleven seconds after the initiation of the abort the canards would deploy. If they do not deploy the crew can command that event through an independent path. The crew can confirm that the earth landing system logic has been armed and provide an alternate path for the jettison of the launch escape tower. Drogue chute deployment can be confirmed at 24,000 feet or below by reference to the barometric altimeter, and in the event that function does not occur they can again provide an alternate command. Once on the main parachutes, no longer requiring the attitude control of the CM RCS, they would close the cabin pressure relief valves, command the necessary dumping and purging of the CM RCS, re-open the valves which have been closed to prevent injestion of the gases and, after landing, release the main chute.

The identifying characteristic of the abort after 61 seconds is that there is not an automatic dumping of the RCS propellants, since above the altitude attained by that point in launch one may require use of the attitude control system for orientation.

Earth Orbit

In the earth orbit phase of the mission the crew has control of the total space vehicle, including the S-IVB, through the interface between the spacecraft guidance computer and launch vehicle guidance computer. After earth orbit insertion the crew would realign the inertial measurement unit (IMU) using the guidance optics. Provided there were time, as would be the case with translunar injection on the second orbit, they can perform certain navigation activities in earth orbit. (Figure 30) These activities would be done to confirm the condition of the onboard system since the ground stations would provide the primary navigation. The technique used for orbital navigation is, however, also used in lunar orbit, with known orbital conditions, to assess the altitude of landing sites.

Prior to translunar injection the crew would fine align the IMU (Figure 31) and begin the countdown for the ignition of the second burn of the S-IVB. The crew can inhibit this command should their onboard data indicate that the vehicle is not ready for the mission. There is, however, only a single opportunity due to the requirements of the S-IVB propellants.

Translunar Injection

During the translunar injection burn, as during all other propulsive maneuvers, the crew will have prepared the vehicle for peak power loads by bringing both batteries and fuel cells on the line. They will monitor the condition of the various systems and in particular monitor the guidance performance. (Figure 32) Perhaps of some interest is that during this maneuver we see the first of a number of very low acceleration environments, approximately 1 1/3 g during this burn.

Transposition and Docking

Transposition and docking is the first of a number of activities which the crew directly and completely controls. Immediately after the conclusion of the translunar injection burn the crew initiates a roll and pitch maneuver with the total space vehicle to place the high gain antenna for the S-IVB in proper orientation to illuminate the Manned Space Flight Network stations on the earth. This maneuver also assures optimum lighting for the docking maneuver to follow. This activity takes approximately 5 minutes due to the low rates of maneuver used with this high mass configuration. It takes approximately 10 minutes for the ground stations to confirm the adequacy of the orbit to which the spacecraft has been injected and subsequent to this point in time the crew is free to proceed with the maneuvers separating the CM to a distance of approximately 100 feet. (Figure 33) Again they enter a roll and pitch maneuver to provide antenna orientation for the command and service module to the ground stations and begin the sequence of activities leading up to the docking. (Figure 34)

Figure 35 shows the fashion in which the couches are moved about pivot points and along a rack in order to provide an optimum view for the crew through the rendezvous and docking windows, which can be used by the crewmen on the left and on the right. The view available to the pilot performing the maneuver is illustrated in Figure 36. Figures 37 and 38 show the docking target that the pilot sights on to perform this precision maneuver.

The docking target is observed by the CM pilot through an optical device much like a gun sight. The white cross stands 14 inches above the red target. In its final configuration it will be T-shaped with a diamond in the center of the intersection of the bars. As long as that diamond remains within the white circle the docking is being performed within the capture range of the probe and drogue. The verticle stand-off gives the pilot some cue as to his errors out of the line of approach. The lines of the T are an index of the proper rotational indexing of the CM and the LM.

After having made the initial contact with the probe and drogue the probe mechanism provides for drawing the LM and the CM together and setting four latches at which time a soft docking is achieved. A sequence of activities is begun to pressurize the tunnel area against a leak rate. A pressure hatch and a thermal hatch are removed by the crew. Eight additional latches are hand-set by the crew to achieve a hard dock and structural integrity. (Figure 39) Redundant umbilicals are connected to the LM to provide for the electrical current required in the LM during the translunar leg of the mission and to provide a path for the pyrotechnic device actuations which will release the LM from the adapter. After this operation has been performed the docking mechanisms are restored in the tunnel and the crew activates the pyrotechnic device and withdraws the LM from the adapter by using the SM RCS in a minus-X translation mode. This is done using a series of short intermittent burns to reduce the amount of impingement of the SM reaction control engines upon the thermal coatings of the LM. The crew establish a separation rate of approximately three feet per second and then orient the vehicle for passive thermal control during the remainder of the mission.

Translunar Coast

Translunar coast is characterized primarily by the mid-course correction requirements determined by the precision of the initial guidance and by the necessity to monitor and maintain periodic checks on the systems. (Figure 40) Periodic maintenance of systems such as the fuel cells and the ECS and a number of the general housekeeping activities are necessary for living during the period while we go to the moon.

This seems a suitable point to discuss the housekeeping problem. Figure 41 shows the various stowage compartments in the three walls of the vehicle - the lower equipment bay, the right hand equipment bay, the left hand equipment bay, and the rear side of the main display console. There are a large number of items stowed in each of these compartments - the numbers being indicated in Figure 42. These items and operations in a small volume in which three men must live for a protracted period make a situation in which everything must have its place and be in that place. The numbers are reasonably impressive all by themselves, while obviously subject to a good deal of discussion in terms of what should be defined as an operation or what one should define as a unit. The numbers indicate that there is a significant problem in simply keeping track of the location and usage of each of the devices. This number of things and operations contributes to the requirement for a good deal of formality and care in various procedures. It also emphasizes how a small error in estimate of time required can adversely affect flight plans.

Figure 43 illustrates the configuration of the vehicle during those periods when the crew would sleep in the CM. Two of the crewmen sleep under the couches in sleeping bag arrangements which provide some measure of thermal control and which further allow the crew to be restrained in the zero g environment. The third man sleeps in his couch where he has direct access to the environmental and electrical control systems which sustain the vehicle. Should any anomaly develop the crew can be awakened in the CM by a direct updata link command. At this point the suits and helmets would be stowed in bags.

The arrangement for the preparation of food is illustrated in Figure 44. Food is stowed in the compartment facing the crewman in man/meal type containers. A water delivery probe located to the crewman's left provides the source for reconstitution of freeze-dried food. The system has the capability for providing three hot meals at the same time to the crew. There is a velcro covered work shelf provided so that equipment can be manipulated in the environment.

Figure 45 illustrates the arrangements of the mechanical components of the ECS which allow the crew to have access to the LiOH cannisters installed in the system. There are two cannisters in parallel. Each has a 24-hour life and they are cycled so that there is a cylinder change each 12 hours. In addition to providing the removal of carbon dioxide these cylinders also contain an amount of carbon to minimize the accumulation of unpleasant odors. That problem, however, is relieved primarily by venting such odors directly overboard.

Typical of the types of status checks which may be conducted during this portion of the mission is an ECS periodic review where the parameters characterizing the nominal performance of the vehicle can be examined. (Figure 46) A number of these are displayed continuously, such as the glycol steam pressure and the glycol discharge pressure. In the event that the crew are in suits it is possible to confirm the oxygen flow and pressure. Some of the displays are time shared. The radiator outlet temperature is an index of the adequacy of the passive thermal control maneuver. Glycol temperature indicates the condition of the electronics cooling. The partial pressure of carbon dioxide is an indication of the safety of the atmosphere. Any event which is critical to the safety of the crew is in the logic of the caution and warning system where an array of annunciations can direct the crew's attention should something occur that is not immediately perceived in general monitoring of the systems.

Navigation activities can also be performed on the translunar and transearth portions of the mission. (Figure 47) The geometry of this activity has been described in an earlier paper.

In the early missions we contemplate checking out the LM prior to lunar orbit entry in order to have the benefit of its systems for certain abort contingencies and to confirm the condition of the vehicle prior to the commitment to lunar orbit insertion. (Figure 48) This has some additional advantage in that it allows the complete sequence of activity to be observed by the ground. In order to perform this activity the crew must enter the LM and since it has probably depressurized since launch the procedure is to pressurize the tunnel area and the LM from the CM and confirm that the pressure is equivalent across the tunnel and in the vehicles. The crew then removes the pressure hatch, the ablative hatch, the probe, and the drogue, opens the LM hatch, and enters the LM. (Figures 49 and 50) Figure 49 shows the configuration of the CM during the stowage of the pressure and ablative hatches, the drogue and the probe. One crewman's foot can be seen as he is beginning the transfer. It would be expected that two of the crewmen would enter the LM to conduct the checkout.

The orientation of the crew within the vehicle can be appreciated by noting Figure 52. One of the devices noteworthy in this configuration is the sequence camera in the right LM window mounted in a fixed bracket, parallel to the crew's line of sight so that it can observe the lunar landing and record it. The TV camera is stowed at the lower right front. There is stowage of much of the needed equipment in the bags below the right and left side-panels.

Characteristic of the checkout sequence to be conducted is the procedure shown in Figure 53. The regulator status would be verified by confirming the position of the talk-backs associated with each switch and examination of the pressurization of the system by checking the descent pressurization indicator, helium pressure, and the other system status points shown. This would be the series of activities for the descent propulsion system up to the point where the crew would begin that sequence associated with the throttle manipulation.

Lunar Orbit Insertion

Lunar orbit insertion is again a guidance maneuver which has been discussed in significant detail in a preceeding paper. The activities are very similar on the part of the crew for this maneuver as for the translunar injection and for earth orbit insertion. (Figure 54) There is an aspect of the geometry which has not been commented upon. Figure 55 shows the growth in the apparent size of the moon as a function of mission time. The schematic in Figure 56 shows the effect of the lighting conditions. At the 75:36:36 point in time the moon occupies 23 degrees, an hour later it occupies 63 degrees, and 15 minutes later it has grown to approximately 130 degrees with a 134 degree apparent size in lunar orbit. The significant thing is that we will be approaching the dark of the moon and the earth shine is from a half earth. The crew can, however, perceive that this is not a collision course by observing the apparent regression of the limb of the moon against the line of sight.

After entry into lunar orbit, the two crewmen would again transfer into the LM and transfer a significant amount of equipment. (Figures 57 and 58) The PLSS has been carried in the CM because it provides a capability for emergency extravehicular activity. The emergency oxygen supply (EOS) provides a five minute capability in a high pressure bottle, and is used to provide an emergency backup to the PLSS. These have been stowed in the CM in order to have them in a more favorable thermal environment. The extravehicular gloves and the TMG have been kept in the CM to provide for an extravehicular capability if required. There is but a single radiation survey meter and it would now go to the LM where the inherent shielding by the vehicle is less effective. There is only one TV camera. The EVCT is the extravehicular crew transfer device. The umbilicals would remain in the CM.

Lunar Descent

The series of maneuvers associated with the LM descent to the lunar surface has been discussed in considerable detail in preceeding papers. Figure 59 is a very brief review [illegible] the activities assigned to the crew in conjunction with [illegible] primary guidance system and the abort guidance system. In addition to these there are a number of pilotage activities which can be performed by the crew as opportunity allows. Through the overhead window, which is above the left crewman, it is possible to have a view of the lunar horizon during the descent phase of the mission, which may provide some useful indications of attitude and altitude. The primary function of the crew during the braking phase is to monitor the automatic systems and to provide suitable initiation of various equipments. They would confirm the status of the RCS and the ascent propulsion system prior to final approach in order to confirm the existence of their abort capability. It is interesting to note that during this portion of the mission the crew will experience approximately one third of a g with the vector along their body axis as in standing. (Figure 60)

Figure 61 shows the LLRV which has now flown approximately 140 flights. To simulate a LM descent trajectory it takes off in a VTOL mode using the jet engine as its thrust source with attitude controlled by a hydrogen peroxide RCS. The pilot climbs to approximately 600 feet and establishes the initial conditions for the entry to the simulated portion of the LM descent trajectory. He establishes these conditions at approximately 400 feet altitude, having 45 feet per second forward velocity and approximately 9 feet per second descent velocity at that point in time. There is a departure from the fidelity of the simulation in that the line of sight of the seated crewman in this vehicle cannot be depressed so far as in the LM and he is seated rather than standing. The seating requirement derives from the use of an ejection seat for safety. The pilot has cross-range and down-range indicators, the flight director, and thrust-to-weight indicators. There is a three axis hand controller for attitude control. The geometry of viewing to the instruments and to the available window is correct. When the pilot has established his initial conditions, he transfers to a lunar simulation mode in which the engine is gimballed and provides 5/6 of a g thrust. The simulated descent engine, also a hydrogen peroxide engine, is ignited and from this point on the attitudes are characteristic of the LM descent and its piloting characteristics can be evaluated. The control authority is that of the LM. The attitude of the vehicle is approximately 12 to 15 degrees pitch up to reduce the horizontal velocities. He enters a hover at approximately [illegible] and begins a controlled rate of descent to the touchdown point, reducing the rate to approximately 3½ feet until he is very near the surface where a major departure in the simulation occurs. This vehicle is flown in a thrust to touchdown mode to minimize landing loads. The weight limitations do not allow adequate attenuation for the routine practice of thrust-off landings.

Lunar Stay

The graph in Figure 62 is in the format used earlier to show the activities characteristic of the lunar stay time. Immediately after landing the vehicle is checked to assure that conditions are suitable for remaining on the surface. The descent tanks are vented in order to preclude problems due to thermal buildups and the crew immediately aligns the IMU and places it in a standby mode to have an any time departure capability. Postlanding checkout is scheduled for 33 minutes and this time estimate is based upon rehearsals conducted in a LM mockup.

This particular plan shows the immediate donning of the extravehicular equipments and an initial exploration activity. Representative tasks are inspection of the vehicle for any apparent exterior damage or leaking, confirmation of the conditions of the landing such as depressions in the lunar surface, slide marks, and measurement of gear stroke distance. (Figure 63) Other early extravehicular activity would be the deployment of the necessary equipments, such as the antennas, and an initial survey from the LM platform to do TV and film scans of the area for later analysis.

The next series of Figures shows the configuration of the vehicle for various activities. In Figure 64 the crewman is donning the PLSS. It shows one crewman in the TMG with the PLSS and the EOS attached. The second PLSS is shown in the recharge station. The EVCT is shown above the wall mounted PLSS. Food and other equipments are stowed immediately below the EVCT.

As noted, one of the first extravehicular activities would be a camera survey of the area and a postlanding inspection. Figure 65 shows the net and cable device used to transfer equipment to the surface from the ascent stage.

Figure 66 illustrates the sleeping arrangements within the vehicle. One of the crewmen suspends a hammock arrangement from the front edge of the ascent engine dome back to the rear wall and the other crewman sleeps across the floor of the vehicle.

Since the vehicle is pressurized during descent and touchdown and because it enhances the speed and efficiency of operations the crew could be expected to remove their helmets and gloves while donning the extravehicular gear. These activities can be performed by a single individual but they go much more rapidly and with considerably more confidence when two men can perform an operation and check each other. Again, the complexity of the operation is one which makes it a formal checklist operation. The PLSS transfered from the CM and stowed temporarily on the floor of the vehicle is mounted in the harness called the donning station where it is suitably mounted adjacent to the checkout controls and where it can be held in place while the crewman donns the rest of the equipment which he requires. Figure 67 shows the TMG trousers.

The carrying straps for the PLSS are threaded through the TMG jacket to minimize the difficulties of thermal shorts in the garment. The jacket is difficult to donn but this is a tradeoff between ease of donning and the later cumbersomeness of excessive material.

Two men with the PLSS's and various other elements of the suit occupy a pretty substantial volume and when pressurized they move in a somewhat awkward fashion. It takes a great deal of time to perform a number of these operations because they are performed very very carefully.

The extravehicular visor, which provides various degrees of transmissibility, is donned prior to egress to protect the crewman from the extreme range of lighting values to be encountered.

Stationed on the initial portion of the ladder is an A-frame type step ladder. This is provided to enhance the ease of access to various portions of the descent stage, to contend with various orientations of the vehicle, and to make easier the crew's access to the fixed ladder on the front leg. Figure 68 shows that the first man out would free the device and guide its descent to the surface while the man remaining in the cockpit lowered it via the equipment lift line. Figures 69 and 70 show the completion of the descent cycle for the first man. The height of the fixed ladder above the surface for a "soft" landing, as illustrated, shows why the ladder is required.

It is apparent from the nature of these activities that substantial amounts of time are required to perform tasks that one expects to be done rather expeditiously. This accounts to some degree for the fashion in which we schedule crew time and for the fact that it is not very difficult during the course of the mission to encounter conditions wherein things take substantially more time than we have contemplated. The mere act of fastening a snap which can be very straightforward in one g and street clothes can become a very demanding operation in a pressurized suit under weightless conditions. The problem is less severe but not removed at 1/6 g.

All TMG's, PLSS's, and associated equipment must be returned to stowage locations so that the arrangements for sleep can be made. The harness which was used to hold the PLSS in a donning configuration is used to provide the hammock configuration for sleeping. The crew are to sleep suited in the LM. (Figure 71) The oxygen supply system has some capability in the event of a puncture but the thin skin and location on the lunar surface have a higher, though very low, probability of pressure vessel failure than we find in the CM. If there is a puncture the system can hold pressure in the vehicle for a substantial time period.

When the crew is ready to depart there is a substantial amount of equipment left behind (Figure 72) - the used LiOH cannisters from the LM itself and from the PLSS's, the batteries, bags containing the condensate collected in the PLSS, the urine, and other fecal materials (treated with germicidal agents) are all stowed in a compartment in the descent stage. These equipments are left behind in the interest of saving weight and volume in the ascent stage enhancing the amount of material which can be returned for scientific purposes.

Lunar Ascent

Lunar ascent has been described in considerable detail in a previous paper. Figure 73 shows the configuration of the vehicle at this time. The PLSS is stowed on the floor to ease access to the optical telescope which would be used to align the IMU just prior to liftoff. The other PLSS previously stowed in the recharge station has been discarded. The equipment is stowed in such a fashion as to maintain the symetry of loads as well as possible.

The crew activities during this maneuver are essentially the same as those which we have reviewed earlier for other powered flight maneuvers. (Figure 74) During the ascent portion of the mission the crew will again experience a 1/3 g acceleration environment which is quite acceptable on the standing configuration. The details of the rendezvous and docking sequence as executed by the guidance system have been discussed in detail earlier. It is perhaps well at this point to note that in addition to those activities it is possible to use the pilotage routines based upon observation of the FDAI and of the other instruments as they have been used in Gemini.

Rendezvous and Docking

In this phase of the mission the LM is the active element of the docking sequence. (Figure 75) The pilot of the LM can observe the command and service module through an overhead window (Figure 76) and he has a docking aid similar to the one shown earlier installed in the window of the CM. (Figure 77)

When the two vehicles have been docked there is a repetition of the sequence in which the docking mechanisms must be removed from the tunnel and then the crew can transfer the equipment and themselves back to the CM. (Figure 78) At this point in time a number of devices are left in the LM to relieve conjestion in the CM and to enhance the arrangements for stowage of significant items returned from the lunar surface. It is perhaps noteworthy that one of the PLSS's is retained in the CM, its oxygen supply being used as a backup to the entry oxygen supply during that portion of the mission. The TV camera is returned to the CM and the LM flight plan which is now the log and record of that flight is returned for record purposes. The sample return containers and the films from the sequence camera and other cameras are returned. The data storage electronic assembly is the voice recorder provided for crew usage in the LM. The devices transferred to the LM are those no longer required in future phases of the mission.

It is possible that during the docking sequence there can be a failure of the mechanism which would not allow transfer through the tunnel. There is a device called the EVCT (Figure 79) which is a metalic tape boom reeled out to some 25 feet in length which engages a device called the bailer bar at the command and service module interface. (Figure 80) Either the crewman in the CM can open the hatch or the hatch can be opened after the vehicle is depressurized by the crewman on the outside of the CM. The procedures would call for transfer of one of the crewmen from the LM using the PLSS and this device. He would then use this device to return the PLSS to the other crewman or, having established the configuration of the CM, the second crewman could be brought across using this device as a tether and using the EOS to provide breathing gases.

Transearth Injection

Transearth injection for the crew operations is very similar to the other flight maneuvers previously described, and the trajectory and guidance considerations are reviewed in other papers. It has a somewhat higher acceleration than the maneuvers discussed earlier, the light weight of the system now providing an effective 1/2 of a g acceleration.

Transearth Coast

Transearth coast is characterized by those activities which were discussed for the translunar leg of the mission. (Figure 81) The phenomena of apparent growth in size of the earth will occur on this leg.

Entry

The entry sequence has been described in some detail in a previous paper but it is perhaps important to note that a large number of sequential events are controlled directly by the crew and those not directly under their control in the nominal mode are subject to crew backup. (Figures 82 and 83) Some of the particular events are those noted in Figure 29 in items 5 and subsequent, excepting, of course, those items associated with LES tower operation.

SUMMARY

The critical role of the crew in providing flexibility and reliability during the mission has been emphasized. The relation of training equipments and procedures to such requirements has been noted. The lunar landing mission illustrates the wide range of capabilities of a manned spacecraft.

Questions and Answers

CREW TASKS AND TRAINING

Speaker: Joseph P. Loftus

1. Mr. Nix - Has consideration been given to opening the top hatch and looking around?

   ANSWER - Yes, and this approach has been rejected because it requires removing the drogue from the LM top hatch which is awkward and potentially hazardous. Also, the increased length of oxygen umbilical imposes pressure drop penalties upon the ECS and interference with general crew mobility. The view from the front of the LM is greater than 200 degrees and is considered adequate.

2. Mr. Davidson - How much of the time can the crew see the earth during translunar and transearth phases of the missiion mission?

   ANSWER - A substantial portion of the time. The field of view of the windows and optics is large and will make some view available most of the time.

3. Mr. Davidson - What is the effect of crew movement on spacecraft attitude?

   ANSWER - Effects are expected to be minor.

4. Mr. Beattie - Is the copilot task during LM landing essentially a monitoring one?

   ANSWER - No, essentially a team operation is planned.

5. Dr. Reiffel - Is there time allocated for inflight experiments?

   ANSWER - There will possibly be some time available since not all of the crew time has been allocated to specific spacecraft or mission operations.

6. Dr. Reiffel - Do you plan to sterilize the food containers and fecal cannisters?

   ANSWER - Yes. Germicidal provisions are included in each of the containers.

7. Dr. Von Braun - Since there is concern over the effect on LM thermal protection from the SM/RCS during extraction and subsequent operations, is short duration pulsing of the RCS really going to be effective in reducing degradation of the thermal coating?

   ANSWER - The condition is still under study; however, the limited total duration of RCS firing is not expected to result in significant degradation.

NASA-S-66-5241 JUN

CREW PERFORMANCE REQUIREMENTS

- ABORT INITIATION
- FLIGHT CONTROL
- SYSTEM MANAGEMENT
- NAVIGATION
- ONBOARD MISSION MANAGEMENT

Figure 1

NASA-S-66-5236 JUN

CREW ORGANIZATION

- CREW POSITION
    - COMMAND PILOT
    - SENIOR PILOT
    - PILOT
- CROSS TRAINED IN ALL FUNCTIONS
- TRAINED AS A TEAM

Figure [illegible]

NASA-S-66-5239 JUN

TYPES OF CREW TRAINING

- SYSTEMS FAMILIARIZATION
- SPECIFIC TASK TRAINING
- MISSION SEGMENT TRAINING
- SPECIFIC MISSION TRAINING
- EXPERIMENT ACTIVITY TRAINING

Figure 3

NASA-S-66-5238 JUN

ELEMENTS OF CREW TRAINING

- NORMAL MODE CREW ACTIVITY
- MALFUNCTION RECOGNITION AND RESPONSE
- EMERGENCY MODE CREW ACTIVITY
- CREW INTEGRATION

Figure 4

NASA-S-66-5237 JUN

SUBSYSTEMS

- LAUNCH ESCAPE
- GUIDANCE & NAVIGATION
- STABILIZATION & CONTROL
- REACTION CONTROL
- PROPULSION
  - SERVICE PROPULSION
  - DESCENT PROPULSION
  - ASCENT PROPULSION
- ENVIRONMENTAL CONTROL
- ELECTRICAL POWER
- COMMUNICATION
- RADAR
- SEQUENTIAL EVENTS
- EARTH LANDING
- EXTRAVEHICULAR MOBILITY UNIT

Figure 5

NASA-S-66-5240 JUN

TRAINING FACILITIES

- SYSTEMS TRAINERS
- PART TASK TRAINERS
  - MOCKUPS
  - DYNAMIC CREW PROCEDURES TRAINER
  - TRANSLATION AND DOCKING TRAINER
  - LUNAR LANDING TRAINING VEHICLE
  - EGRESS TRAINER
- MISSION SIMULATORS
  - COMMAND & SERVICE MODULE
  - LUNAR MODULE
- SPECIAL FACILITIES
  - CENTRIFUGE
  - AIR BEARING TRAINER
  - ZERO 'G' AND 1/6 'G' AIRCRAFT FLIGHTS
  - ENGINEERING SIMULATORS

Figure 6

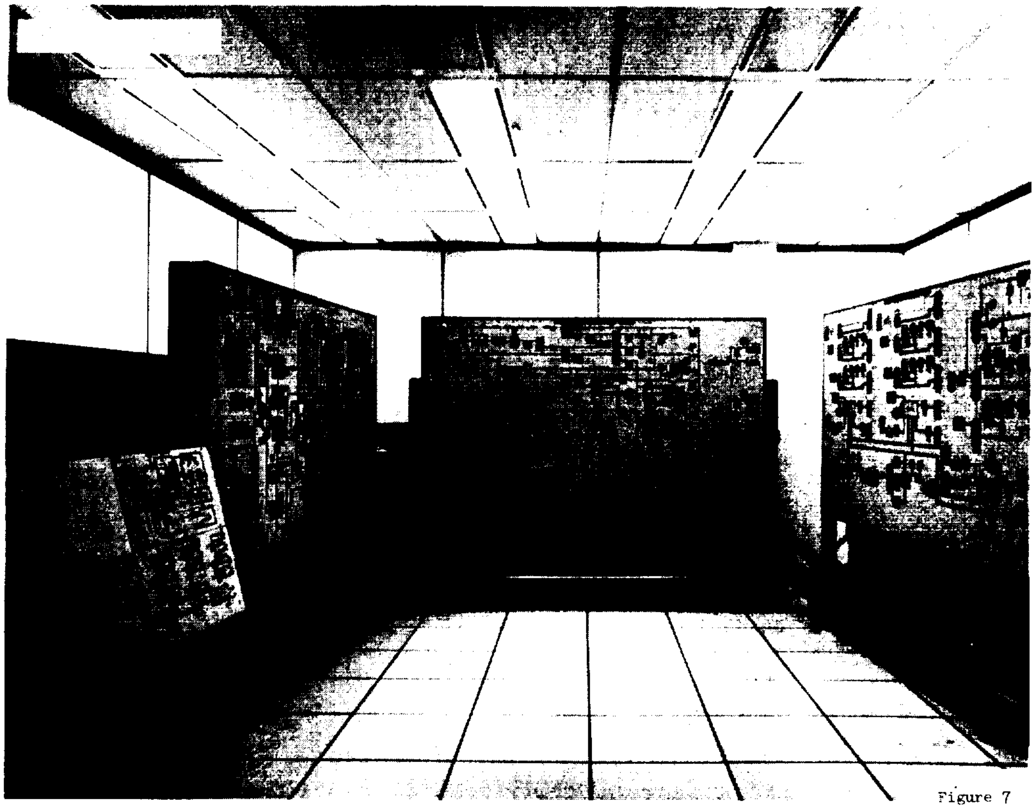


Figure 7

NASA-S-65-5753 APRIL 28, 1965

APOLLO BLOCK II MOCKUP

COMMAND MODULE BLOCK II MOCKUP - INTERIOR CONFIGURATION ONLY - SHOWN ON TRANSPORTATION DOLLY

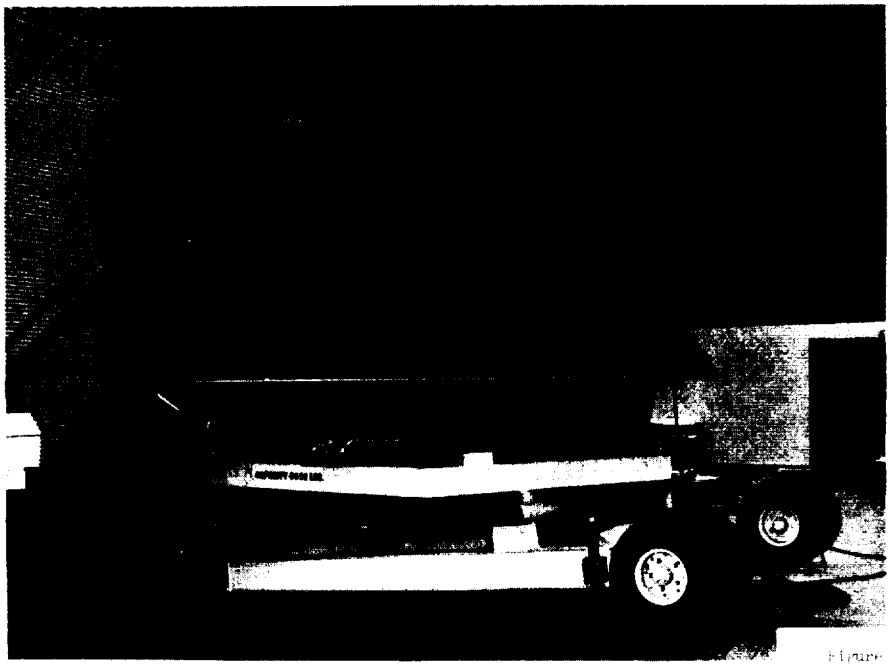


Figure 8

NASA-S-66-6147 JUNE

LM MOCKUP

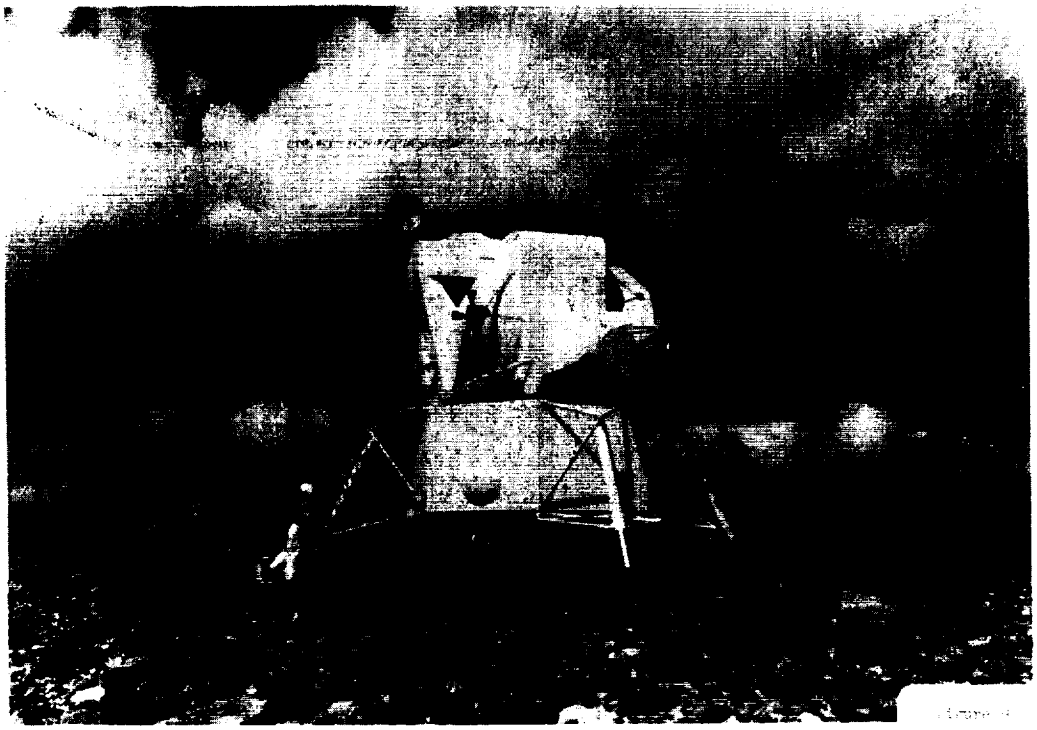


Figure 9

NASA-S-66-5247 JUN

DYNAMIC CREW PROCEDURES TRAINER CM CONFIGURATION

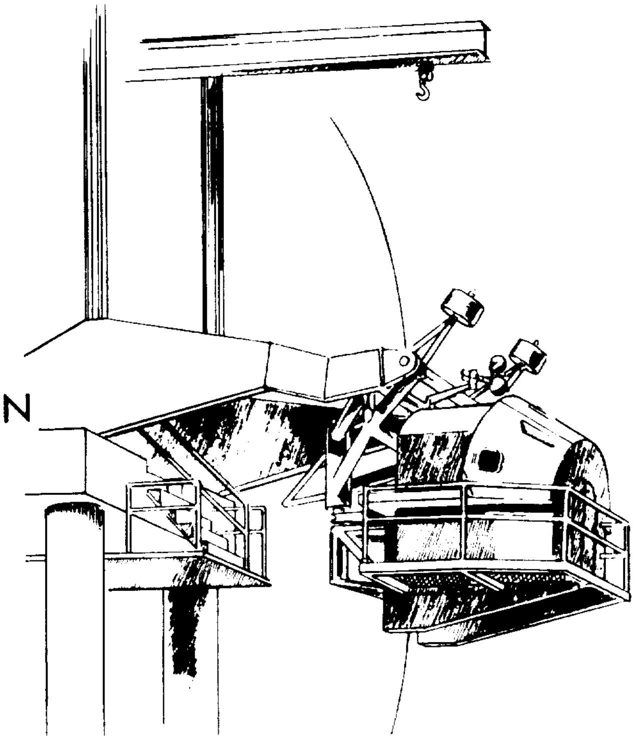


Figure 10

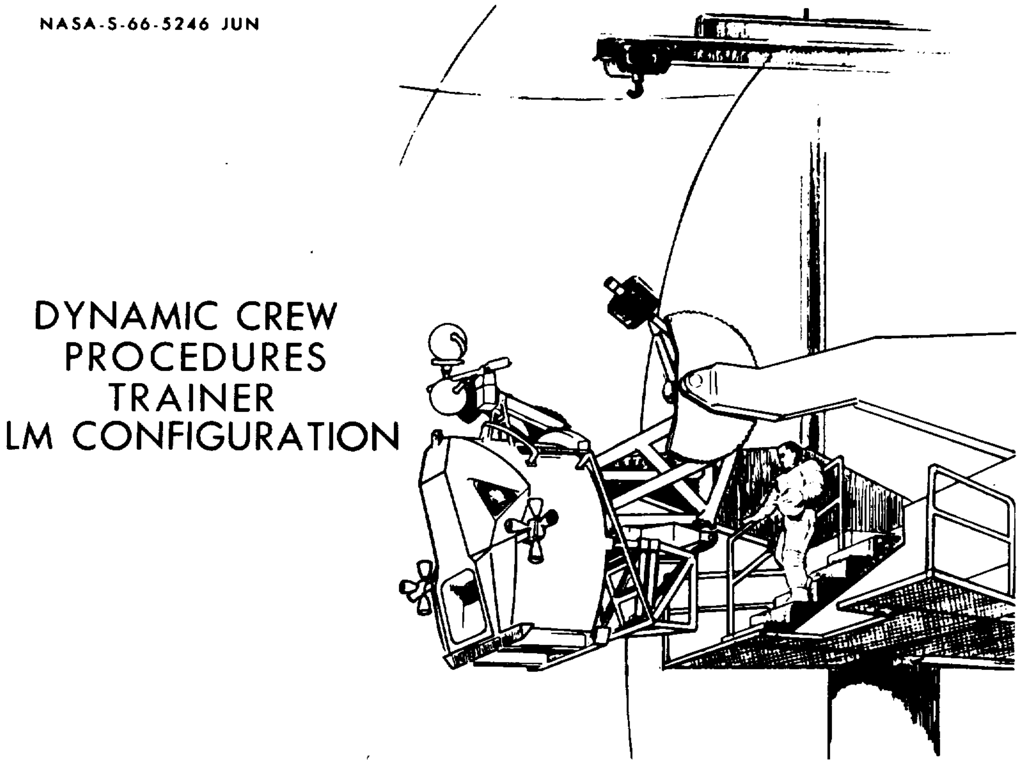

NASA-S-66-5246 JUN
DYNAMIC CREW
PROCEDURES
TRAINER
LM CONFIGURATION

Figure 11

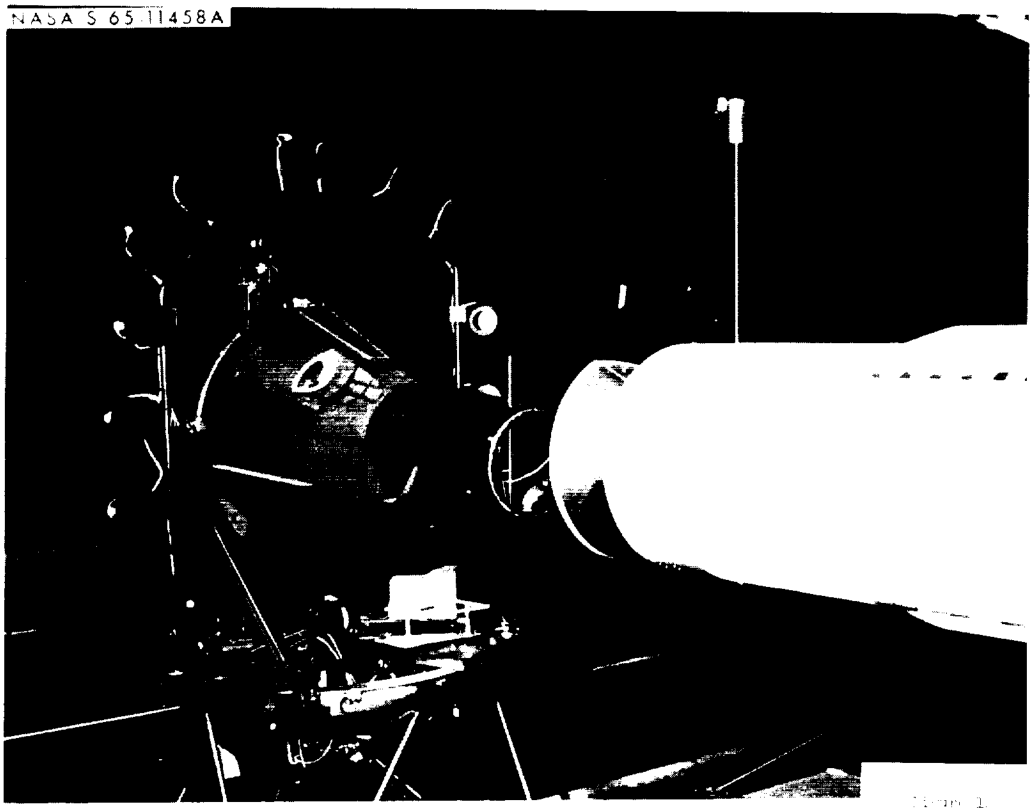

NASA S 65-11458A

Figure 1

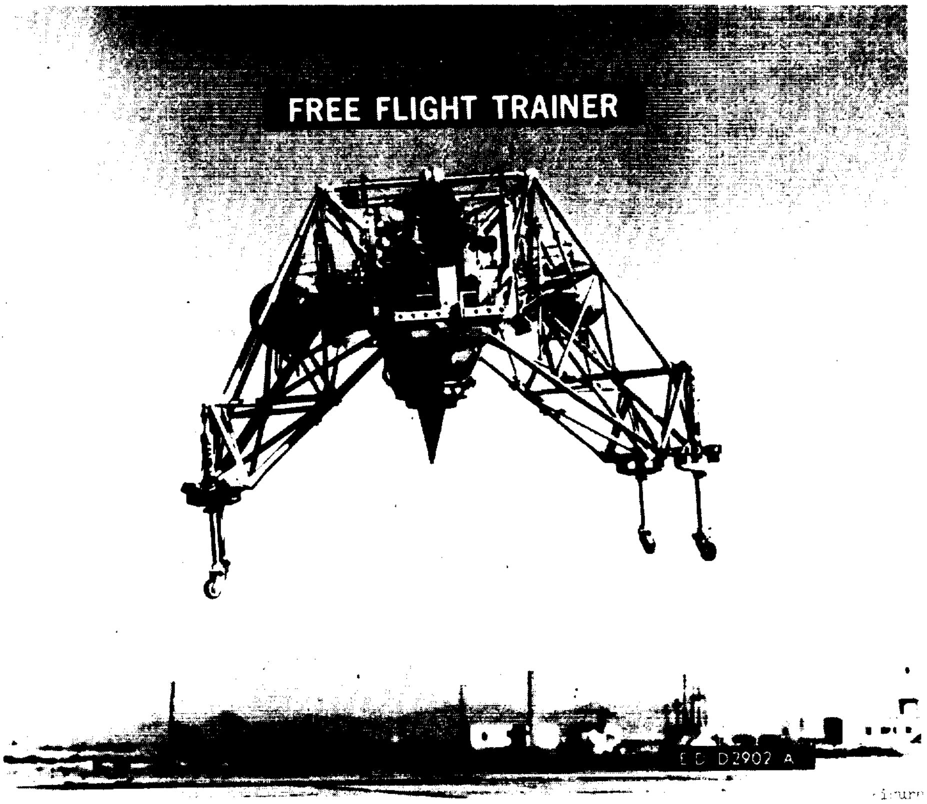

FREE FLIGHT TRAINER
E C D2902 A

Figure 13

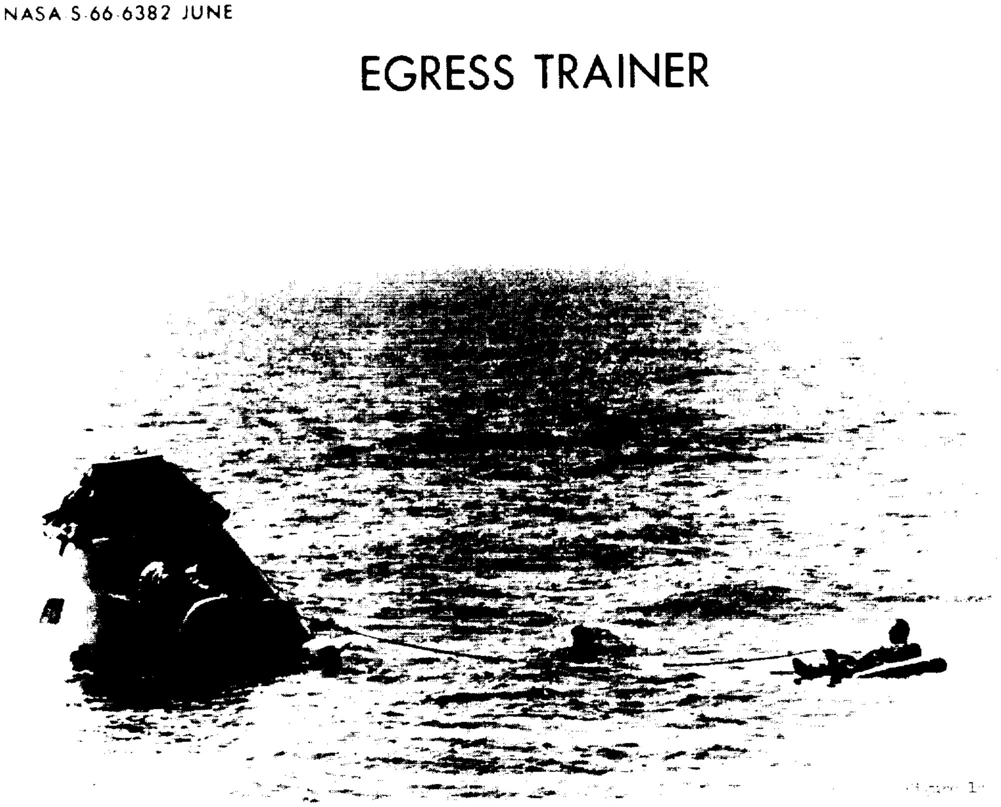

NASA S-66-6382 JUNE
EGRESS TRAINER

Figure 1

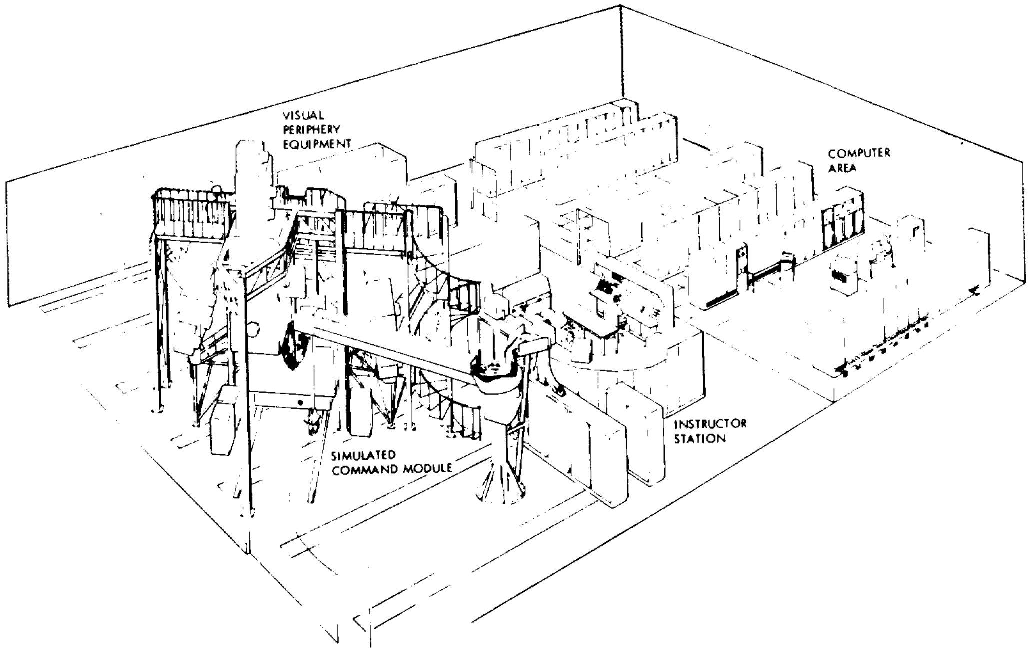

VISUAL
PERIPHERY
EQUIPMENT
COMPUTER
AREA
INSTRUCTOR
STATION
SIMULATED
COMMAND MODULE

Figure 15

NASA-S-66-5231 JUN

LM MISSION SIMULATOR

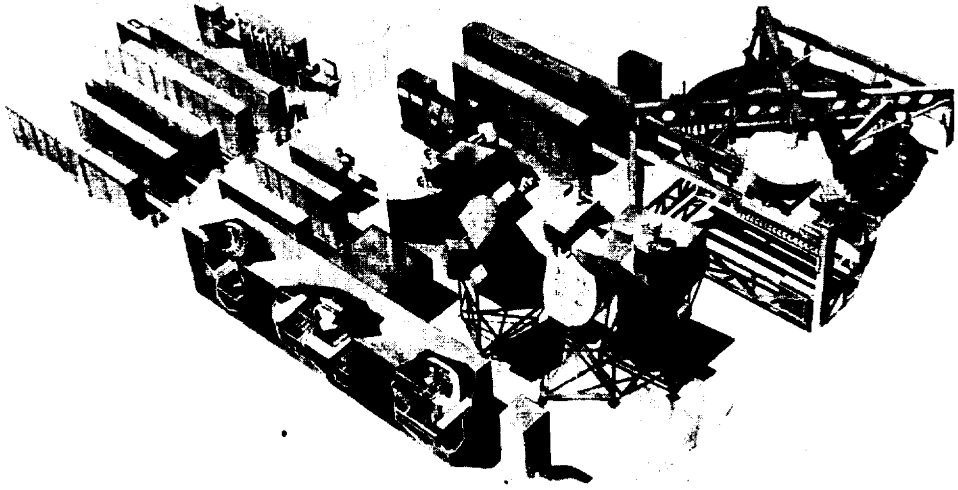
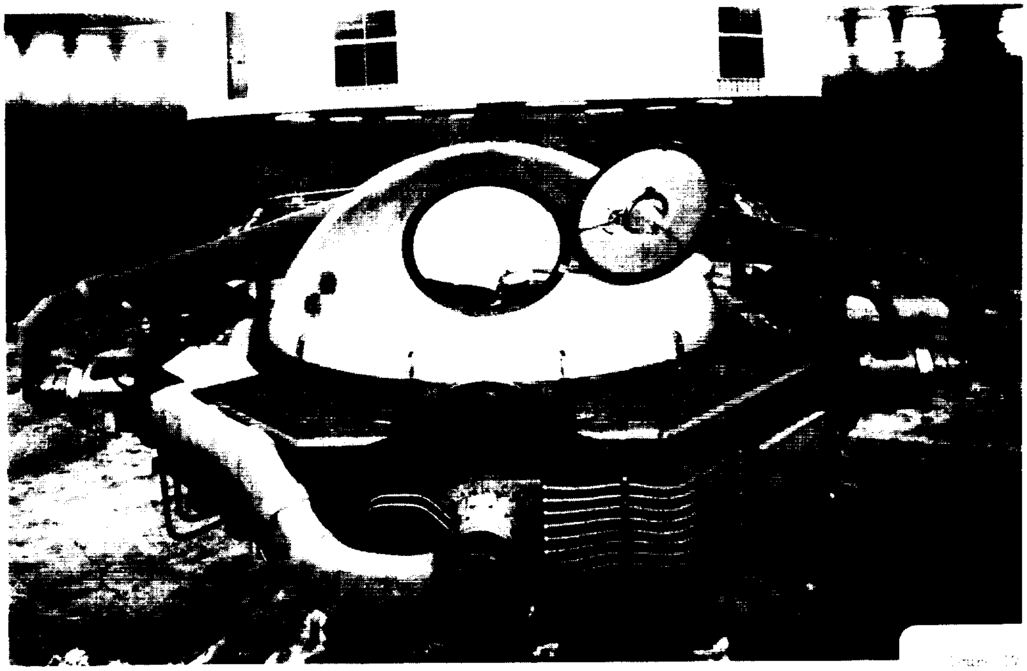


Figure 16

NASA S 66 4240 APR 26

FLIGHT ACCELERATION FACILITY

THREE MAN GIMBLING GONDOLA
SIMULATES ACCELERATION UP TO 30 G



[illegible]

U.S. AIR F
3129

[illegible]

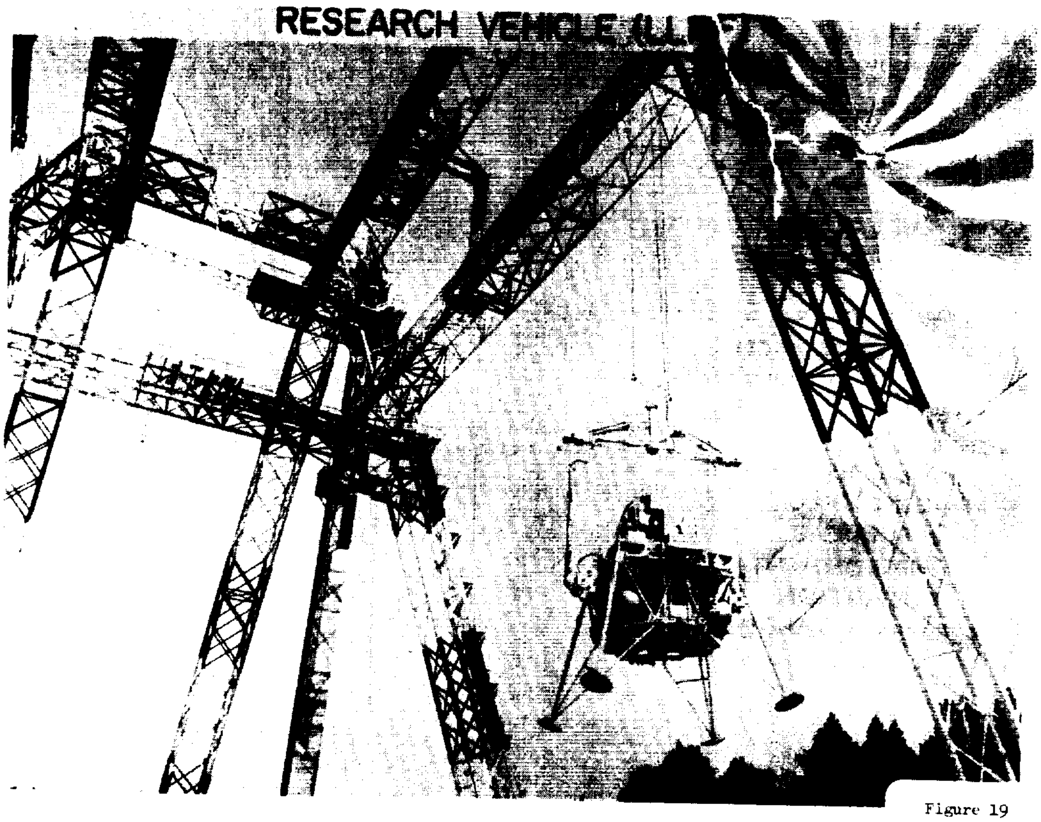

RESEARCH VEHICLE (LL

Figure 19

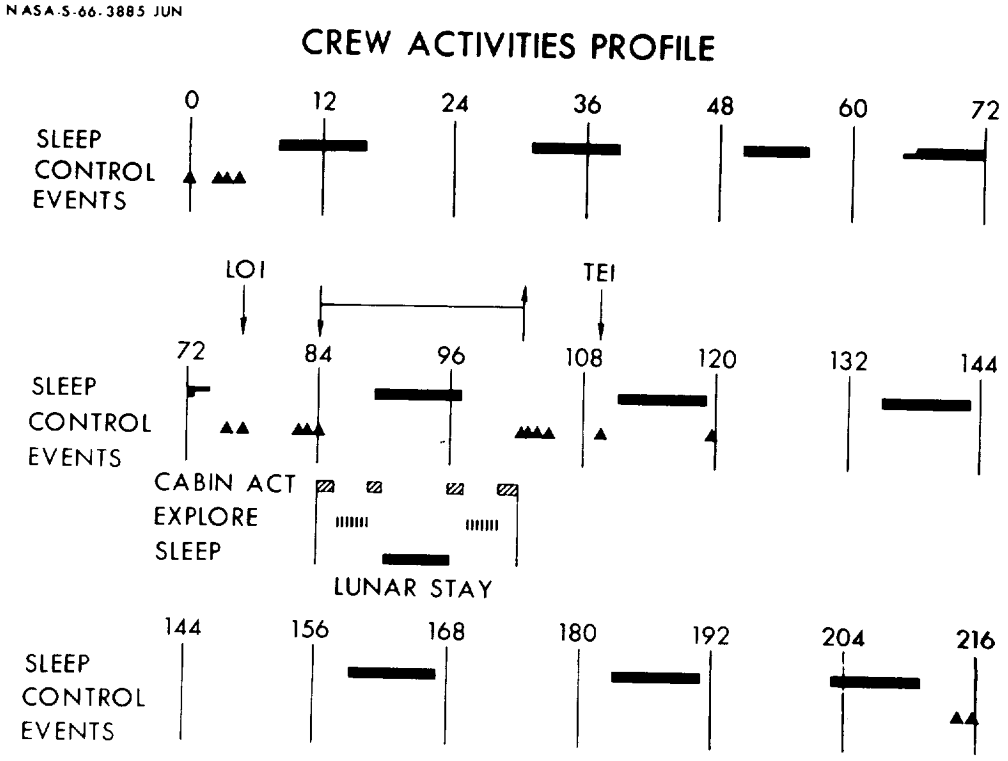


Figure 20

NASA-S-66-6075 JUN

CREW ACTIVITIES

TASK	MISSION TIME % TIME	MISSION TIME CUM %	CREW MANHOURS % TIME	CREW MANHOURS CUM %
CONTROL	01	01	01	01
ATTITUDE				
ΔV BURNS				
TRANSPOSITION & DOCKING				
LM DESCENT & ASCENT				
RENDEZVOUS & DOCKING				
ENTRY				
MONITOR	12	13	04	05
GENERAL DISPLAY SCAN				
PERIODIC CHECK				
ELECTRICAL				
ENVIRONMENTAL				
PROPULSION				
SUBSYSTEM MONITORING				

Figure 21

NASA-S-66-6076 JUN

CREW ACTIVITIES (CONT)

TASK	MISSION TIME % TIME	MISSION TIME CUM %	CREW MANHOURS % TIME	CREW MANHOURS CUM %
SYSTEM MAINTENANCE	03	16	01	06
LiOH CANNISTER CHANGE				
FUEL CELL H_2 PURGE				
FUEL CELL O_2 PURGE				
BATTERY CHARGE				
PLSS RECHARGE				
GUIDANCE & NAVIGATION	03	19	02	08
RADAR TRACKING				
OPTICAL TRACKING				
COMMUNICATION	03	22	02	10
SCHEDULED VOICE REPORTS				
DATA RECORDING				
DATA DUMP				
BIO-MED TRANSMIT				
CHECKOUT	02	24	02	12
LM PRE-SEPARATION				
LM PRE-LAUNCH				

Figure

NASA-S-66 6077 JUN

CREW ACTIVITIES (CONT)

TASK	MISSION TIME		CREW MANHOURS	
	% TIME	CUM %	% TIME	CUM %
FUNCTIONAL TASKS	03	27	03	15
STATION CHANGES				
SUIT DON & DOFF				
EQUIPMENT REMOVAL				
EQUIPMENT STOWAGE				
MAKING HARD DOCK				
CLEAR TUNNEL				
SECURE TUNNEL				
CREW TRANSFER				
EQUIPMENT TRANSFER				

Figure 23

NASA-S-66 6078 JUN

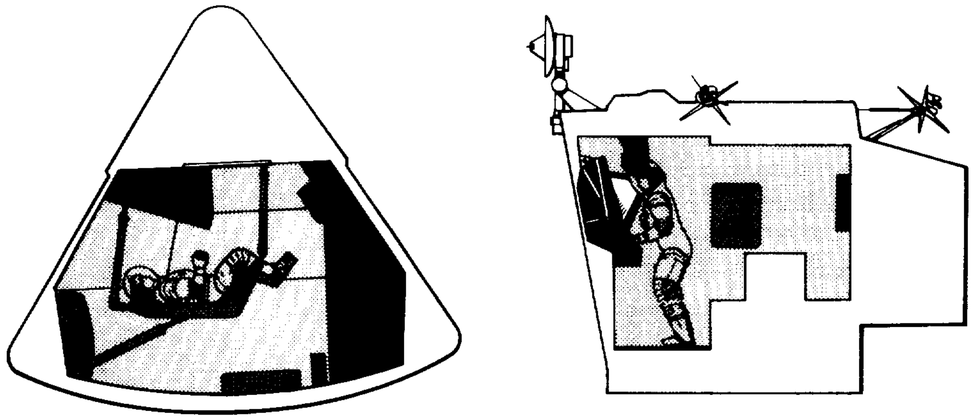
CREW ACTIVITIES (CONT)

TASK	MISSION TIME		CREW MANHOURS	
	% TIME	CUM %	% TIME	CUM %
EXPLORATION	02	29	02	17
PHOTOGRAPHY				
SCIENTIFIC EQUIPMENT SETUP				
SAMPLE GATHERING				
VEHICLE INSPECTION				
SURFACE INSPECTION				
LIFE SUPPORT	07	36	04	21
FOOD PREPARATION & EATING				
BODY FUNCTIONS				
HYGIENE				
SLEEP	33	69	33	54
NO SCHEDULED ACTIVITY (INFLIGHT EXPERIMENTS WILL CONSUME A PORTION OF THIS TIME)	31	100	46	100
TOTAL HOURS		216		648

Figure 24

NASA-S-66-6131-JUN

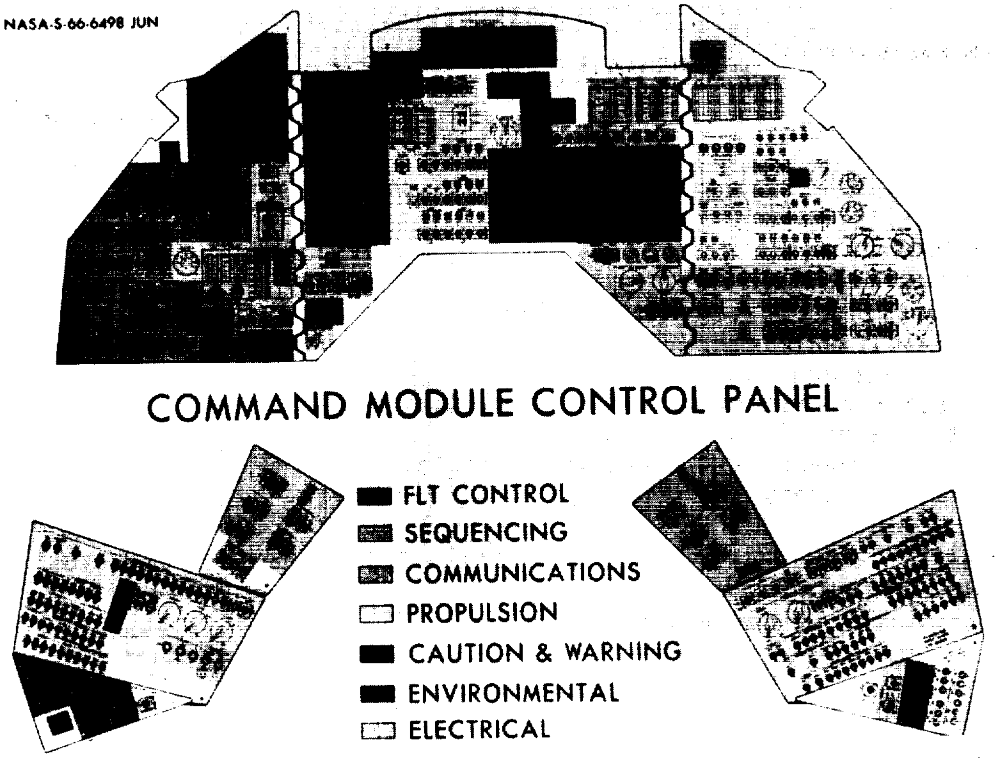
VEHICLE VOLUMES



TOTAL PRESSURIZED VOLUME	- 306 FT^3	TOTAL PRESSURIZED VOLUME	- 235 FT^3
CREW COMPARTMENT VOLUME	- 245 FT^3	CREW COMPARTMENT VOLUME	- 218 FT^3
EFFECTIVE FREE VOLUME	- 210 FT^3	EFFECTIVE FREE VOLUME	- 190 FT^3

Figure [illegible]

NASA-S-66-6498 JUN



COMMAND MODULE CONTROL PANEL

■ FLT CONTROL

▩ SEQUENCING

▨ COMMUNICATIONS

☐ PROPULSION

■ CAUTION & WARNING

■ ENVIRONMENTAL

☐ ELECTRICAL

Figure 16

NASA-S-66-6438 JUN

LAUNCH

- **MONITOR LAUNCH VEHICLE PERFORMANCE**
  - **ATTITUDE-ATTITUDE RATE**
  - **STAGING SEQUENCES**
  - **THRUST LEVELS**
  - **TANK PRESSURES**
- **MONITOR GUIDANCE**
- **MONITOR MISSION SEQUENCES**
- **MONITOR SPACECRAFT SYSTEMS**
- **MONITOR COMMUNICATION MODES**

Figure 27

NASA S 66 6139 JUN

LAUNCH CONFIGURATION

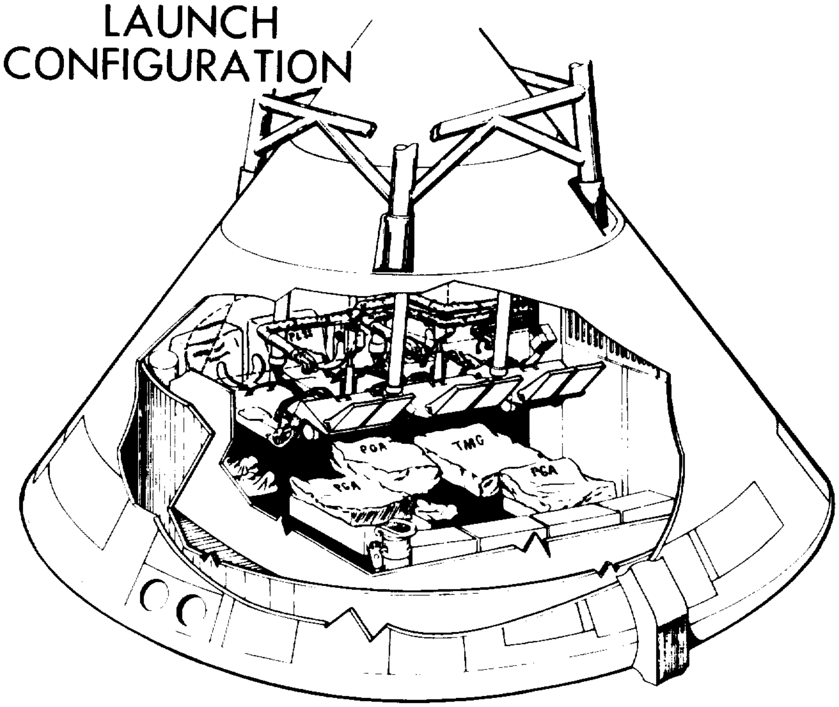


Figure 28

NASA-S-66-6824 JUN

ABORT AFTER 61 SEC

1 ABORT DECISION
2 TRANSLATION HAND CONTROLLER (ROTATE CCW)
3 EVENT TIMER "CHECK"
4 LES MOTOR FIRE
5 CM RCS PRESSURIZED ?
6 CANARDS DEPLOY ?
7 ELS LOGIC CIRCUIT ARMED ?
8 LES TOWER JETT ?
9 APEX COVER JETT ?
10 DROGUE CHUTES DEPLOYED ?
11 ALTIMETER - MONITOR CHUTES DEPLOY
12 MAIN CHUTES DEPLOYED ?
13 CABIN PRESS RELIEF VALVES "CLOSE"
14 CM PROP JETT DUMP SW "DUMP"
15 PROP JETT PURGE SW "PURGE"
16 RELEASE MAIN CHUTE - *AFTER LANDING*

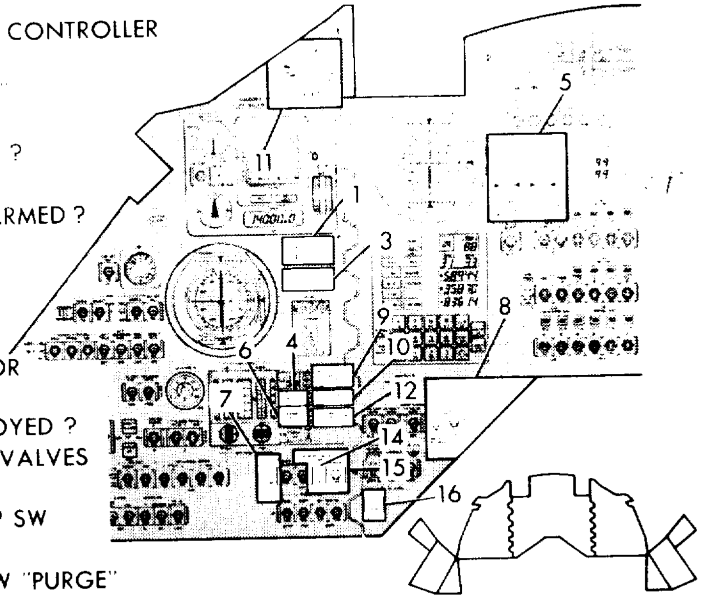


Figure [illegible]

NASA-S-66-6590 JUN

EARTH ORBIT

- ORIENT VEHICLE FOR IMU ALIGNMENT
- ORIENT FOR LANDMARK TRACKING
- EXTEND AND SECURE DOCKING PROBE
- ORIENT FOR IMU ALIGNMENT
- ORIENT FOR TLI

- ENTER LEB
- ALIGN IMU
- TRACK LANDMARKS
- ALIGN IMU

- CHARGE BATTERY
- TRANSMIT HI BIT RATE PCM
- VERIFY CAUTION & WARNING & SPACECRAFT SYSTEM STATUS

Figure 30

NASA S 66 6154 JUNE

OPTICAL SIGHTING

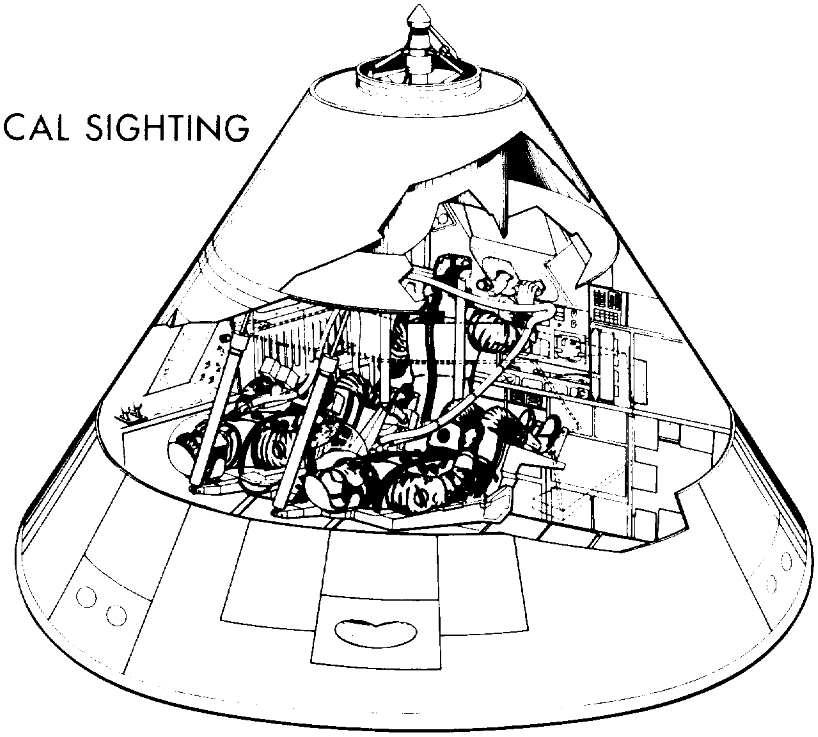


Figure 31

NASA-S-66-6592 JUN

TRANSLUNAR INJECTION

- ORIENT FOR ALIGNMENT
- MONITOR IMU ALIGNMENT
- CAGE BODY MOUNTED GYROS
- COUNTDOWN
- MONITOR SIVB

- PERFORM IMU FINE ALIGNMENT
- MONITOR SYSTEMS

- PURGE FUEL CELLS
- CHARGE BATTERIES
- PREPARE FOR PEAK POWER
- MONITOR SYSTEMS

NASA-S-66-6451 JUN

TRANSPOSITION DOCKING

- PREPARE FOR CSM SEPARATION
- SEPARATE CSM-LM FROM S-IV-B
- PERFORM SEPARATION PITCHOVER AND CLOSING MANEUVER
- ALIGN AND EXECUTE DOCKING

- CONFIRM MSFN TRAJECTORY
- MONITOR STATE VECTOR UPDATE
- PRESSURIZE TUNNEL AND LM
- REMOVE HATCHES
- SET DOCKING LATCHES
- RESTORE TUNNEL AREA

- SHIFT COUCHES TO DOCKING POSITION
- POWER DOWN EPS AND CHANGE BATTERIES
- ALIGN HIGH GAIN ANTENNA

Figure 33

NASA-S-66 6825 JUN

TRANSLUNAR TRANSPOSITION AND DOCKING

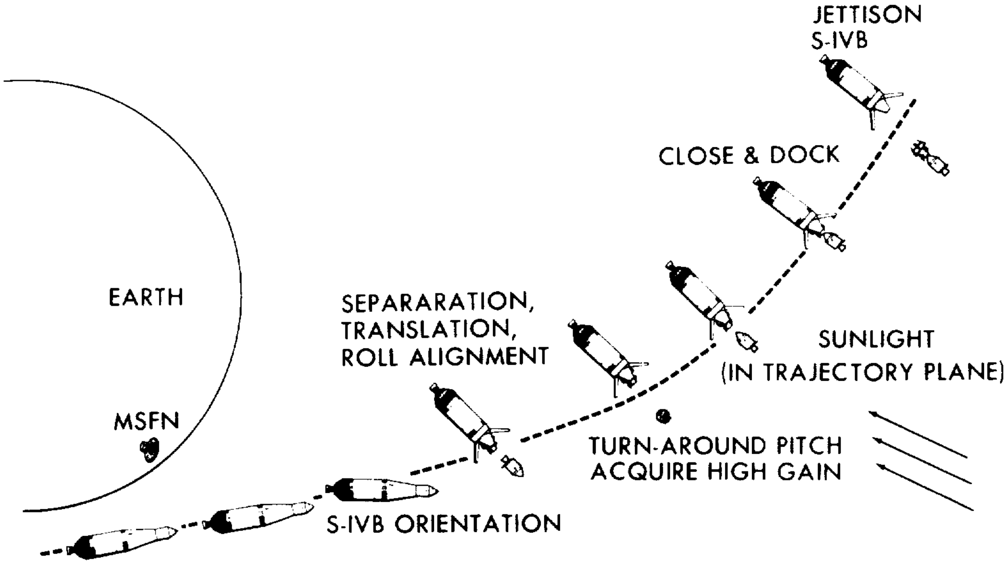

JETTISON
S-IVB
CLOSE & DOCK
EARTH
SEPARARATION,
TRANSLATION,
ROLL ALIGNMENT
SUNLIGHT
(IN TRAJECTORY PLANE)
MSFN
TURN-AROUND PITCH
ACQUIRE HIGH GAIN
S-IVB ORIENTATION

Figure 34

NASA-S-66-6132-JUN

DOCKING AND FLIGHT CONFIGURATION

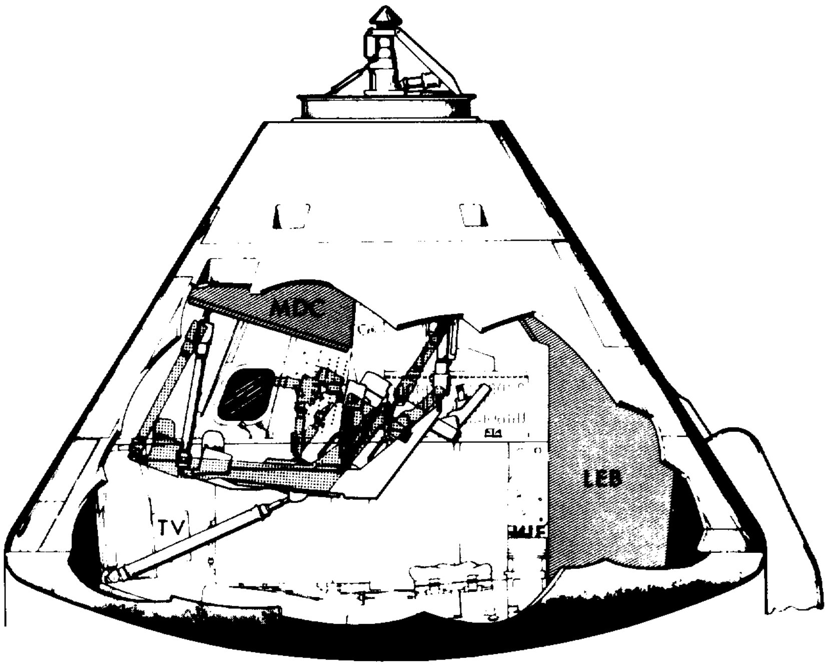


Figure 35

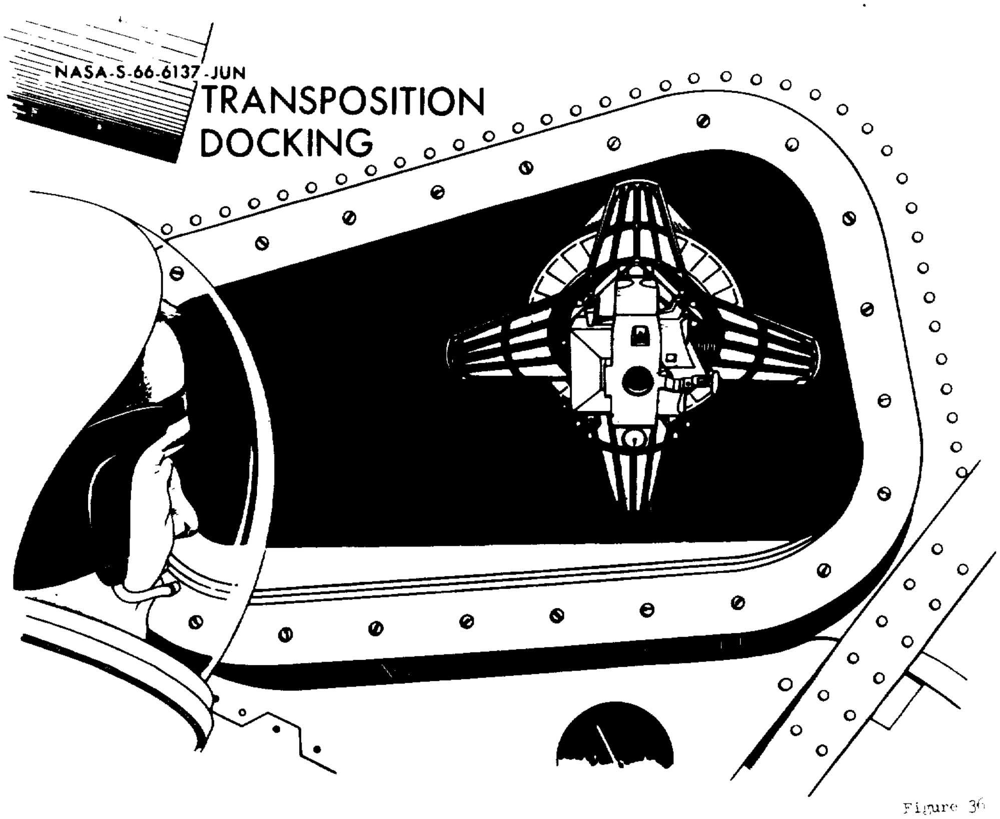

NASA-S-66-6137-JUN
TRANSPOSITION
DOCKING

Figure 36

NASA
S-65-17771

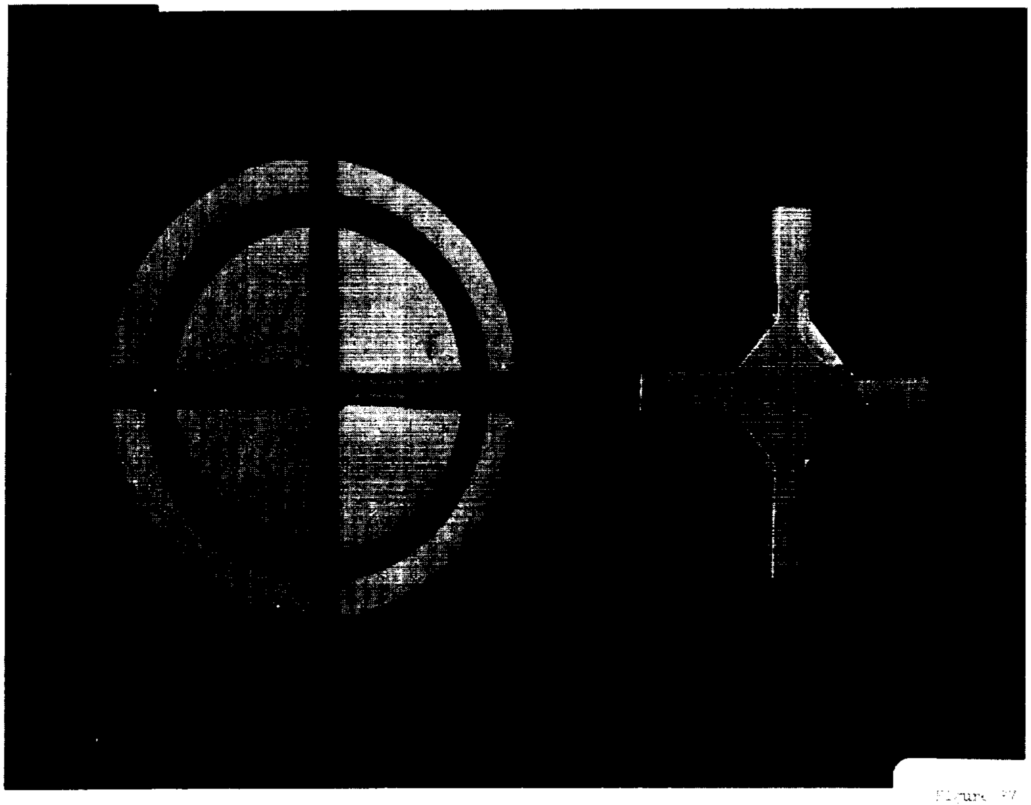
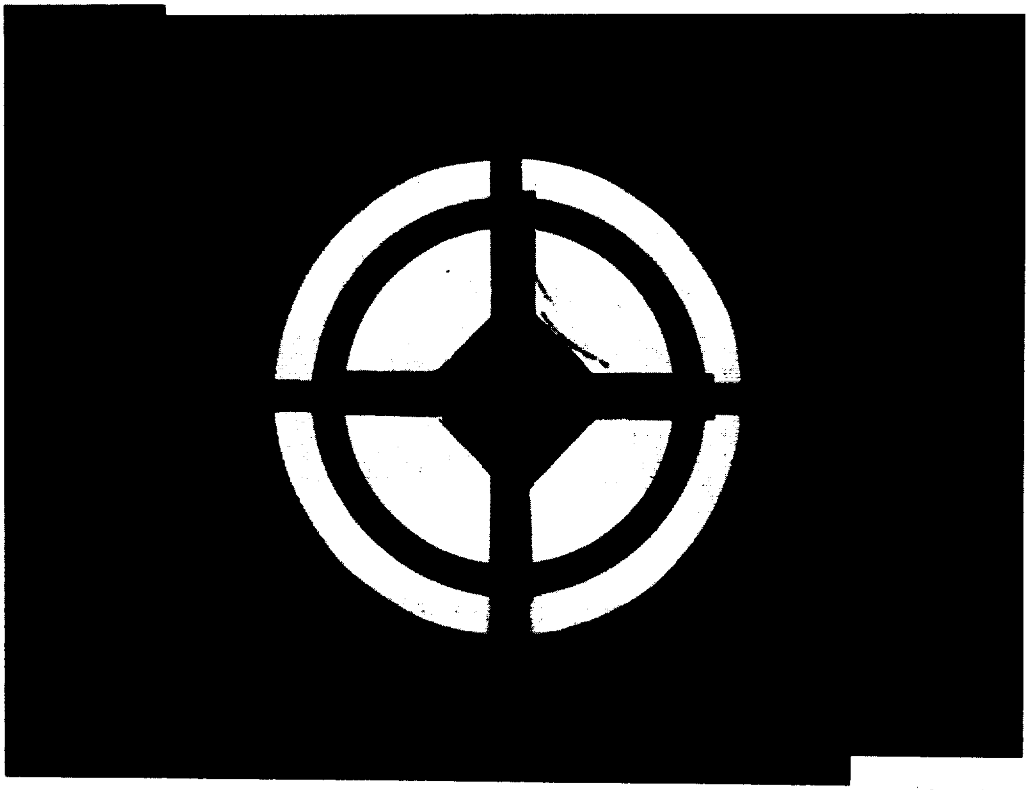


Figure 37

NASA
S-65-17774



NASA-S-66-6136-JUN

HARD-DOCK LATCHING

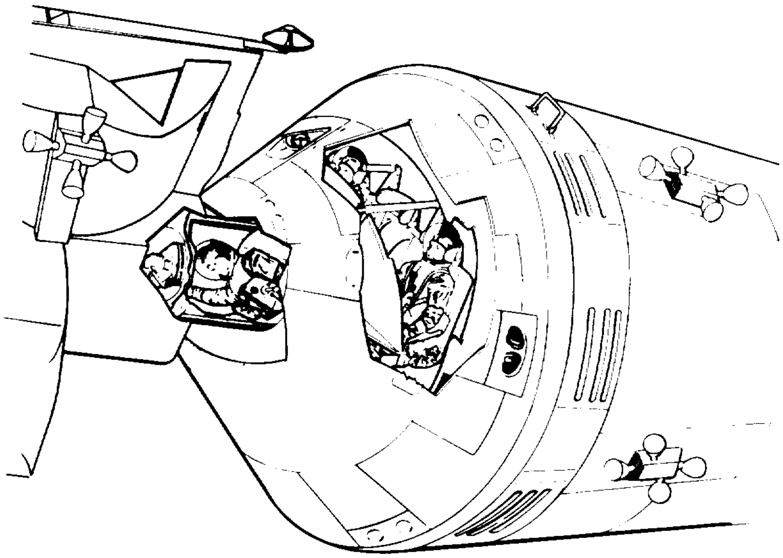


Figure 39

NASA-S-66-6489 JUN

TRANSLUNAR COAST

- MIDCOURSE CORRECTIONS
- PERIODIC CHECKS
- PERIODIC MAINTENANCE
- SLEEP

Figure 40

NASA-S-66-6845 JUNE

STOWAGE COMPARTMENTS

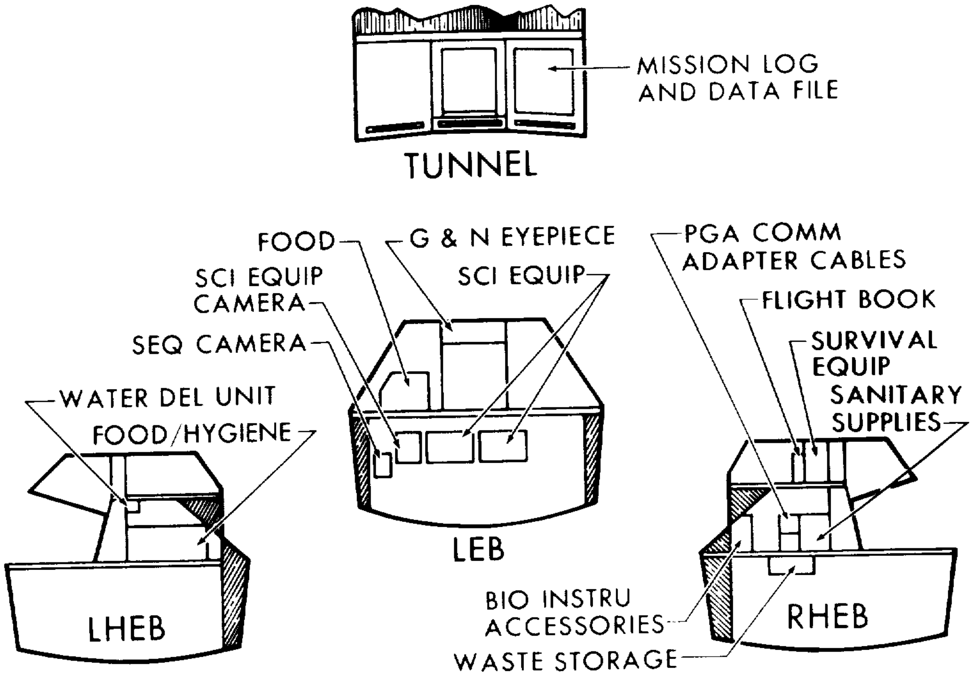

MISSION LOG
AND DATA FILE
TUNNEL
FOOD
G & N EYEPIECE
SCI EQUIP
CAMERA
SCI EQUIP
SEQ CAMERA
WATER DEL UNIT
FOOD/HYGIENE
LEB
LHEB
PGA COMM
ADAPTER CABLES
FLIGHT BOOK
SURVIVAL
EQUIP
SANITARY
SUPPLIES
BIO INSTRU
ACCESSORIES
WASTE STORAGE
RHEB

Figure 41

NASA-S-66-6861 JUN

BLOCK II STOWAGE ITEMS

		UNITS	TOTAL OPERATIONS (ALL UNITS)
A	EMU HARDWARE & CONTAINERS	28	160
B	TV & SEQUENCE CAMERA & ACCESSORIES	22	109
C	HYGIENE EQUIPMENT	105	218
D	MEDICAL EQUIPMENT	108	253
E	RADIATION MONITORING EQUIPMENT	22	87
F	CREW CARRY-ON ACCESSORIES	27	289
G	SURVIVAL EQUIPMENT	6	9
H	DOCKING & EXTRAVEHICULAR TRANSFER EQUIPMENT	8	23
I	G&N LOOSE EQUIPMENT	7	65
J	LOOSE SPACECRAFT SUPPORT HARDWARE	34	812
K	FOOD PACKS	168	168
L	LiOH CANISTER	56	117
M	EXPERIMENT HARDWARE, TYPICAL	9	90
		600	2400

[illegible]

NASA S 66-6153 JUNE

SLEEP STATIONS

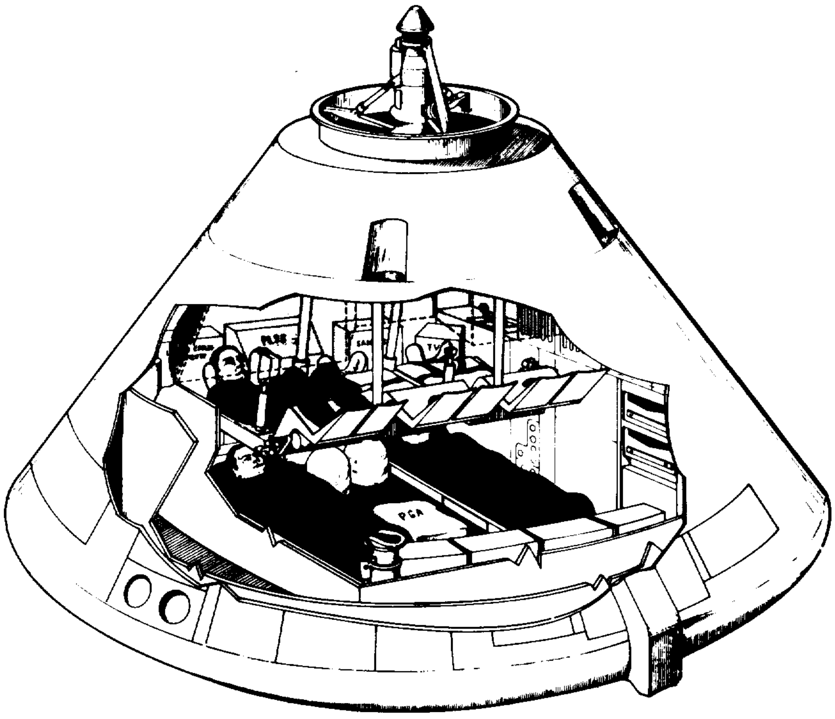
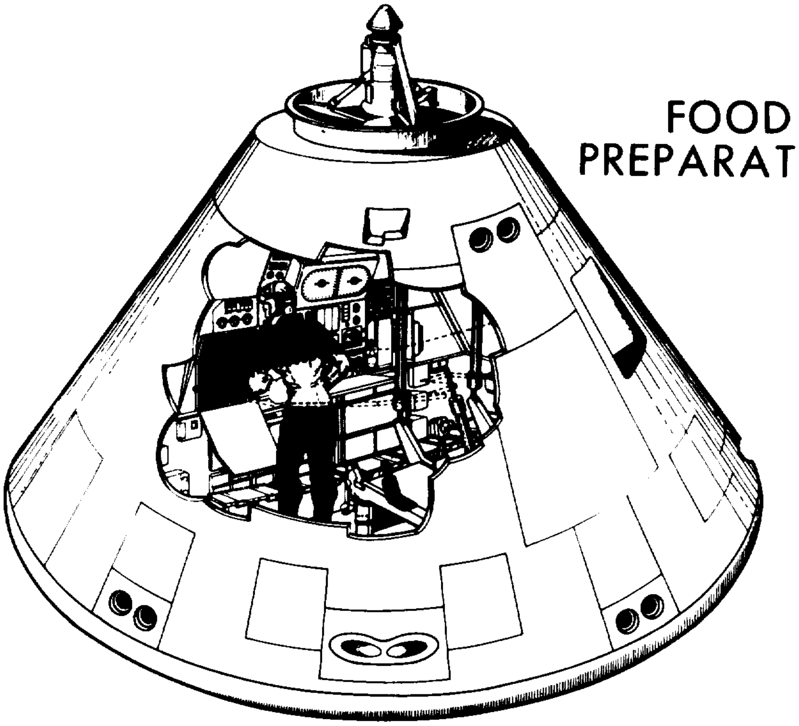


Figure 43

NASA-S-66-6148 JUNE



FOOD PREPARATION

Figure 44

NASA-S-66-6138-JUN

LiOH CANISTER CHANGE

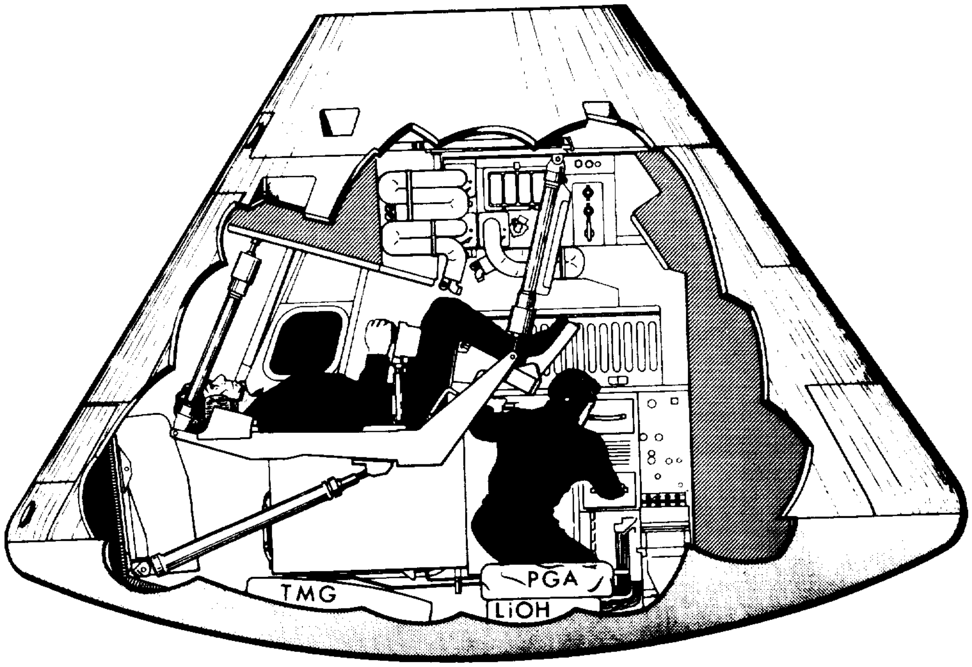
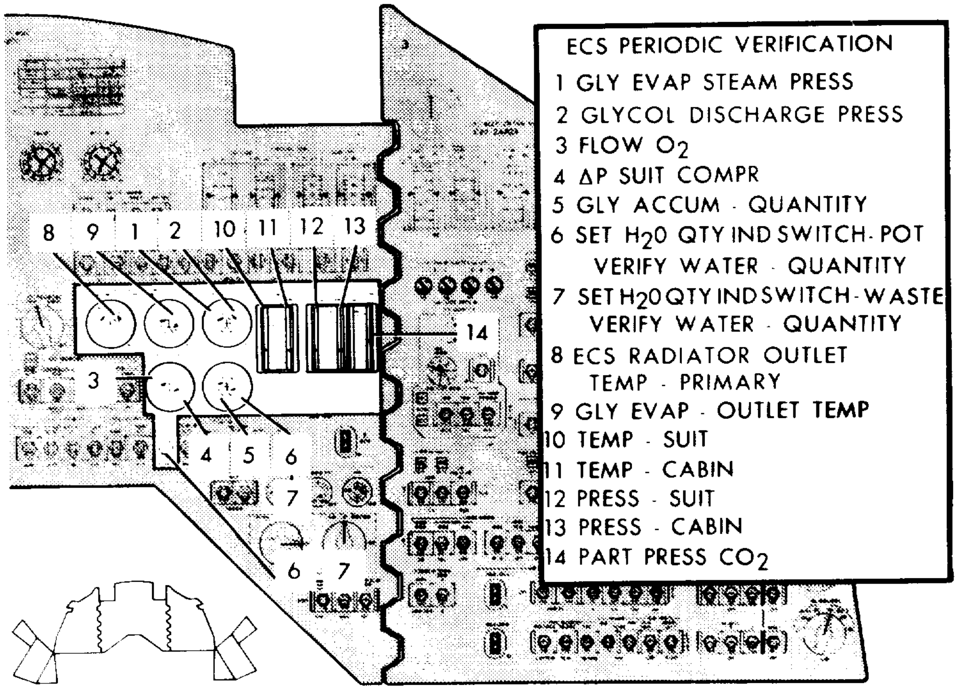


Figure 45

NASA-S-66-6823 JUN



ECS PERIODIC VERIFICATION

1 GLY EVAP STEAM PRESS
2 GLYCOL DISCHARGE PRESS
3 FLOW O_2
4 ΔP SUIT COMPR
5 GLY ACCUM - QUANTITY
6 SET H_2O QTY IND SWITCH - POT
VERIFY WATER - QUANTITY
7 SET H_2O QTY IND SWITCH - WASTE
VERIFY WATER - QUANTITY
8 ECS RADIATOR OUTLET
TEMP - PRIMARY
9 GLY EVAP - OUTLET TEMP
10 TEMP - SUIT
11 TEMP - CABIN
12 PRESS - SUIT
13 PRESS - CABIN
14 PART PRESS CO_2

Figure 46

NASA-S-66-6149 JUNE

OPTICAL SIGHTING

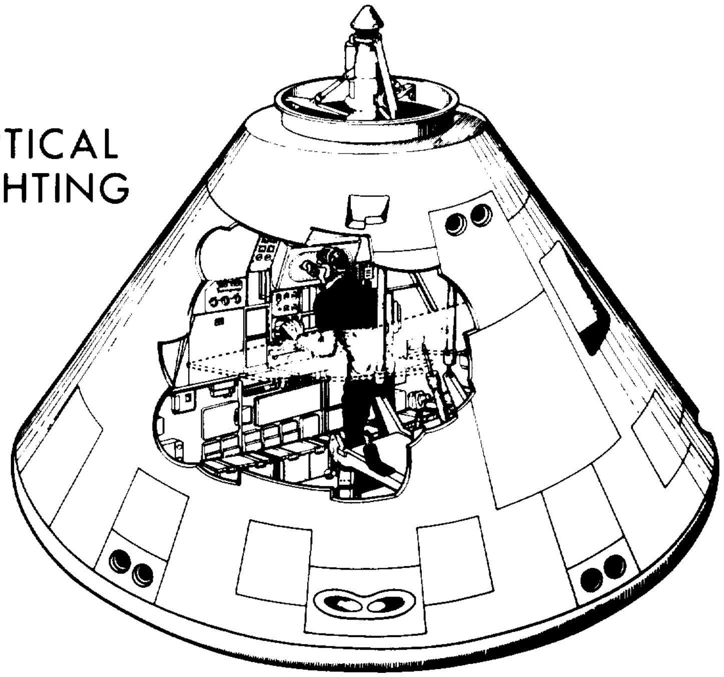


Figure 47

NASA-S-66-6445 JUN

LM CHECKOUT

- PRESSURIZE TUNNEL AND LM
- REMOVE PRESSURE HATCH, ABLATIVE HATCH, PROBE, & DROGUE
- ENTER LM - CHECK CONTROL SYSTEMS

- ENTER LM
- CHECK ELECTRICAL AND ENVIRONMENTAL SYSTEMS
- ALIGN IMU

- STOW DOCKING EQUIPMENT
- MONITOR CSM SYSTEMS

Figure 48

NASA-S-66-6152 JUNE

PREPARATION FOR TRANSFER

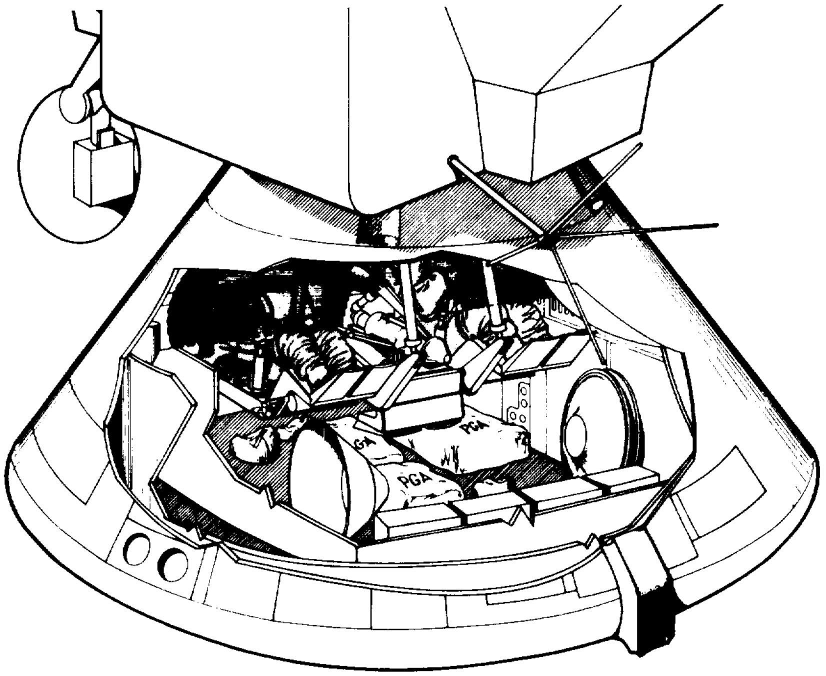

PGA
PGA
PGA
PGA

Figure 49

NASA-S-66-6150 JUNE

CREW TRANSFER

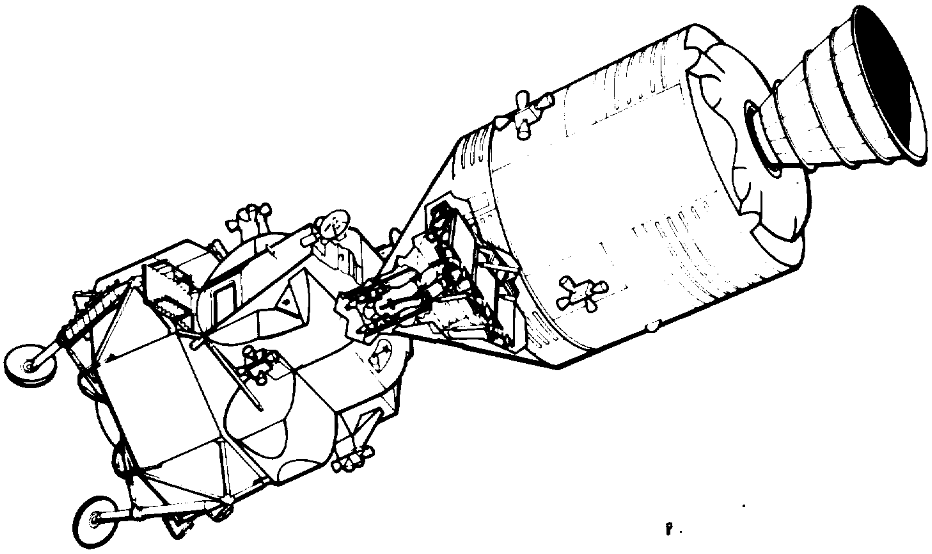


Figure 50

NASA S 66 6497 JUN

LM CONTROL PANEL

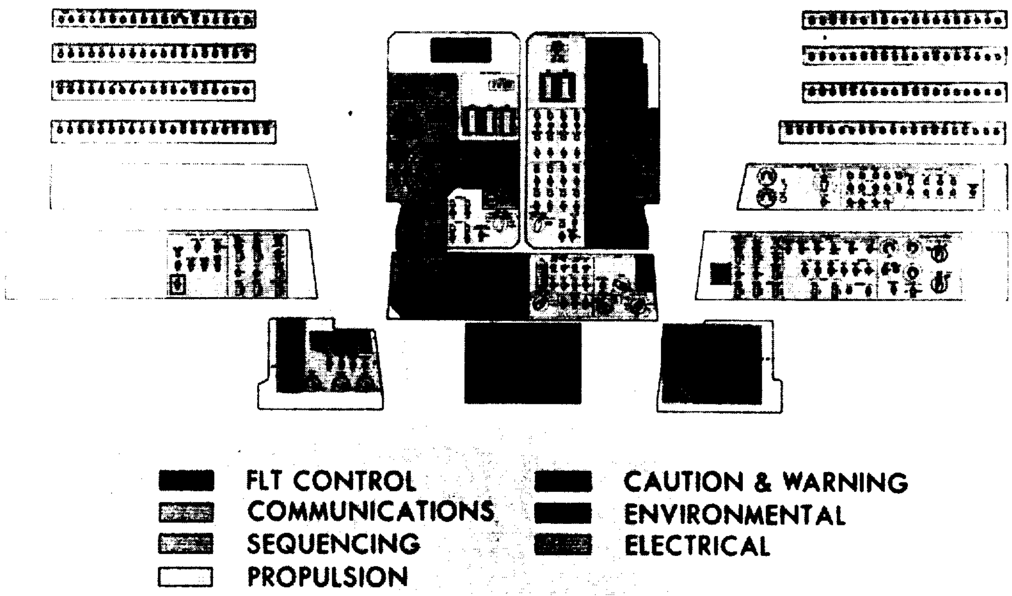

■ FLT CONTROL
▨ COMMUNICATIONS
▨ SEQUENCING
☐ PROPULSION
■ CAUTION & WARNING
■ ENVIRONMENTAL
▨ ELECTRICAL

Figure 51

NASA S-66-6140 JUN

LM FLIGHT CONFIGURATION

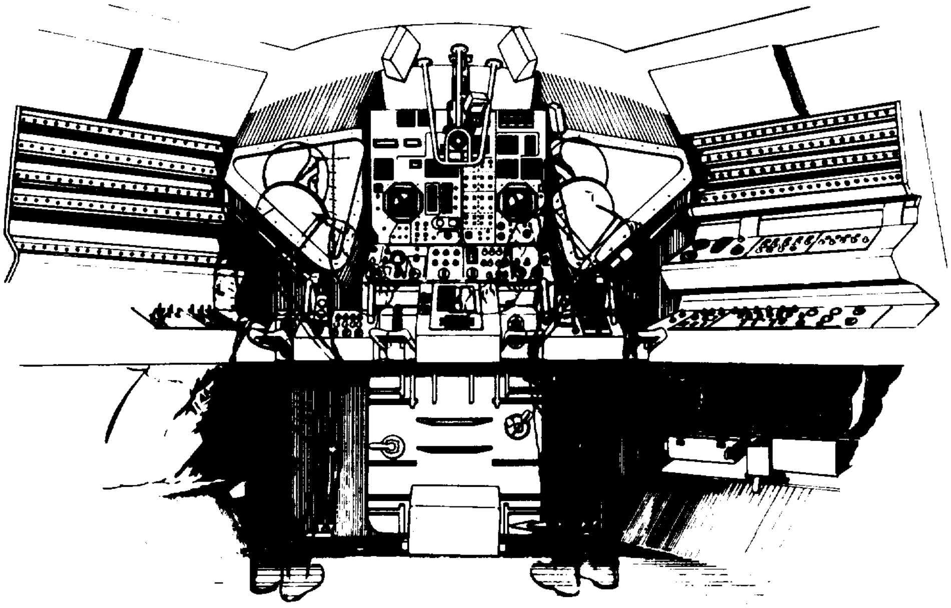


Figure [illegible]

NASA S 66 6822 JUN

DESCENT ENGINE CHECKOUT

1 DES REG 1 SW - "OPEN" - FLAG "GRAY"
2 DES REG 2 SW - "CLOSED" - "FLAG STRIPED"
3 HELIUM MONITOR SW "DES PRESS"
4 READ HELIUM INDICATION <1250 PSIA
5 PROPELLANT TEMP/PRESS SW TO "DES 1"
6 FUEL/OXID DES 1 TEMP - 70°F ± 20°F
7 FUEL/OXID DES 1 PRESS - 175 ±55 PSIA
8 PROPELLANT QTY MONITOR SW - "DES 1"
9 QTY GAGES AT OR ABOVE 95%

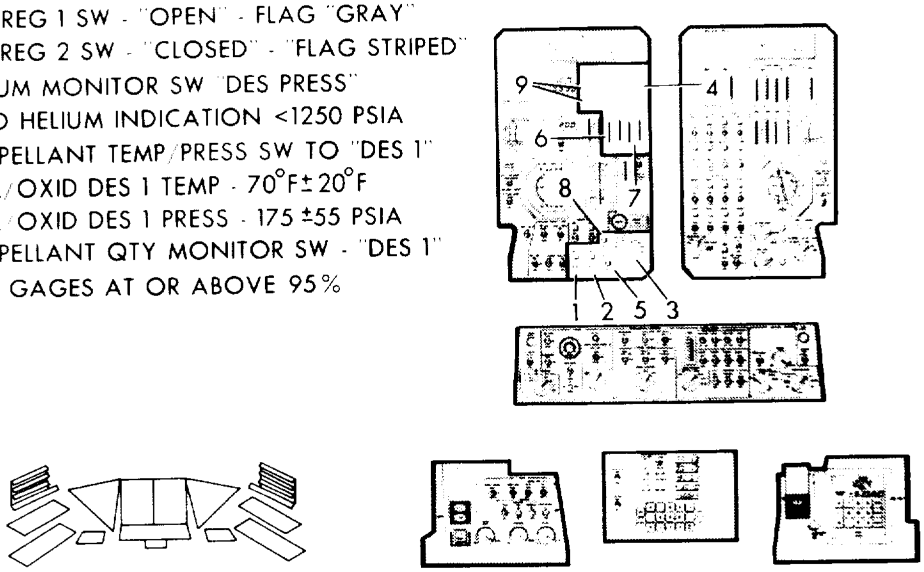


Figure [illegible]

NASA S 66 6484 JUN

LUNAR ORBIT INSERTION

- ORIENT CSM/LM FOR MANEUVER
- COUNTDOWN AND MONITOR THRUSTING
- ASSESS RESULTS OF THRUSTING
- ORIENT FOR ATTITUDE HOLD

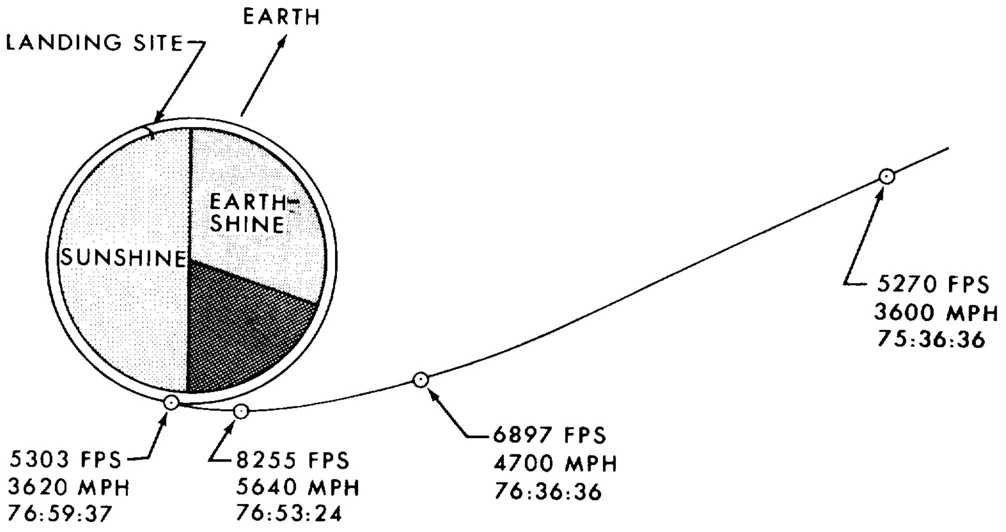
- ALIGN IMU FOR THRUSTING
- REALIGN IMU FOR LOCAL ATTITUDE HOLD

- PREPARE EPS FOR PEAK POWER
- MONITOR SPS PROPELLENT RATIOS & QUANTITIES
- CHARGE BATTERIES

Figure 54

NASA-S-66-6869 JUNE

LUNAR ORBIT INSERTION



LANDING SITE
EARTH
EARTH-SHINE
SUNSHINE
5270 FPS
3600 MPH
75:36:36
5303 FPS
3620 MPH
76:59:37
8255 FPS
5640 MPH
76:53:24
6897 FPS
4700 MPH
76:36:36

Figure 55

NASA-S-66-6593 JUN

APPARENT SIZE OF MOON

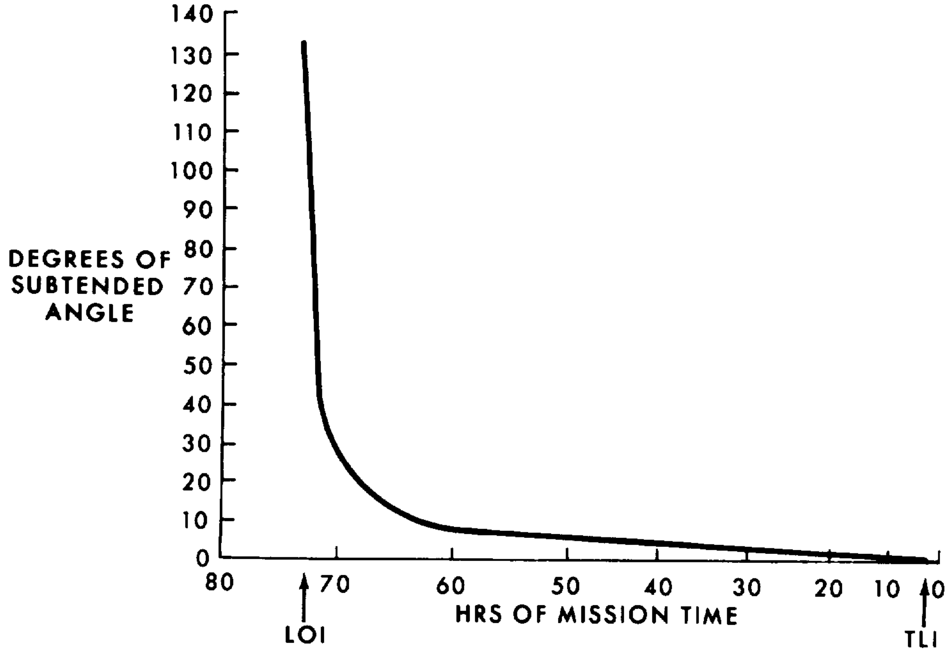


Figure 56

NASA-S-66-6146 JUNE

TRANSFER OF EQUIPMENT

(TRANSLUNAR)

TO LM

- 1 PLSS
- 2 EOS
- 1 PAIR EV GLOVES
- 1 TMG (LESS BOOTS)
- 1 RADIATION SURVEY METER
- 1 TV CAMERA
- 1 LM FLIGHT PLAN
- 1 EVCT DEVICE
- 2 EV VISORS
- 2 HELMET STORAGE CONTAINERS (CONTAINING VISORS)

TO CSM

- 2 LM ELECTRICAL UMBILICALS

Figure 57

NASA-S-66-6151 JUNE

TRANSFER OF EQUIPMENT

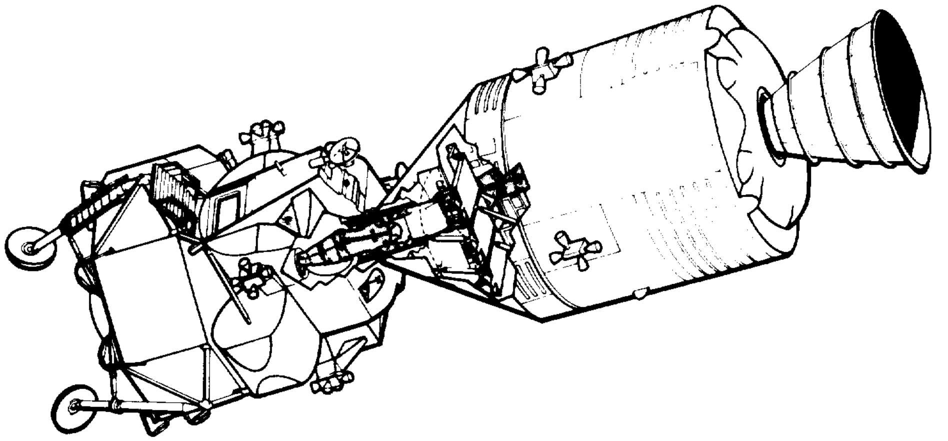


Figure 58

NASA-S-66-6447 JUN

LUNAR DESCENT

- MONITOR DESCENT ENGINE IGNITION AND ENGINE GIMBAL ALIGNMENT
- MONITOR PROPULSION QUANTITIES
- PREPARE FOR PGNCS/AGS DIFFERENCE CHECK
- COMPARE RANGE RATE WITH GUIDANCE BOUNDARIES AT CHECK TIME
- DETERMINE THAT LANDING RADAR DATA ARE AVAILABLE
- COMPARE LANDING RADAR ALTITUDE WITH GUIDANCE BOUNDARIES AT CHECK TIME
- CHECK STATUS OF RCS AND ASCENT PROPULSION PRIOR TO FINAL APPROACH
- MONITOR PROGRAMMED PITCH ATTITUDE CHANGE AT HI-GATE
- MONITOR THRUST INDICATOR FOR THRUST REDUCTION TO ABOUT 60 PC
- ACTIVATE DSKY DISPLAY OF LANDING AREA ELEVATION AND COMMUNICATE DISPLAYED VALUES
- EVALUATE LANDING AREA FOR ACCEPTANCE OR REJECTION
- NULL ALL RATES EXCEPT DESCENT FOR TOUCHDOWN

Figure 59

NASA-S-66-6140 JUN

LM FLIGHT CONFIGURATION

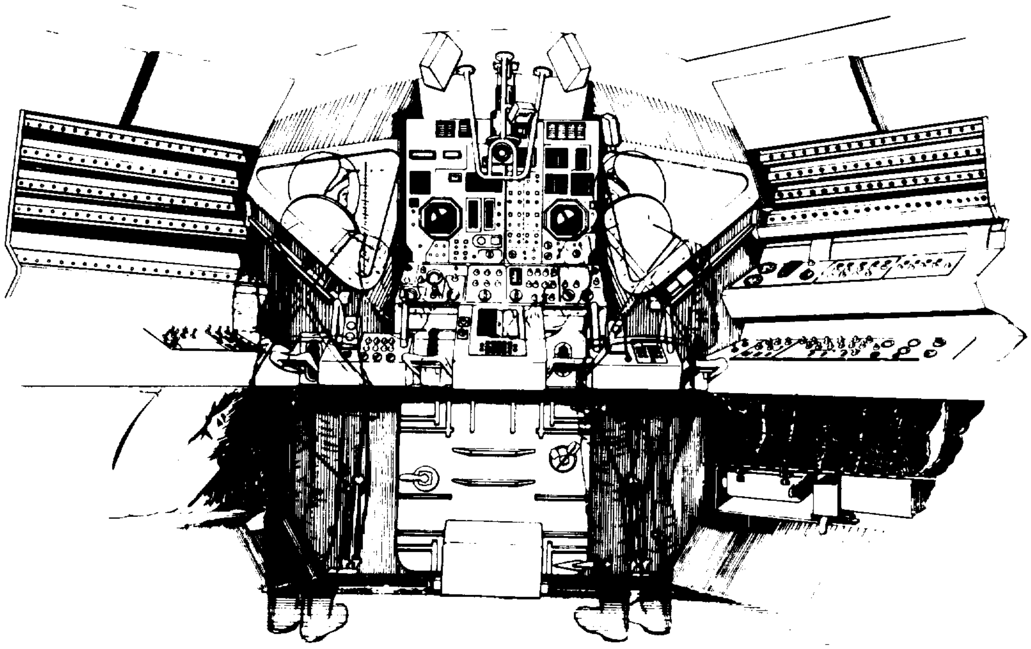


Figure 60

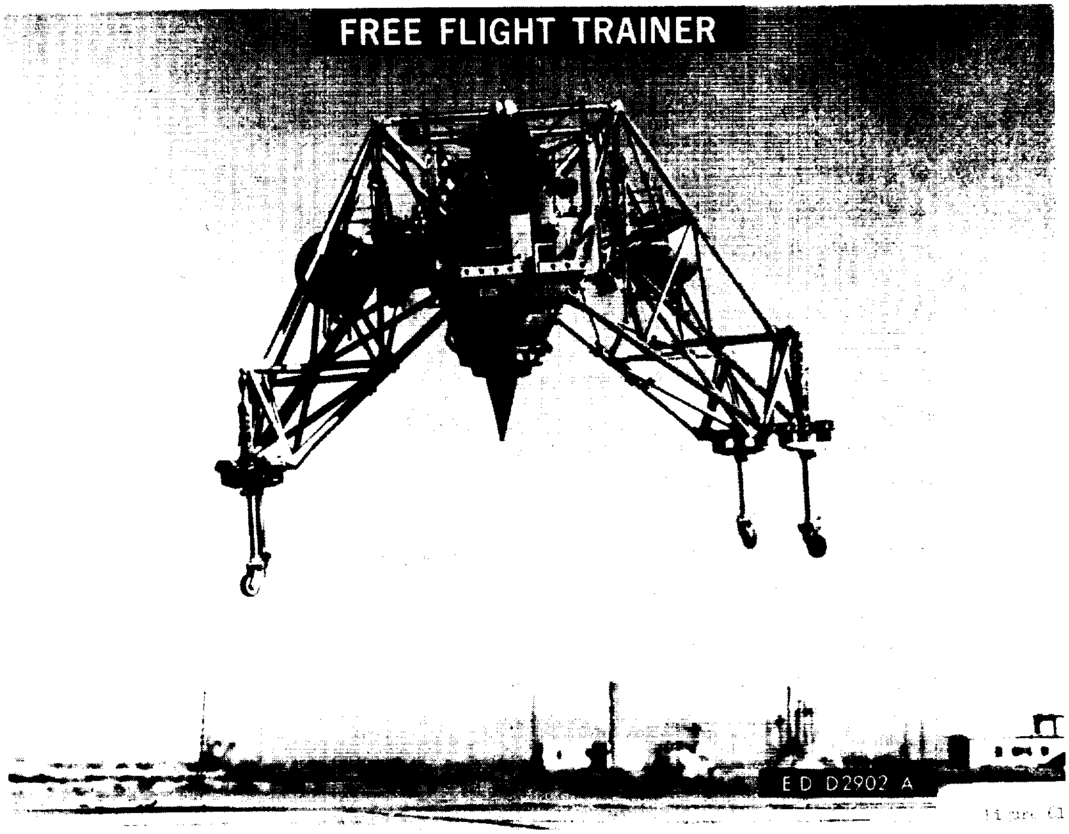

FREE FLIGHT TRAINER
E D D2902 A

Figure 61

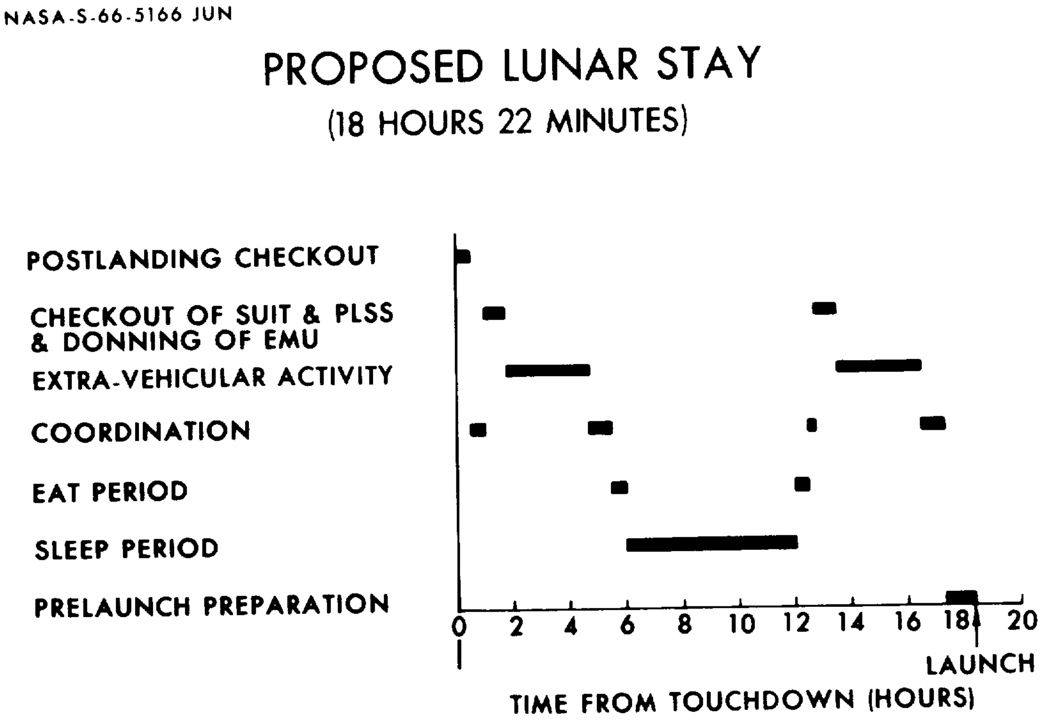


Figure 6

NASA-S-66-5073 JUN 3

LUNAR STAY

CREW ACTIVITY SUMMARY

ACTIVITY	HR	MIN	% TIME
POSTLANDING CHECKOUT		33	03
DON EMU & CHECKOUT	1	22	08
EXPLORATION	6	07	33
CABIN ACTIVITY	2	23	13
MONITOR SYSTEMS			
EVALUATE EXPLORATION			
COMMUNICATE W/MSFN			
RECHARGE PLSS			
UNSTOW & STOW EQUIPMENT			
PREPARE FOOD & EAT		55	05
SLEEP & REST	6	07	33
PRELAUNCH PREPARATIONS		55	05
TOTAL STAY TIME	18	22	100

Figure [illegible]

NASA-S-66-6143-JUN

PLSS DONNING

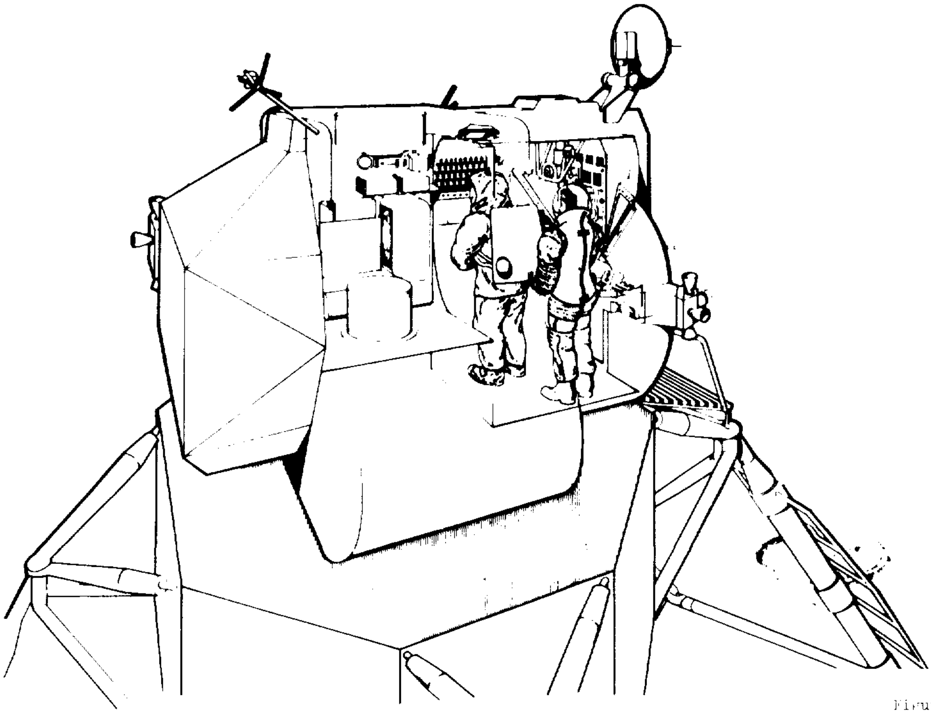


Figure [illegible]

NASA-S-66-6134-JUN

POST LANDING INSPECTION

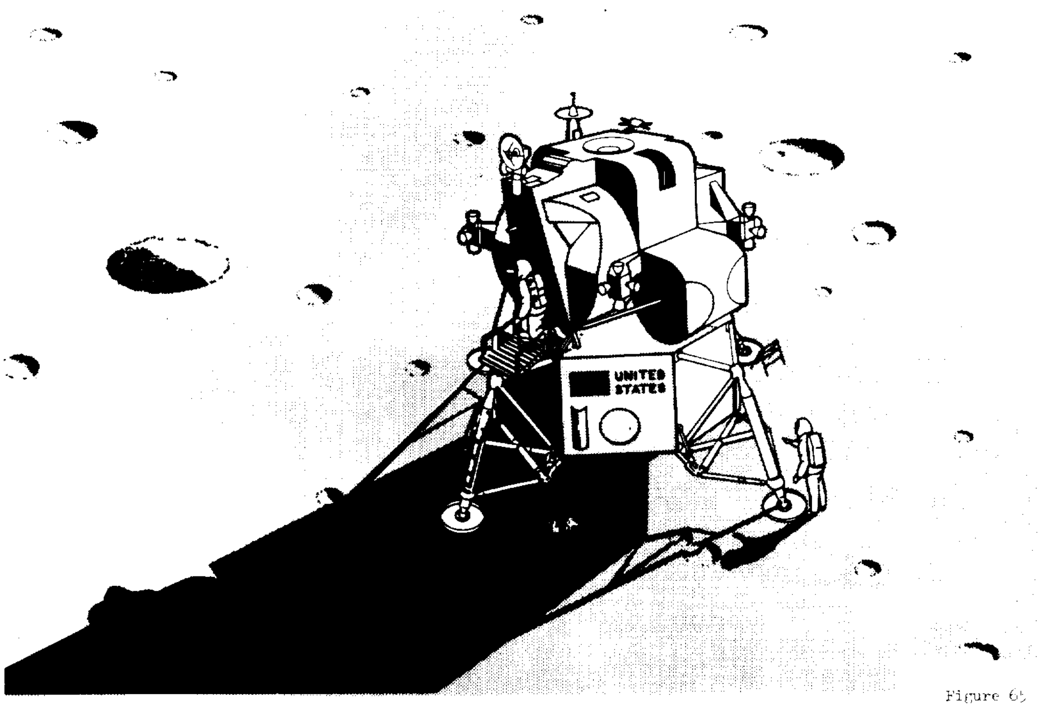

UNITED STATES

Figure 65

NASA-S-66-6141-JUN

LM SLEEP STATIONS

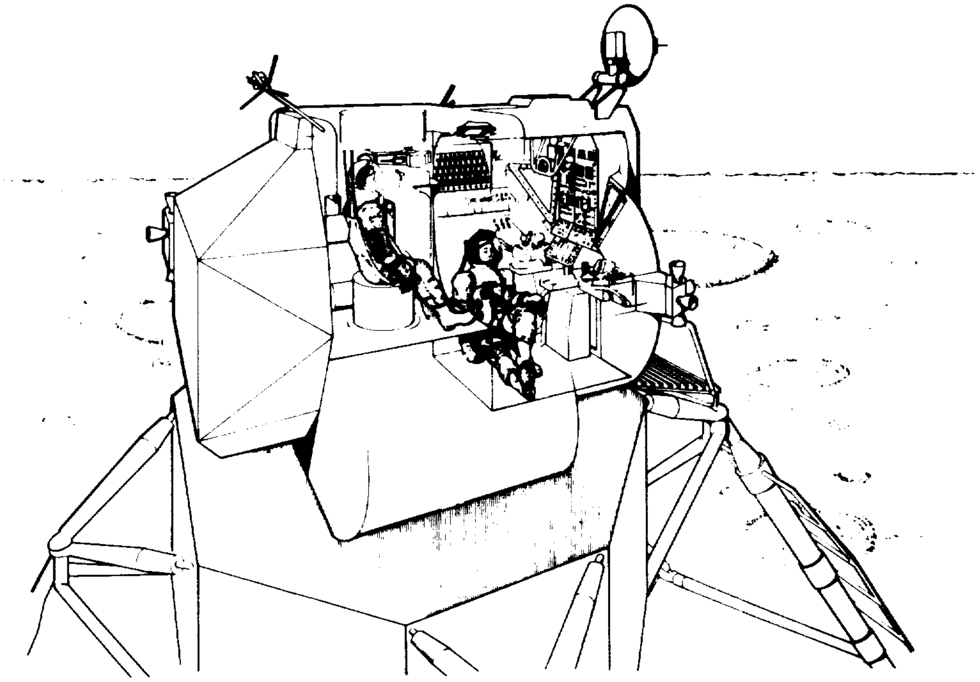


Figure 66

Figure 67

UNITED
STATES

Figure 68

UNITED
STATES

NASA-S-66-6145 JUNE

DISCARDED LUNAR SURFACE

- 5 LiOH CANISTERS*
- 4 PLSS BATTERIES
- 1 BAG PLSS CONDENSATE (TREATED)*
- 1 BAG URINE (TREATED)*
  SEALED FECAL CONTAINERS (TREATED)*
- 1 PLSS
- 1 STILL CAMERA
- 1 SEQUENCE CAMERA
- 2 PAIR EV BOOTS
- ? USED FOOD CONTAINERS

*STORED WITHIN DESCENT STAGE

Figure 7.

NASA-S-66-6595 JUN

LUNAR ASCENT

- COUNTDOWN TO LAUNCH WINDOW
- MONITOR ASCENT PROFILE AND VELOCITY
- CONFIRM ORBIT WITH MSFN

- ALIGN IMU
- VERIFY DATA WITH MSFN
- ACTIVATE RENDEZVOUS RADAR
- CLOSE RCS/ASCENT INTER-CONNECT

- ORIENT FOR RADAR TRANSPONDER AND LM OPTICAL TRACKING

Figure 73

NASA-S-S-66-6383 JUNE

AOT SIGHTING

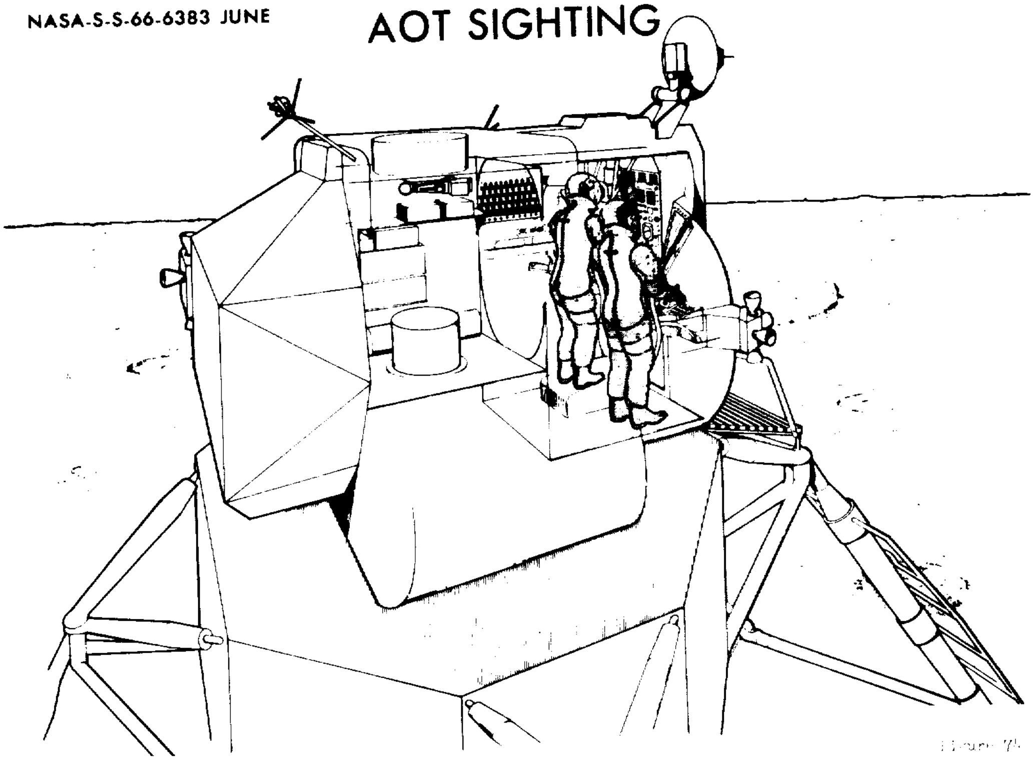


Figure 74

NASA-S-66-6591 JUN

RENDEZVOUS AND DOCKING

- CONFIRM TRAJECTORY PARAMETERS WITH CSM
- MONITOR RENDEZVOUS GATE MANEUVER BURNS
- MANEUVER TO DOCKING

- CONFIRM DELTA-V AND TIME TO GO WITH MSFN
- CHECK SUBSYSTEM STATUS

- ALIGN IMU
- ORIENT FOR ATTITUDE HOLD

Figure 75

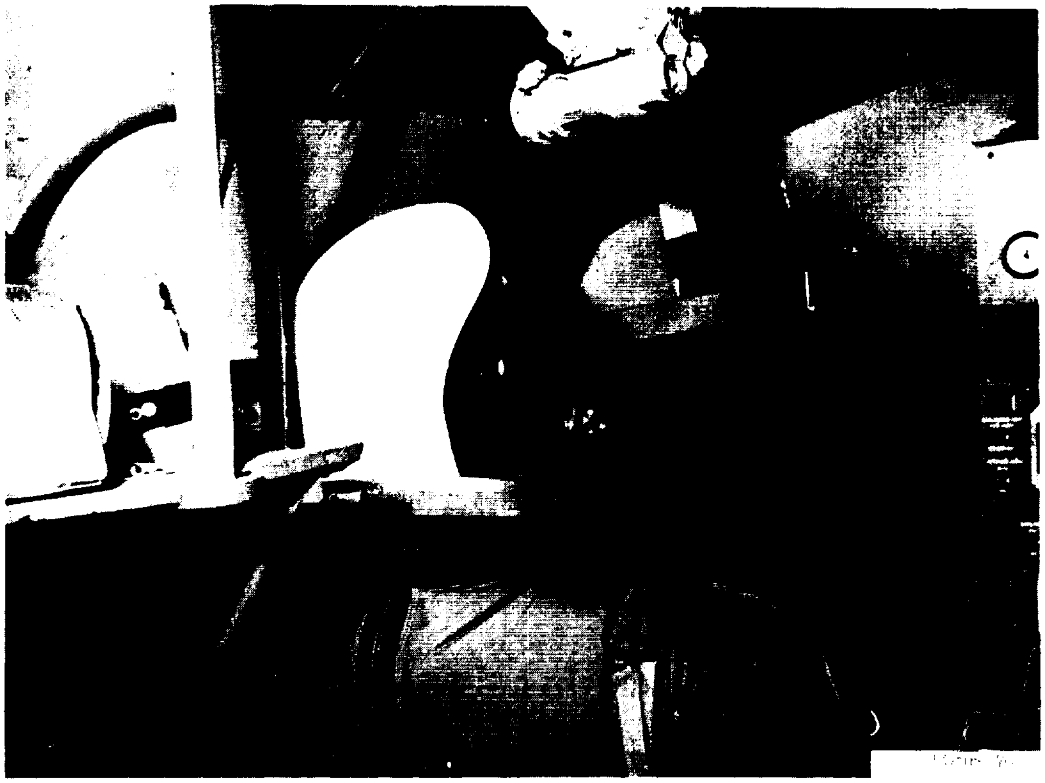


Figure 76

NASA-S-66-6142-JUN

ASCENT DOCKING

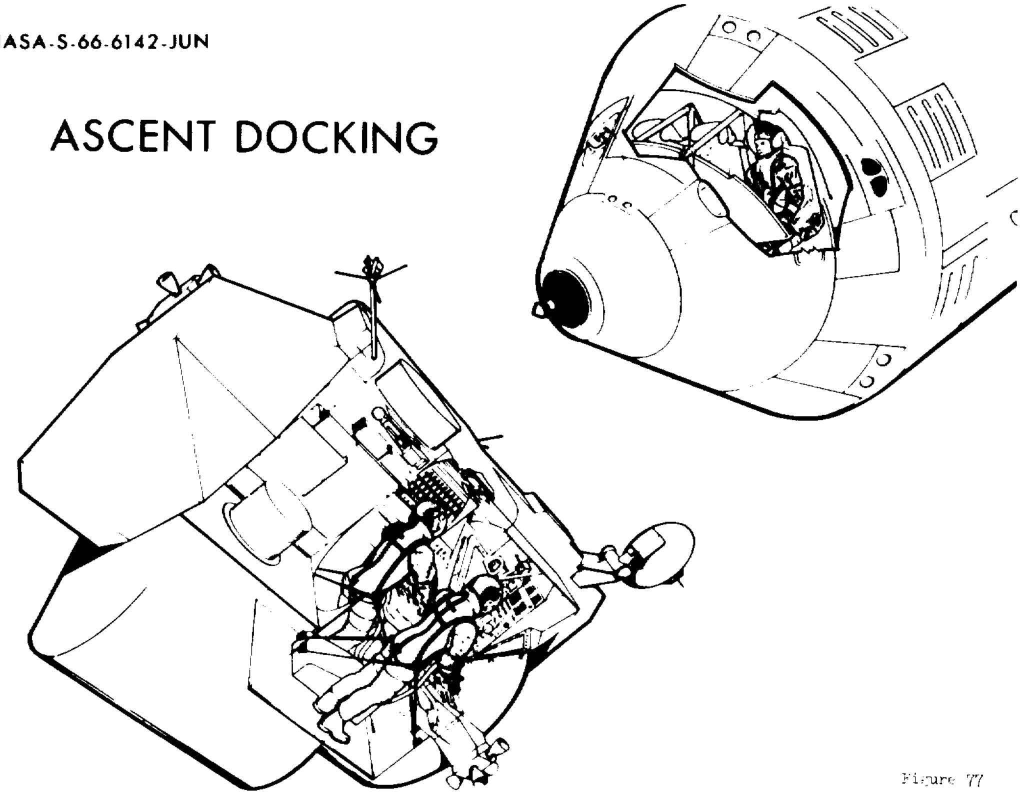


Figure 77

NASA-S-66-6144 JUNE

TRANSFER OF EQUIPMENT

(TRANSEARTH)

TO CSM

- 1 PLSS
- 1 PAIR EV GLOVES
- 1 TV CAMERA
- 1 LM FLIGHT PLAN
- 2 HELMET STORAGE CONTAINERS
- 2 SAMPLE RETURN CONTAINERS
- 1 SEQUENCE CAMERA FILM CONTAINER
- 1 FILM & TAPE CONTAINER
- 1 DATA STORAGE ELECTRONIC ASSEMBLY
- 1 RADIATION SURVEY METER

TO LM

- 2 EOS*
- 2 TMG*
- 1 PAIR EV GLOVES*
- 3 CWG
- 2 LIQUID COOLED GARMENTS
- 1 EVCT DEVICE*
- 1 DOCKING PROBE
- 1 DROGUE*
- 2 ELECTRICAL UMBILICALS
- 2 EV VISORS*
- USED FOOD CONTAINERS
- SEALED FECAL CONTAINERS

*LEFT IN LM UNLESS REQUIRED FOR EVT

Figure 78

NASA-S-66-6506 JUN

EVCT

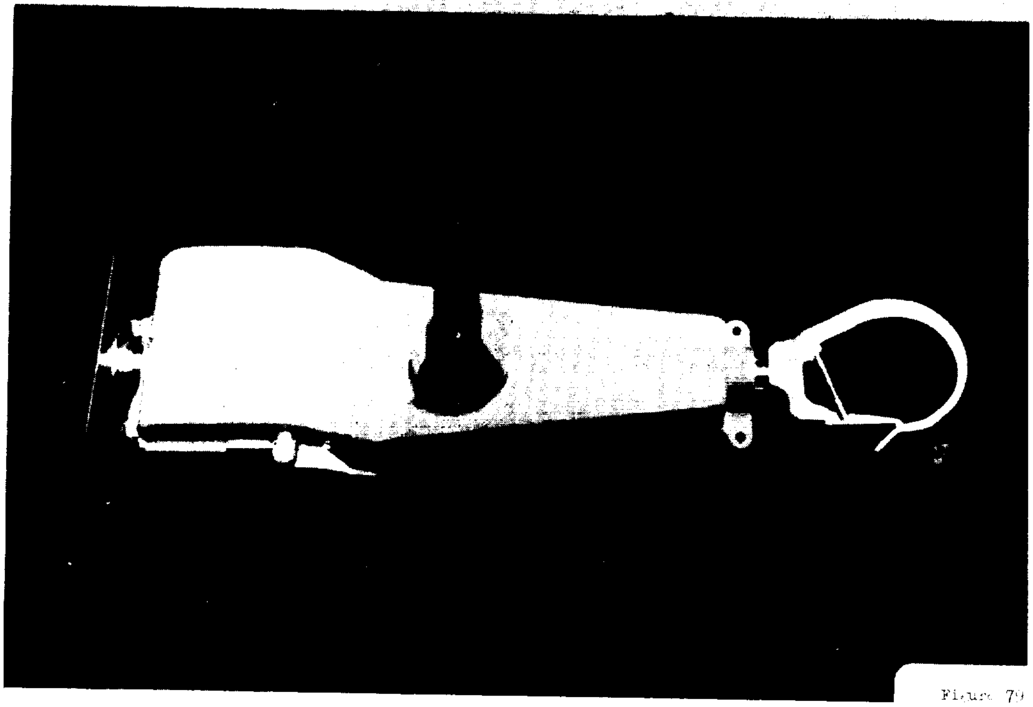


Figure 79

NASA-S-66-6130-JUN

CSM LUNAR ORBIT

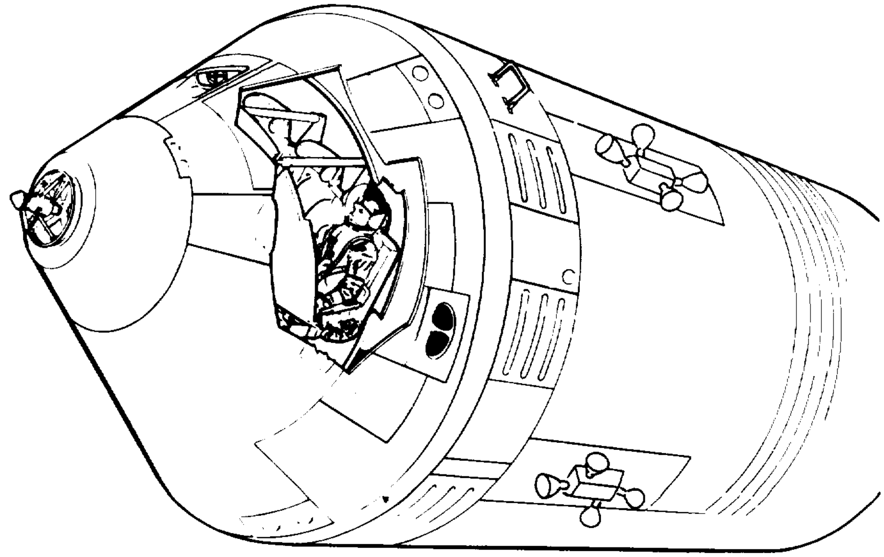


Figure 80

NASA-S-66-6594 JUN

TRANSEARTH

- MIDCOURSE CORRECTIONS
- PERIODIC SYSTEM CHECKS
- PERIODIC MAINTENANCE
- SLEEP

Figure 81

NASA-S-66-6860 JUN

ENTRY

- INITIATE SEPARATION
- USE ROTATIONAL CONTROL FOR LIFT VECTOR
- DUMP RCS PROPELLANT
- ENGAGE ELS

- CONFIRM SEQUENCING
- MONITOR ATTITUDE
- CONFIRM DESCENT WITH MSFN
- UNLOCK COUCHES

- ENTRY BATTERIES ON BUS
- DEACTIVATE FUEL CELLS
- MONITOR CHUTE DEPLOY

Figure 82

NASA-S-65-1414

TYPICAL ENTRY TRAJECTORY

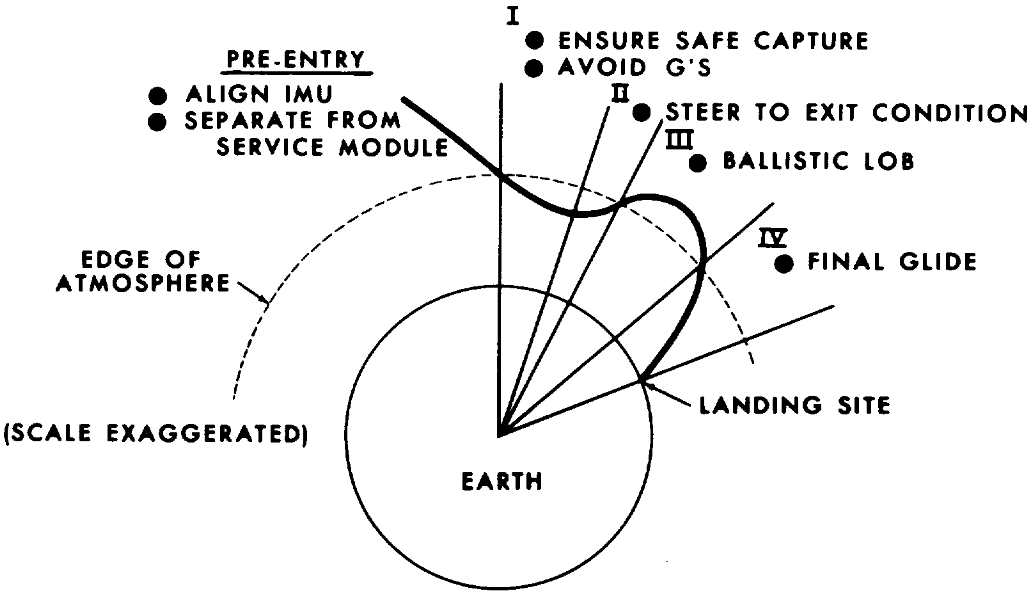

I
● ENSURE SAFE CAPTURE
● AVOID G'S
PRE-ENTRY
● ALIGN IMU
● SEPARATE FROM SERVICE MODULE
II
● STEER TO EXIT CONDITION
III
● BALLISTIC LOB
EDGE OF ATMOSPHERE
IV
● FINAL GLIDE
LANDING SITE
(SCALE EXAGGERATED)
EARTH

FIGURE 8

Figure 83

EXTRAVEHICULAR MOBILITY UNIT OPERATIONS

by

William C. Kincaide

EXTRAVEHICULAR MOBILITY UNIT OPERATIONS

The Extravehicular Mobility Unit (EMU) is in reality a spacecraft module in itself, operating completely independent of the other modules while on the lunar surface. The EMU has its own communications, electrical power, and environmental control system. Its guidance and navigation, propulsion, and reaction control systems are the astronaut working in conjunction with the EMU.

The EMU consists of seven major subsystems (figure 1): a liquid cooled undergarment, a pressure garment assembly, a portable life support system, a thermal-meteoroid garment, lunar boots, gloves, and protective over visor.

In addition to the lunar exploration capabilities, the EMU will allow the crew to perform free space extravehicular transfer from the Command Module to the Lunar Module or vice versa in event of a docking system malfunction. And as in both Projects Mercury and Gemini, the space suit provides a backup for unscheduled loss of cabin pressure. In the event of a noncatastrophic cabin depressurization, the suit is capable of being donned prior to the cabin reaching an unsafe pressure level. This capability allows the crew to remove the suits for a good share of the mission if they so choose.

Specific EMU requirements for the lunar mission are: To allow explorations of the lunar surface for distances of at least one-half nautical mile; provide up to four hours of continuous separation time from the spacecraft for each excursion and a total capability of 24 hours by multiple excursions; permit two crewmen to be extravehicular simultaneously; and permit recharge of the EMU within one hour for rapid turn around. The four hour separation period has been further defined to be three hours of nominal mission and one hour contingency. The twenty-four hour total exploration time is attained by eight excursions of four per EMU.

The design requirements of the portable life support system, PLSS, are shown in figure 4. The main function of the PLSS is to control and replenish the atmosphere within the space suit. The Unit, which weighs 65 pounds fully charged, will provide oxygen and CO_2 control for respiration and cooling for average work rates up to 1600 Btu per hour. This is similar to a man walking at 4 to 5 miles per hour here on earth. The total capacity for cooling is 4800 Btu. The unit is also designed to accommodate high work rates of up to 2000 Btu per hour for periods of up to ten minutes. In actuality, tests have shown that the unit will handle short peaks of 3000 to 4000 Btu per hour, however, expendables will be consumed at an accelerated rate, thereby shortening the mission time available. The PLSS also has the capacity to handle external heat leaks into the system of up to 250 Btu per hour.

The basic system schematic is shown in figure 5. Thermal control is achieved through a unique approach which does not rely upon crew sweating as the primary means of cooling as in the present conventional spacecraft environmental control systems. This is accomplished by circulating cool water at a rate of four pounds per minute from the PLSS through tubes in direct contact with the skin so that metabolic heat is conducted away. Thus, the problems of astronaut dehydration, sweat in the eyes, skin maceration, etc., are minimized.

The basic gas ventilation loop is still required to replenish oxygen, remove carbon dioxide and moisture, and control total pressure. However, flow rates are reduced to a total flow of 6 cfm to the helmet, as compared to 15 to 20 cubic feet per minute which would be required without water cooling.

A typical maximum integrated metabolic profile is shown in figure 6. Profiles like this are being used primarily to test the PLSS, however, they are based upon a reasonable estimate of lunar excursion. This profile is based upon a three hour excursion at an average metabolic rate of 1600 Btu's per hour, which is the maximum design point for the PLSS. Periods of heavy work such as egress from the spacecraft, and walking are assumed to require about 2000 Btu's per hour. Moderate tasks such as setting up experiments rates will require around 1600 Btu's per hour.

Obviously, an actual metabolic profile for a lunar excursion will be a series of sharp peaks and valleys, many of which will exceed 2000 Btu's per hour for short periods.

The EMU will maintain the astronaut in a thermally comfortable condition, however, at these sustained high metabolic rates he will undoubtedly become tired and may require periodic rest periods.

The PLSS also houses a redundant two-way simultaneous voice communications unit and a seven channel telemetry unit for system and biomedical data transmission. The data which will be available is shown in figure 7. Information available to earth will include: Total suit pressure, status of PLSS consumables, such as cooling water quantity, primary oxygen remaining, battery use rate, thermal performance of the PLSS, and electrocardiogram. From this data, it will be possible to assess the metabolic rates associated with lunar tasks.

The astronaut will be able to monitor his suit pressure, coolant water quantity remaining, and primary oxygen quantity. Audible warning tones will also alert the astronaut to low suit pressure and high oxygen use rates.

One of the most difficult EMU design problems to date has been the location of PLSS controls, figure 8. It is difficult to make switches and valves readily accessible without degrading mobility.

Many approaches have been evaluated from over the shoulder cables to side arm extensions. The concept finally chosen was a behind the back-fingertip operation as being the least complex insofar as PLSS design and operational usage. Electrical and communications switches are located on the left lower corner of the PLSS. Manual water and oxygen valves are located on the other corner. As seen in figure 8, the corners are easily accessible and, with training, the controls can be operated quickly and accurately. All controls will be operated at start-up; however, only the water garment inlet temperature control and communications switch require operation on the lunar surface.

An independent emergency oxygen system is also available to the astronaut. This 3.5 pound unit contains 0.2 of a pound of oxygen stored at 7500 psi. The system is actuated by pulling a "green apple" which provides regulated oxygen directly to the helmet. The emergency system will provide from 5 to 38 minutes of additional time, depending upon the point in time at which the failure occurs. The system is not rechargeable. Figure 9 typifies the progress made in the EMU program toward compacting hardware. The system on the left is the emergency oxygen system as of about two years ago, which weighed over 5.5 pounds. The volume improvement speaks for itself.

The pressure garment assembly, shown in figure 11, consists of a basic torso enclosure, intravehicular helmet, boots, and gloves. The suit provides pressure protection with sufficient mobility to accomplish the lunar mission. Unlike its predecessors, the Apollo suit uses convoluted bellows to allow flexure of joints by maintaining a constant volume in the suit thus reducing the effort required to compress the gas. Joints are located at the shoulders, elbows, wrists, thighs, knees, and ankles. Bearings are located in the upper arms and wrists to allow the arm and hand to be rotated. Entry into the suit is made through a dual zippered opening in the back identical to the Gemini suit. The Gemini zipper has proved to be exceptionally reliable by comparison to past closure designs. In fact, Gemini space suit technology, operational experience and components are being used to fullest extent possible in the Apollo program.

Figures 12 through 14 depict some of the basic movements that can be performed with the Apollo suit. The lunar explorer will be able to walk, kneel, crawl on all fours, get up from a prone position either from the side, front, or back, bend down at the waist, squat and reach most frontal and side areas of his suit.

A major factor contributing to the capabilities of the suit is familiarization and training on the part of the wearer. For example, many arm positions are more easily attained by a combination of motions rather than a direct movement to that position. Proper use of the helmet tie down can make sitting and other similar operations which require bending at the waist much easier.

The helmet has a fixed visor with no neck bearing, which allows the wearer to turn his head without having to move the helmet with him. This gives the crewman a greater field of vision and eliminates the torque required to move a neck bearing. Downward vision is particularly important to the lunar mission since the astronaut must be able to see his feet to be able to select each step in rough terrain. Also, the crewman will be required to make and break his gas ventilation connections within two minutes to transfer from vehicle to PLSS operation. The helmet is capable of being quickly donned without assistance, over a communications cap. The cap is also worn for "shirt sleeve" communications.

Visor fogging, which occurred during the GT-9 Gemini extravehicular experiment, has been and will continue to be given a great deal of consideration in the EMU test program.

Fogging is, of course, caused by condensation of the warm wet expired breath and ventilation gas on the relatively cold inner surface of the visor. Manned tests on an EMU at an altitude of 300,000 feet, with -300°F cold walls simulating lunar night conditions, showed that satisfactory clearing of the visor occurred at metabolic rates up to 2000 Btu per hour. The test subject could force a small patch of fogging by purposely blowing on the visor, however, the clouded area cleared itself within one to two seconds. Visor inside surface temperature was only slightly lower than the temperature of the ventilation flow, in spite of an outside surface temperature of 20°F.. Low emissivity coatings will be provided on the visors to control visor temperatures and a wetting agent coating for the internal visor is being investigated to completely preclude fogging.

The Apollo visor is made of polycarbonate which was selected primaril for its impact strength. Quarter inch polycarbonate will withstand impact loads of 80 foot pounds or more depending upon the area and velocity of the object striking the visor. This energy level is very important in Apollo where the astronauts will be moving about on the lunar surface and inside the spacecraft exposing the helmet to knocks on bulkheads, brackets, etc. For comparison, Mercury and early Gemini visors were made of acrylic or plexiglass which break at energy levels of under 5 foot pounds. This was satisfactory because the crew did not leave the couch.

Tests were conducted in the 1/6 gravity aircraft with a mockup fiberglass visor, to determine the energy loads that could be expected for a fall on the lunar surface (figure 15). We would certainly expect this to be the worst case. During this series, energy levels of 135 foot pounds were recorded. The polycarbonate helmet and over visor combination will accommodate this type of blow without injuring the astronaut through its ability to deform and absorb energy. The helmet has been impacted with 130 foot pounds by dropping a 16 pound rod with a 2-inch diameter hemispherical tip without rupturing. Polycarbonates are relatively new and production process controls and fabrication techniques are not as yet quite developed to the point where a quality helmet visor with acceptable distortions and surface imperfections can be produced consistently. While this is still an existing problem several contractors are investigating forming processes and it is felt that an acceptable helmet will be qualified by January 1967.

Provisions are made to store body wastes within the suit. The urine collection bag is identical to the Gemini device, except for minor interface revisions. It can also be emptied in flight and reused if necessary. Fecal collection is not expected to be required during the planned pressurized suit modes, however, during an emergency return which could take up to 115 hours, provisions must be made to contain the feces. Numerous types of systems have been examined, however, to date none have been totally acceptable.

The liquid cooled garment (figure 16) is worn under the space suit, next to the skin. Unlike regular underwear the garment has 300 feet of polyvinyl chloride tubing sewn into it which directly contacts the skin of the wearer. The PLSS circulates cool water through the tubes in forty parallel paths to reduce pressure drop. An astronaut will have a choice of inlet temperature settings, from 45° to 85° F, depending upon his activity level. During a three hour manned mission profile in which metabolic load varied from 400 Btu/hr to 2000 Btu/hr for varying lengths of time, test subject comfort was maintained with only six changes in position of the control valve. The subject's skin temperature was maintained at a comfortable level while the average sweat rate was only 71.7 cc/hr for the test. The testing to date on this garment has been extensive. Subjects have worn the garment for several days without significant comfort problems.

The thermal-meteoroid garment (figure 18) is worn over the entire pressure garment to protect the space suit from cuts, abrasions, and meteoroid penetrations, and to provide passive thermal control. The garment is made up of an outer layer of reflective white dacron, seven layers of super insulation, and finally two layers of neoprene coated nylon. Insulation to limit conduction also has been added in areas which will routinely come into contact with hot or cold surfaces such as the hands, knees, and feet.

Mobility is not appreciably degraded with the addition of these layers over the suit, except for the gloves and boots. The thermal gloves have a fine woven stainless steel palm for durability backed by 16 layers of super insulation for conductive protection. The back side of the glove is identical to the thermal garment cross section. Flexibility is hampered by the relatively small diameter fingers compared to the material bulk and tactility is very difficult to retain, especially at the fingertips where the material comes together. This glove has recently been tested under vacuum conditions, in which a subject was able to grip a hot rod at 250° F for three minutes and a cold rod at -250° F for over thirteen minutes.

The lunar boot is not as problematic as the glove because gripping and tactility are not required, however, a flexible sole is desirable to avoid having to walk flat footed. The soles of the lunar boot have an outer surface of silicone rubber which holds up surprisingly well on sharp rough surfaces. The rubber is followed by 13 layers of super insulation and finally two layers of nomex felt. A current boot design has been successfully tested under the vacuum conditions in a hot plate at 250° F for over one hour.

Unmanned testing of thermal garments has been underway since early 1963, in an effort to obtain adequate protection without severely impairing mobility. The garment has been optimized from a 25 layer cross section to its present seven layers. Access flaps have been purposely left open during these tests to evaluate potential hot spot problems. The outer surface of the garment was purposely dusted with soil of the same absorptivity-reflectivity characteristics as the lunar surface to examine the thermal implications. The test data and computer analysis agreed that the heat flux in could increase by a factor of one. However, this flux is still within the capacity of the PLSS. Even with the minor GT-9 problem, the two Gemini EVA flights and ground tests have validated this concept of thermal protection.

A prototype extravehicular visor assembly, figure 22, will be worn over the primary helmet. The attachment concept is identical to that shown, however, the visor attenuation approach has been modified to the GT-4 concept. Two visors are provided which rely upon reflectance rather than absorption to attenuate infrared and visible light.

The test set-up for unmanned testing of the thermal-meteoroid garment in the Ling-Temco-Vought space simulator is shown in figure 23. In this series a quartz lamp cage was placed around the test specimen, in conjunction with the solar source and cold walls, to simulate the infrared heating expected from the lunar surface. The dummy is rotated about its longitudinal axis to examine transient situations and different view factor situations.

The worst thermal case tested to date simulated an explorer standing in a lunar plane with a sun at 90° from the vertical. The total heat leak reached about 135 Btu/hr. Further unmanned tests are planned to examine other critical thermal situations, such as the EMU working in a crater.

Tests have shown that the Apollo Block II pressure garment assembly is the most mobile soft space suit yet developed. The metabolic energy required to exercise the pressurized suit is small by comparison to the Mercury and Gemini suits and its range of motion significantly greater. Of course, neither of its predecessors were designed for walking, however, even in areas of comparable motion requirements, the Apollo suit is superior.

While the Apollo suit is the best available, it still requires a good deal of metabolic energy to operate. Figure 24 summarizes some of the results of manned treadmill tests conducted to date, with unpressurized and pressurized suits at both 1 g and 1/6 g. It appears that the metabolic expenditure for walking in the pressurized suit will require up to twice as much energy as that of the unpressurized suit in one g. This ground based data also indicates that metabolic expenditure for walking will be reduced up to 50%, due to the 1/6 gravity environment. From this data, we can expect that the astronaut should be able to travel up 2 or 3 miles per hour without any thermal stress depending upon the immediate terrain. To date, metabolic data has been collected primarily by exercising subjects on treadmills because it offers a convenient standard for comparison. More tests will be conducted, in fact this type of testing will be intensified to measure the metabolic penalty associated with a variety of surface conditions.

The MSC lunar crater area has been used periodically to evaluate, subjectively, the problems of terrain and 1/6 gravity as shown in figures 25 and 26. The 1/6 gravity simulator creates some problems itself because of the effort required by the subject to overcome its inertia. However, these tests have proved quite valuable in assessing the balance and stability problems.

Field trips have been made to sites in Oregon and Arizona where the terrain is believed to resemble that of the lunar surface. Subjects wearing full EMU's performed simulated traverses in deep sand and in rocky areas, up a variety of slopes and on flat surfaces, as seen in figures 27 and 28. Various tools, experiments, and walking aids were evaluated in conjunction with the EMU. Field simulation of this sort will continue throughout the program becoming more sophisticated as production hardware and detail on the lunar surface is available.

It has been very apparent from these field trips and the 1/6 gravity simulations in the MSC lunar crater that balance will be difficult to maintain and in fact may well be the constraint in determining the range of expolration. Subjects tended to walk carefully in these tests rather than bounding along, primarily because they have a great deal of difficulty in maintaining their balance and stopping quickly. They could make better time by loping along; however, because of these control problems, would probably only attempt this in an emergency.

EMU failure modes which constrain the lunar mission are primarily directly associated with expendables and therefore determine the time the crew has to return to the spacecraft. For example, one of the more critical failures is a fan malfunction which would require that the astronaut return to the spacecraft immediately. The remedial action for this case would be for the crewman to activate the emergency oxygen system and open a fixed orifice in the suit which causes a two pound per hour or 1.9 cfm in the helmet as the pressure regulators attempt to maintain the suit pressure. The resultant mission constraint is shown in figure 29. As the astronaut leaves the vehicle, he can walk up to a maximum of 26 minutes away before he has to start working his way back. Assuming nominal usage to this point, the amount of oxygen left in his tanks will just give him 26 minutes at 2 pounds per hour. After 26 minutes he will have to work his way back, always staying within this envelope, which is a function of the use rate, i.e., metabolic consumption and leakage. If he works harder or has a greater leakage to contend with, the maximum envelope is reduced. The specific leak rate for the EMU is 200 standard cubic centimeters per minute. A leakage check will be performed prior to egress however the check will be made by measuring pressure decay on a small mounted gage. The test, as such, will be relatively gross, assuring the crewman that all connections are properly made. Oxygen quantity trend information will be calculated on the ground to keep the astronaut informed as to his return status.

The EMU subsystems have completed critical design review. First article configuration inspections will be held during July 1966. In August, a subject wearing the full EMU will walk into a space simulator. This test will be the first real manned thermal test in a simulated environment, completely protected by the EMU subsystems.

Based upon the tests to date, we are confident that this unit will be adequate to perform a valuable lunar mission. When the program was initiated, the design requirements were based upon a relatively poor estimate of the lunar environment. The data which has been acquired since this time from pegases, surveyor, etc., has shown the original estimate to be conservative and as such adds to the confidence in the hardware.

Questions and Answers

EXTRAVEHICULAR MOBILITY UNIT OPERATIONS

Speaker: William C. Kincaide

1. Can the extravehicular astronaut walk sideways?

   ANSWER - Yes.

2. Why is the helmet red?

   ANSWER - Allows better observation of the EVA.

3. Which visor will be down during EVA?

   ANSWER - Normally the inner visor for night-time excursions and the gold visor for daytime excursions.

NASA-S-66-5649 JUN 14

APOLLO EMU

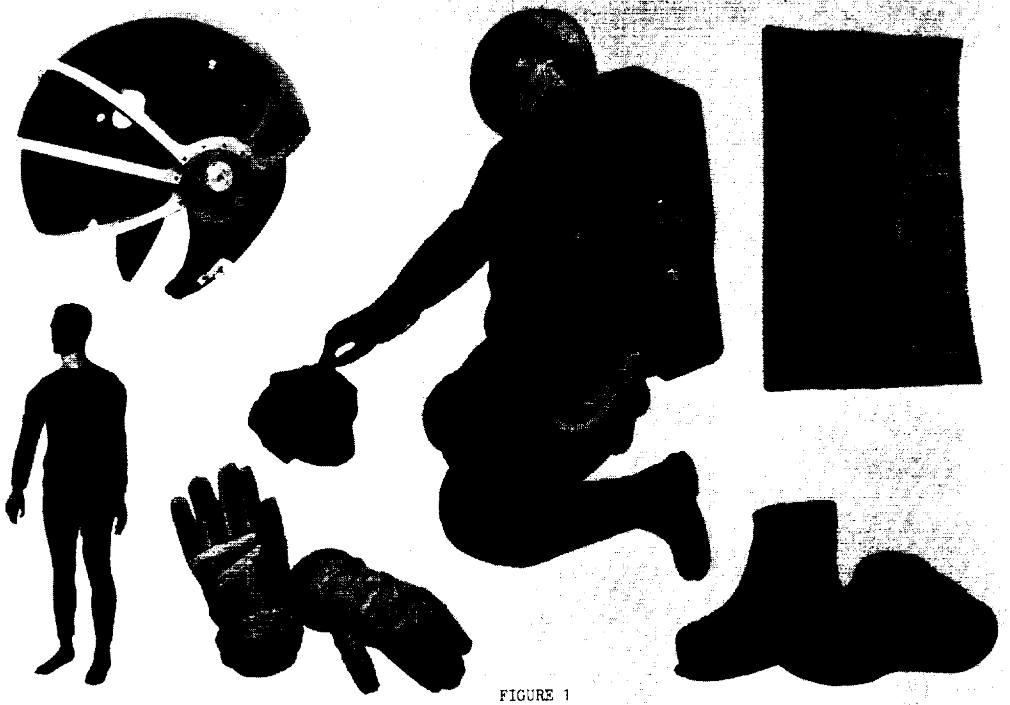


FIGURE 1

NASA-S-66-6854 JUN

EMU MISSION REQUIREMENTS

- PERMIT APOLLO CREWMEN TO LEAVE THE CM OR LM IN FREE SPACE TO ACCOMPLISH EXTRAVEHICULAR TRANSFER
- PERMIT APOLLO CREWMAN TO LEAVE THE LM ON THE LUNAR SURFACE TO EXPLORE & RETURN SAFELY TO THE LM
- PROVIDE BACKUP PROTECTION WITHIN THE SPACECRAFT IN THE EVENT OF UNSCHEDULED CABIN PRESSURE LOSS

FIGURE 2

NASA-S-66-6852 JUN

SPECIFIC LUNAR MISSION REQUIREMENTS

- ALLOW EXPLORATIONS OF THE LUNAR SURFACE FOR DISTANCES OF AT LEAST 0.5 NM FROM THE LM
- PROVIDE FOR 4 HOURS OF CONTINUOUS SEPARATION FROM THE SPACECRAFT FOR EACH EXCURSION & A TOTAL OF 24 HOURS
- PERMIT TWO CREWMEN TO BE EXTRAVEHICULAR AT THE SAME TIME
- PERMIT RECHARGE OF THE EMU WITHIN AN HOUR

FIGURE 3

NASA-S-66-6851 JUN

PORTABLE LIFE SUPPORT SYSTEM (PLSS) DESIGN REQUIREMENTS

• THERMAL CAPACITY	
• METABOLIC	4800 BTU'S TOTAL 1200-1600 BTU/HR AVERAGE RATES 2000 BTU/HR PEAKS
• EXTERNAL LEAKAGE	250 BTU/HR IN 350 BTU/HR OUT
• PRESSURE	3.8 PSIA NOMINAL 3.2 PSIA MINIMUM (EMERGENCY)
• CARBON DIOXIDE	7.5 MM Hg NOMINAL 15 MM Hg MAXIMUM (CONTINGENCY)
• COMMUNICATIONS - TELEMETRY	REDUNDANT 2 WAY SIMULTANEUOUS VOICE 7 CHANNELS OF TELEMETRY

FIGURE 4

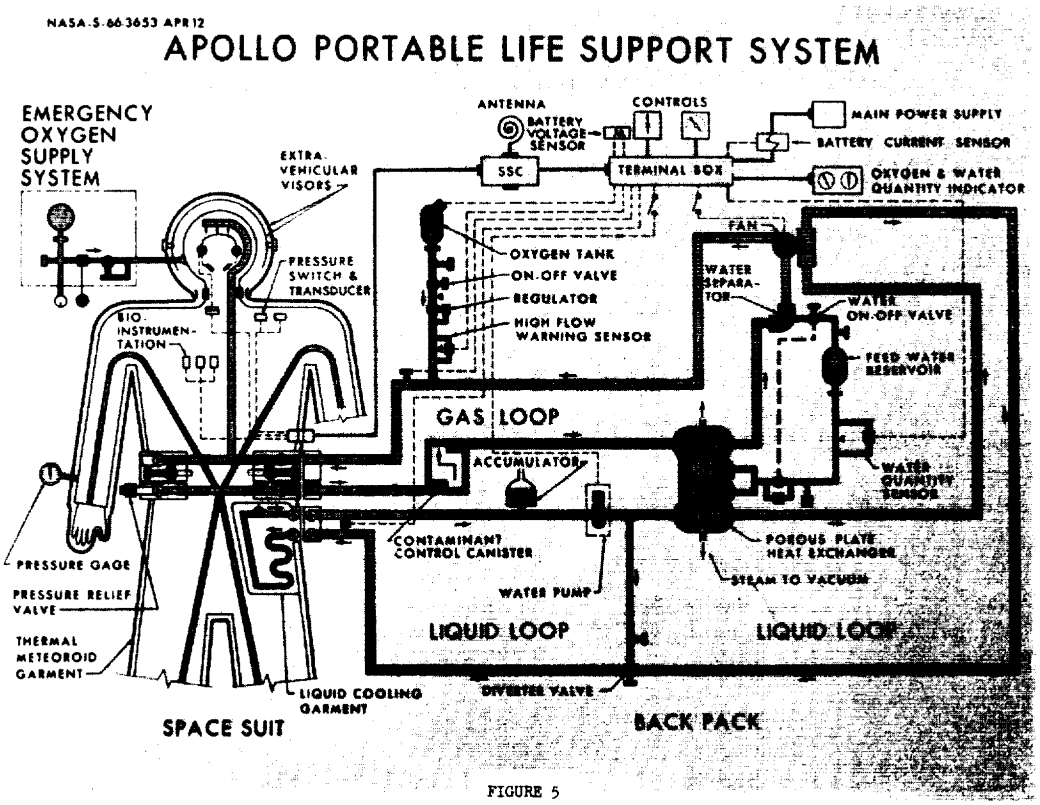
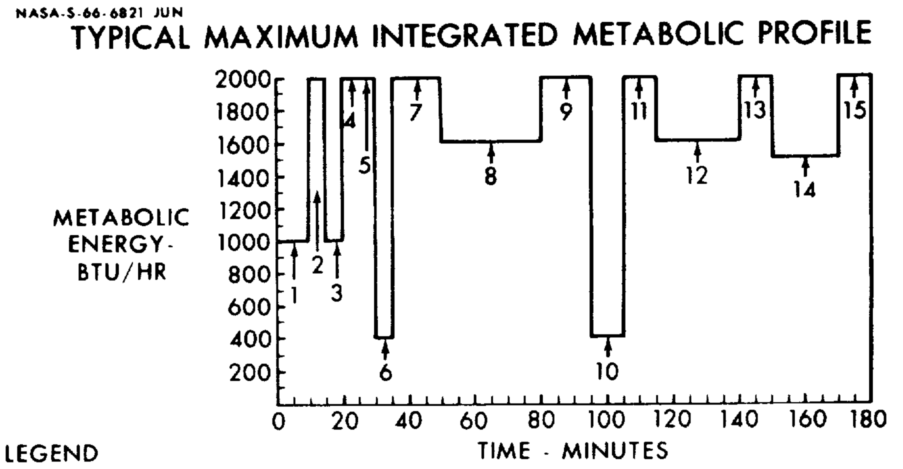

NASA-S-66-3653 APR 12
APOLLO PORTABLE LIFE SUPPORT SYSTEM
EMERGENCY OXYGEN SUPPLY SYSTEM
EXTRA-VEHICULAR VISORS
PRESSURE SWITCH & TRANSDUCER
BIO-INSTRUMENTATION
ANTENNA
BATTERY VOLTAGE SENSOR
CONTROLS
SSC
TERMINAL BOX
MAIN POWER SUPPLY
BATTERY CURRENT SENSOR
OXYGEN & WATER QUANTITY INDICATOR
FAN
OXYGEN TANK
ON-OFF VALVE
REGULATOR
HIGH FLOW WARNING SENSOR
WATER SEPARATOR
WATER ON-OFF VALVE
FEED WATER RESERVOIR
GAS LOOP
ACCUMULATOR
WATER QUANTITY SENSOR
CONTAMINANT CONTROL CANISTER
POROUS PLATE HEAT EXCHANGER
STEAM TO VACUUM
WATER PUMP
PRESSURE GAGE
PRESSURE RELIEF VALVE
THERMAL METEOROID GARMENT
LIQUID LOOP
LIQUID LOOP
LIQUID COOLING GARMENT
DIVERTER VALVE
SPACE SUIT
BACK PACK

FIGURE 5



LEGEND

1 SSA CHECKOUT
2 EGRESS
3 SSA CHECKOUT (LUNAR SURFACE)
4 ERECTION OF SHIELDED STORAGE AREA
5 TRANSFER EQUIPMENT
6 REST
7 WALK TO OBJECTIVE NO. 1
8 SET-UP AND PERFORM EXPERIMENT NO.1
9 RETURN TO LM FOR EQUIPMENT
10 REST
11 WALK TO OBJECTIVE NO. 2
12 SET-UP AND PERFORM EXPERIMENT NO. 2
13 RETURN TO LM
14 PROCESS SPECIMENS
15 INGRESS

FIGURE 6

NASA-S-66-6866 JUN

EMU INSTRUMENTATION

TELEMETERED
- SUIT PRESSURE
- WATER QUANTITY
- OXYGEN QUANTITY
- BATTERY CURRENT
- LIQUID COOLED GARMENT ΔT (INLET-OUTLET)
- SUIT INLET TEMPERATURE
- EKG

SELF MONITORED
- SUIT PRESSURE
- OXYGEN QUANTITY
- WATER QUANTITY
- HIGH OXYGEN FLOW (AUDIBLE WARNING)
- LOW SUIT PRESSURE (AUDIBLE WARNING)

FIGURE 7

NASA-S-66-[illegible] JUN

PLSS CONTROL ACTUATION

FIGURE 8

NASA-S-65-2965

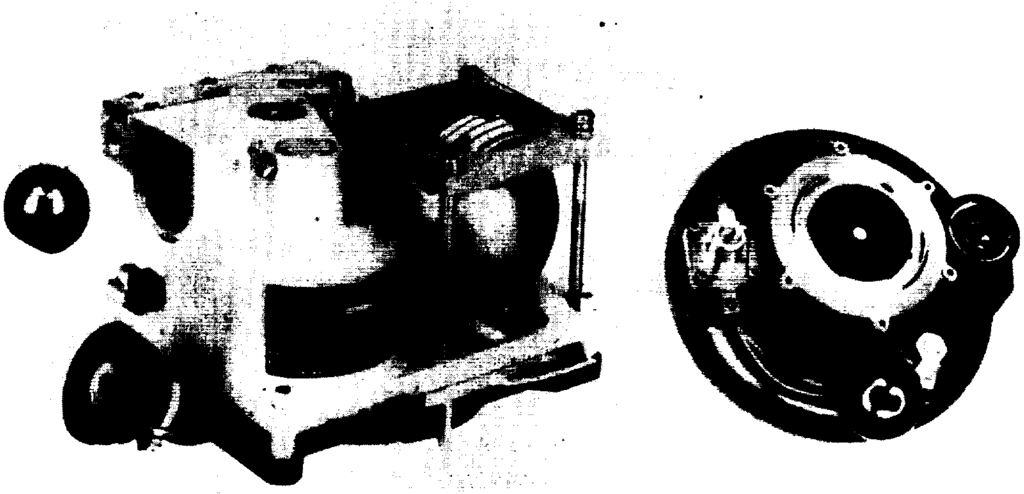


FIGURE 9

NASA-S-66-6853 JUN

PRESSURE GARMENT ASSEMBLY (PGA) DESIGN REQUIREMENTS

- PRESSURE PROTECTION, WITH MINIMUM LEAKAGE
- MOBILITY TO ACCOMPLISH LUNAR EXPLORATION
- STORAGE FOR BODY WASTES
- VENTILATION DISTRIBUTION
- VISIBILITY & EYE PROTECTION
- COMMUNICATIONS
- TOLERABLE FOR CONTINUOUS PRESSURIZED WEAR FOR UP TO 115 HOURS

FIGURE 10

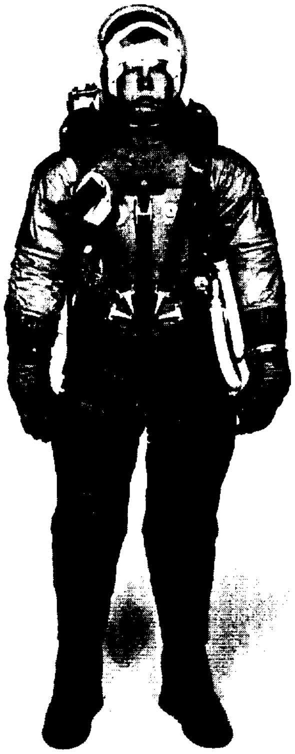


FIGURE 11

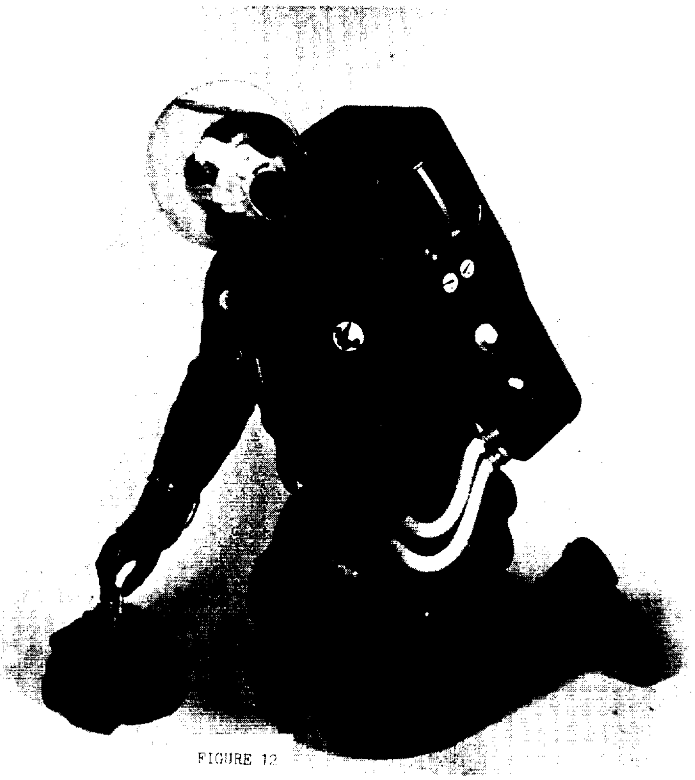


FIGURE 12

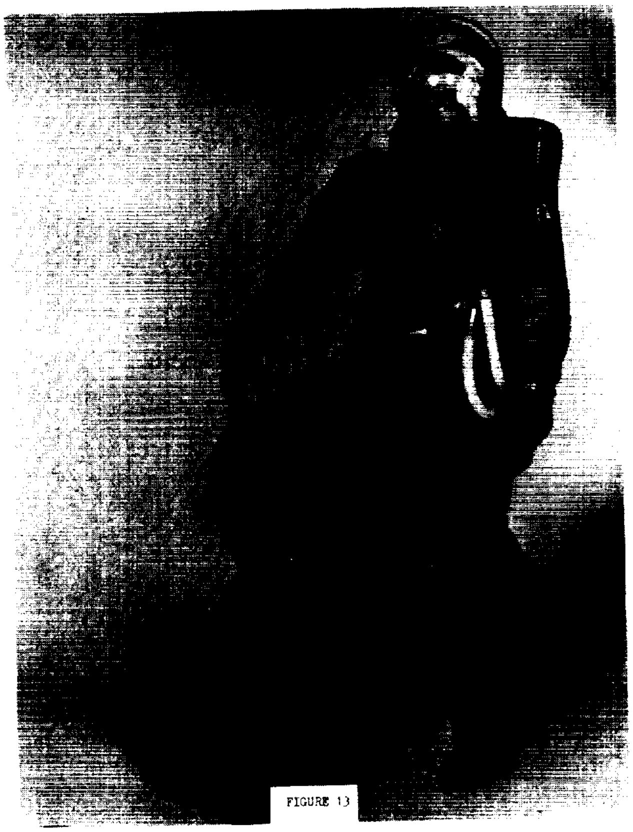


FIGURE 13

FIGURE 14

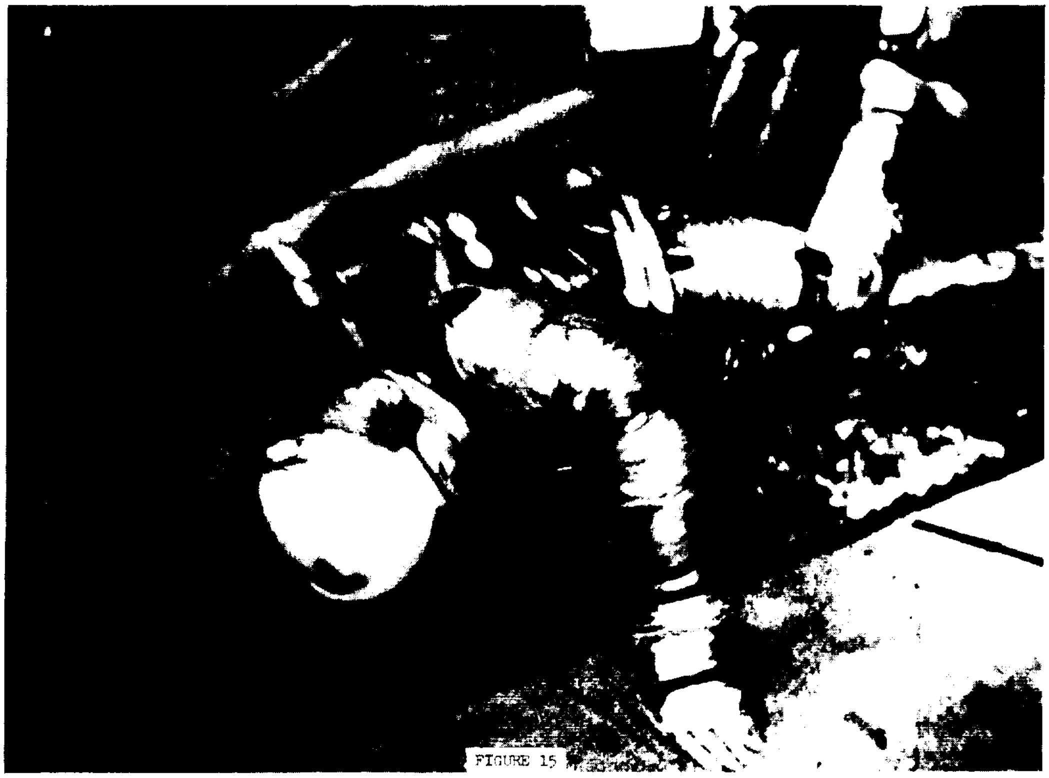


FIGURE 15

WATER COOLED GARMET

FIGURE 16

NASA-S-66-6850 JUN

LIQUID COOLED GARMENT (LCG) DESIGN REQUIREMENTS

- LIMIT PERSPIRATION TO 130 GM/HR AT 2000 BTU/HR
- LIMIT LOCAL SKIN TEMPERATURE TO 105° F MAXIMUM
- LIMIT LOCAL SKIN TEMPERATURE TO 50° F MINIMUM
- TOLERABLE FOR CONTINUOUS WEAR FOR UP TO 115 HOURS

FIGURE 17

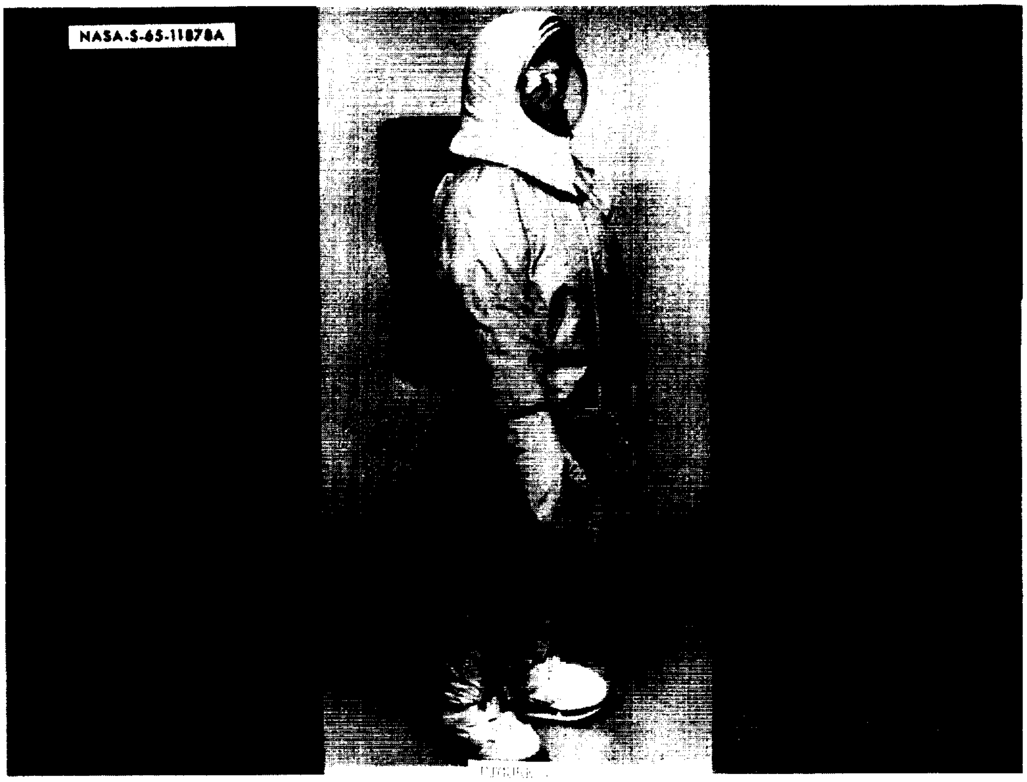


FIGURE [illegible]

NASA-S-66-6849 JUN

THERMAL METEOROID GARMENT (TMG) DESIGN REQUIREMENTS

- ABRASION PROTECTION
- METEOROID PROTECTION
- PASSIVE THERMAL PROTECTION

ELIMINATE "HOT SPOTS"
250 BTU/HR IN
350 BTU/HR OUT

- DON-DOFF CAPABILITY TO PREVENT CONTAMINATION

FIGURE 19

NASA-S-66-6814 JUN

PROTOTYPE APOLLO EXTRA VEHICULAR GLOVE TEST

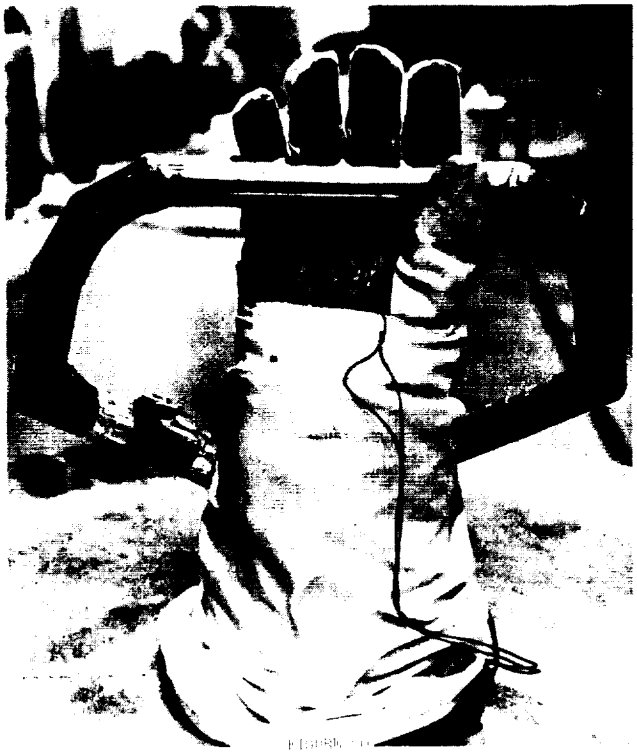


FIGURE 20

PROTOTYPE APOLLO LUNAR BOOTS

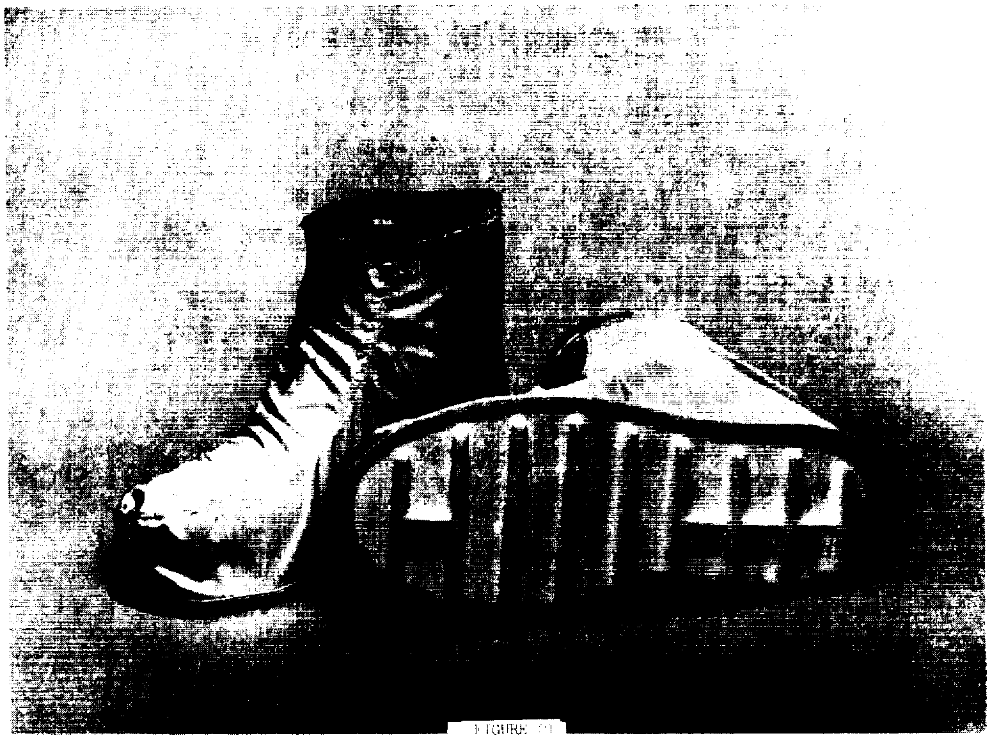


FIGURE 21

PROTOTYPE APOLLO EXTRA VEHICULAR VISOR ASSEMBLY

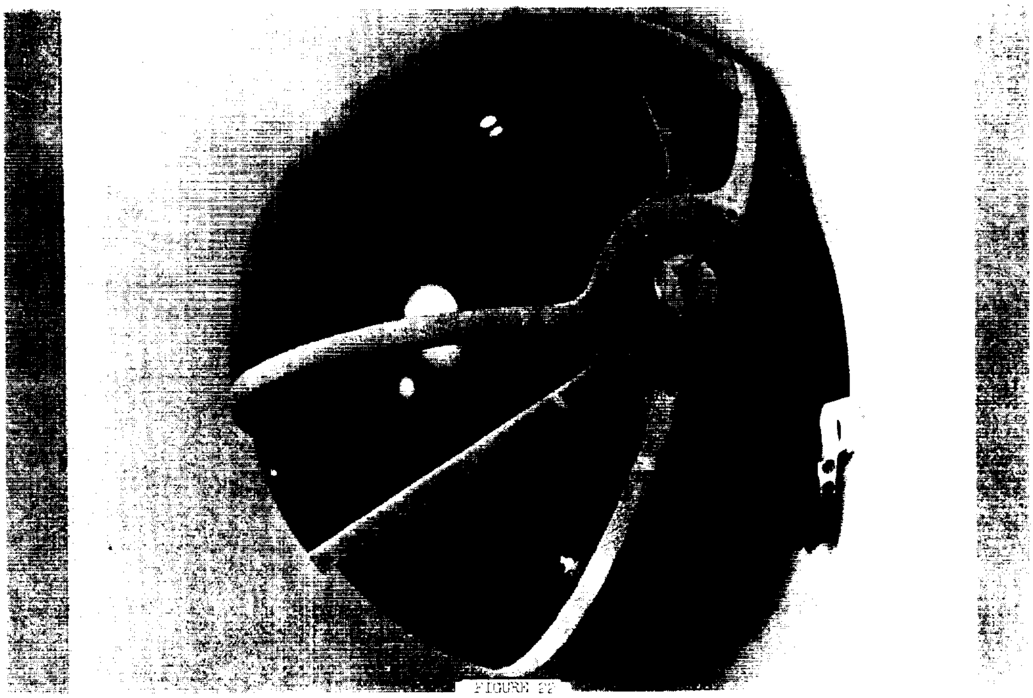
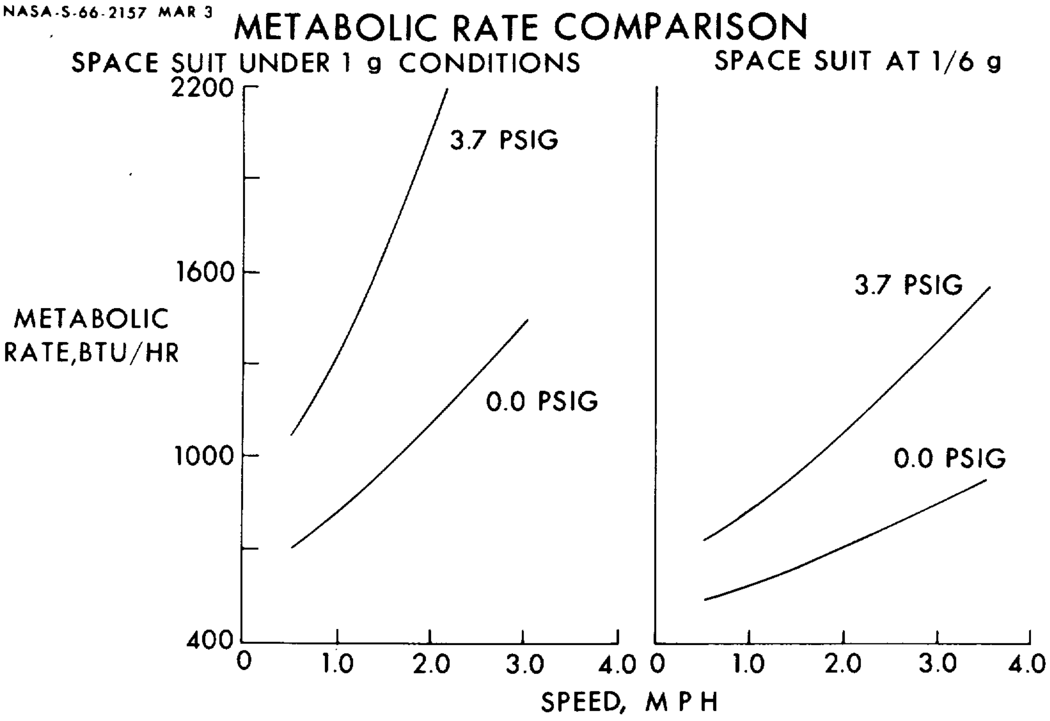


FIGURE 22

APOLLO EMU THERMAL TEST

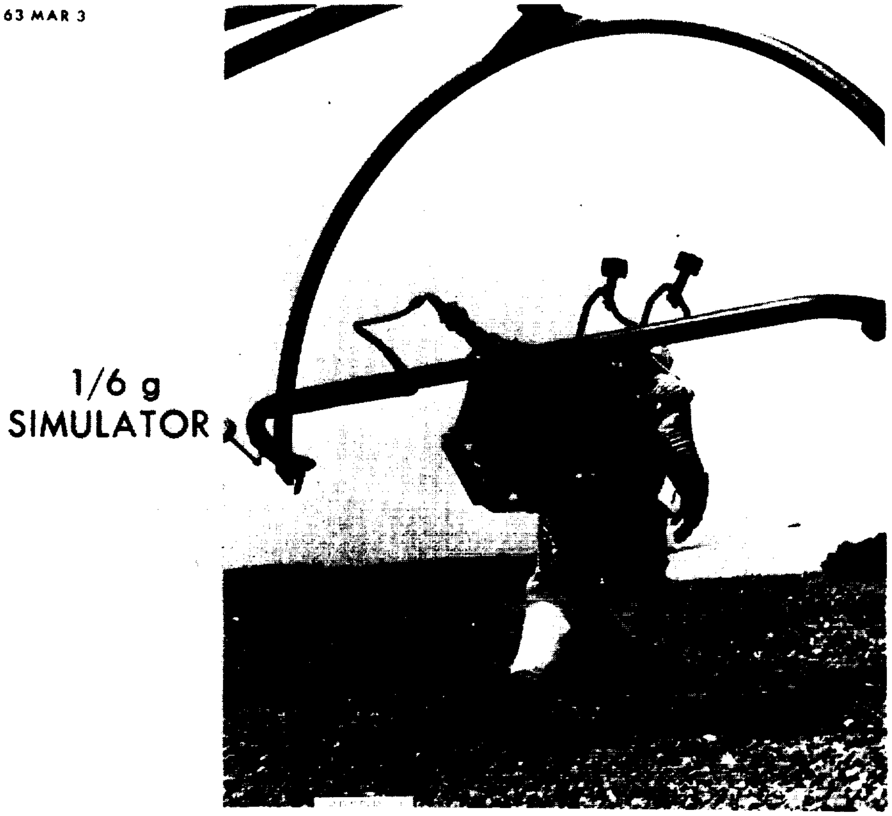
FIGURE 23



METABOLIC RATE COMPARISON

FIGURE 24

NASA-S-66-2163 MAR 3


1/6 g
SIMULATOR

NASA-S-66-6816 JUN

EMU 1/6 GRAVITY - LUNAR SURFACE TESTING

NASA S 65 1866

SUITED ASTRONAUT
USING JACOBS' STAFF
TO ASSIST IN WALKING ON
SIMULATED LUNAR SURFACE

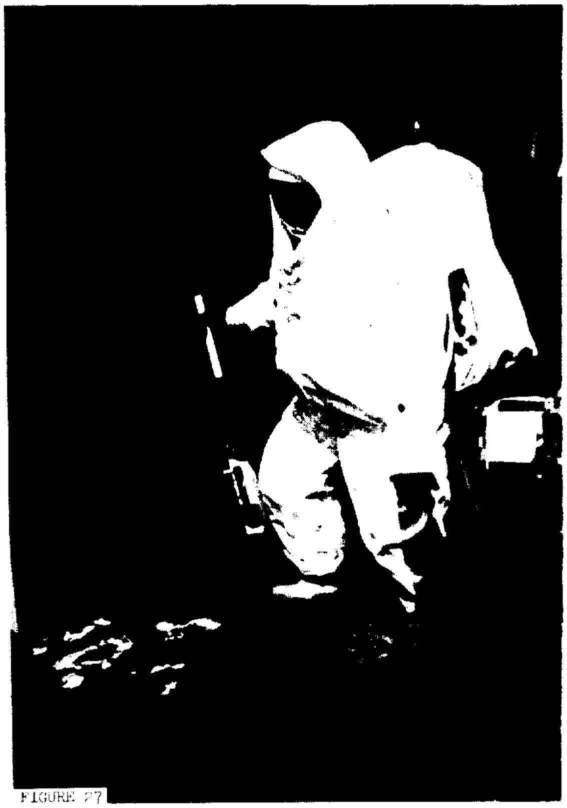
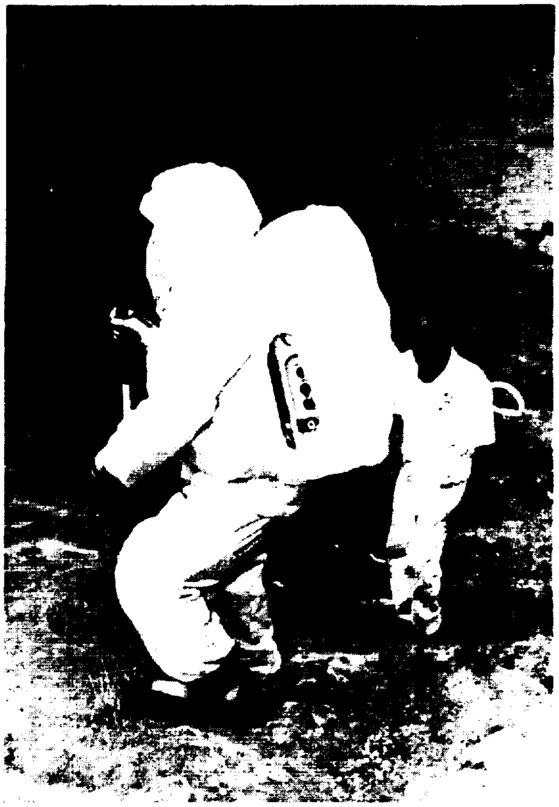


FIGURE 27

NASA S 65 1862

ASTRONAUT WEARING
COMPLETE EMU
DURING SIMULATED
LUNAR SURFACE TESTS
(BEND, ORE)



NASA-S-66-6828 JUN

RANGE LIMITATION RESULTING FROM THE OXYGEN AVAILABLE FOR EMERGENCY USE

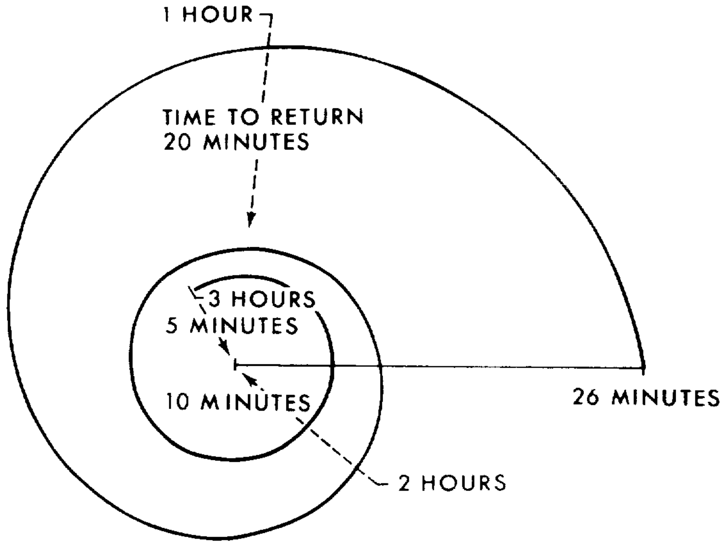

1 HOUR
TIME TO RETURN
20 MINUTES
3 HOURS
5 MINUTES
10 MINUTES
26 MINUTES
2 HOURS

FIGURE 29

LM TOUCHDOWN DYNAMICS

by

Hugh M. Scott

LEM LANDING DYNAMICS

1.0 INTRODUCTION

The landing simulation studies accomplished at MSC define the vehicle state at touchdown. The purpose of this presentation is to give a description of the landing system and to discuss the dynamics of the LEM following the initial landing impact, including the analysis work accomplished and planned to predict vehicle landing performance.

The lunar landing must obviously avoid toppling instability of the LEM, within the flying qualities of vehicle control system, for the range of lunar surface parameters selected for design.

2.0 DEFINITION OF LANDING GEAR DESIGN CRITERIA

Landing gear design was initiated on the LEM before current lunar data, such as Ranger and Surveyor flights, were available. A criteria was chosen for the contractor to design the LEM gear and is summarized in Figure 1. In addition to the geometric characteristics of the design lunar model of 6° general slope with 24-inch depressions or 24-inch protuberances under the foot pads at impact giving a maximum effective slope of approximately 12°, the surface material was considered to be infinitely rigid for shock absorber design. Foot pad size was based on a minimum surface dynamic bearing strength of 12 psi. Shearing or sliding resistance as a friction coefficient was considered to vary from 0.4 to 1.0 in addition to partial or full constraint of the foot pad.

The vehicle velocity, attitudes and rates were established from the 3σ control system capabilities given in the preceding discussion. These criteria are a vertical velocity of 10 ft/sec with a horizontal velocity of 0 ft/sec; a vertical velocity of 7 ft/sec and horizontal of 4 ft/sec. The attitude criteria is $\pm 6^{\circ}$ with a maximum attitude rate of 2°/sec. The control system will be active during landing dynamics, but was not considered so for original gear design and performance analysis. Gear design also considered the possible de-stabilizing moment generated during some landing conditions from crushing of the descent engine skirt extension, and the crush load characteristics used were included as a specification requirement on the engine design. The design landing weights were selected based upon minimum and maximum usage of expendables in the descent from lunar orbit. The maximum landing weight is, in general, critical for the energy absorption requirements while the minimum landing weight, which also has the highest center of gravity position, is critical for the stability requirements.

3.0 DESCRIPTION OF LANDING GEAR

The landing gear design that evolved from this set of requirements is referred to as the "Cantilever" gear. As shown in Figure 2, each of the four gear assemblies consists of a primary strut and two secondary struts. The primary strut consists of an inner cylinder with a foot pad at its lower end, and an outer cylinder connected through a type of universal joint at its upper end to the outrigger support truss. A dual crush level honeycomb cartridge shown at the top of Figure 2 acts in compression to absorb energy at the indicated load-stroke values. Each secondary strut consists of an outer cylinder connected through a ball joint to the primary strut, an inner cylinder connected through the deployment truss to the base of the descent stage, and an arrangement of honeycomb cartridges that can absorb energy, at the load-stroke values shown in the bottom of Figure 2, while the secondary strut is extending or compressing. The 36-inch diameter foot pad has sufficient area and strength to provide flotation and minimal impact penetration on low bearing strength surfaces. The center of each foot pad is 167.57 inches from the LEM centerline.

4.0 LANDING - PERFORMANCE TO DESIGN CRITERIA

To predict landing performance on the lunar surface, considerable effort has been and continues to be expended in extensive landing dynamic analysis. Of prime concern during the development of the analysis was the realistic treatment of the non-linear geometry and loading of the articulating landing gear. Since the properties of the landing surface were not well known, special attention was given to allow the analysis to accommodate a wide variety of conditions. Other significant effects treated in the analysis include the influence of crushing the descent engine skirt extension, fuel slosh, reaction control and engine thrust forces, and the logic required to account for the initiation of engine shutdown and thrust-decay characteristics. Since purely symmetric landings appear unlikely, it became desirable to determine the effect of introducing asymmetric parameters into the initial conditions, (e.g., variations in the flight path with respect to the lunar slope, and vehicle yaw angle with respect to the flight path).

Results of a landing simulation include time histories of all pertinent data (e.g., center of gravity velocities and accelerations, strut loads, strokes, and foot pad position). Another form of the simulation output is a movie describing stroking of the struts, crushing of the engine skirt and vehicle motions resulting from the dynamics. The vehicle is assumed unstable if the LEM center of gravity falls outside a vertical plane passing through any two adjacent foot pads. In a stable run, the minimum distance between the LEM center of gravity and any of these vertical planes is recorded as a measure of stability. Results of such an analysis with various initial conditions, within the criteria, give us performance boundaries indicated by Figure 3. These boundaries then indicate that the present gear design is optimized to a high degreee to meet the original design criteria for worst case energy absorption and stability critical landing conditions.

5.0 LEM STATE AT TOUCHDOWN

Factors affecting the vehicle state at touchdown include probes mounted on the LEM foot pad used to sense the landing surface and to provide a signal for engine thrust termination (See Figure 4). A landing surface bearing strength of about 3 psi acting on the probe tip is sufficient to activate the cutoff signal. Upon receipt of the sensing probe signal, the astronaut manually terminates the thrust.

Terminating thrust before touchdown causes the spacecraft velocity to increase at touchdown. The earlier the thrust is terminated, the higher the touchdown velocity. If the astronaut delays too long in terminating thrust, the engine will be firing at touchdown. Thus, two possible problem areas exist: landing with velocities in excess of design velocity and landing with the engine on.

Figure 5 shows the predicted 99% probability touchdown velocities together with the touchdown velocity envelope. The predicted touchdown velocity includes the effect of astronaut reaction time and system delays. The figure shows that the predicted touchdown velocities fall well within design values. Therefore, no problem is anticipated in this area.

To avoid possible undesirable failure modes of the engine skirt and increased pressure and temperatures on the base heat shield, it is desirable to terminate thrust by the time the footpads contact the surface. At this point, the engine skirt is approximately 19 inches above the surface. The engine may be thrusting at touchdown if the descent velocity is too high or if the astronaut delays too long in initiating engine cutoff. Figure 6 shows the probability of a given vertical velocity occurring at probe surface contact when the nominal descent velocity of 3.5 fps is desired. The figure also shows the probability of a given crew reaction time for initiating engine cutoff. These data were generated during the landing simulator studies mentioned above.

Referring to the figure, there is a 97% probability that the descent velocity at probe contact will be less than 5 fps. For this descent velocity, the astronaut reaction time must be in one-half second before it is physically possible for the engine to be thrusting at touchdown. The probability of the astronaut reaction time exceeding one-half second is about 2%. Combining the two probabilities results in a probability of about 99.94% that the engine will be off at touchdown.

The minimum engine operation height above the surface of 19 inches used at this time is from preliminary plume analysis. To better define the engine/surface interaction limits, MSC has initiated contracts to develop two different analytical models of the engine plume/surface interaction as well as an experimental program.

One analytical approach will develop a free plume program for an ideal nozzle (parallel flow at exit). An initial investigation of a plume impinging on a surface will also be made to obtain approximate surface pressures. The other analytical program will develop a free plume program for a Rao nozzle, assuming no strong shocks and omitting that portion of the flow where a lip shock occurs. An experimental program, using the Langley Research Center 41 foot high altitude facility, will be performed incorporating the Apollo 100-pound thrust Reaction Control System engine (a 1/10-scale LEM descent engine). These tests will correlate the above analytical programs and aid in evaluating the descent engine flow under various conditions. (See Figure 7)

Phase II of this effort will select the more descriptive computer program, and continue using the selected program for engine flow near the surface to evaluate surface interaction. This study will consider both normal and off-normal orientation of the engine plume to the surface. The initial analytical and experimental programs should be completed by September 1966. The more detailed Phase II completion is estimated for February 1967.

6.0 IMPLICATIONS OF LUNAR ENVIRONMENT ON LANDING PERFORMANCE

As noted earlier, the landing performance of the LEM is satisfactory for the design lunar surface model. We now turn our attention to the landing performance on surfaces that are less rigid than the design surface.

Determination of landing performance on soft surfaces relies heavily on a knowledge of the interaction of the landing gear foot pad with the soft surface as the pad penetrates the surface.

The interaction forces and moments depend on the soil dynamics and the foot pad size and shape.

Since soil dynamics in a lunar environment is a relatively unexplored field, a considerable amount of experimental and theoretical work has been initiated in this area. This work is aimed at determining the foot-pad/soil interaction for several soil types that have a high probability of occuring on the lunar surface. The necessary data should start becoming available within the next six months.

In the meantime, some landing performance bounds can be established through the use of simplified theoretical soil/foot-pad interaction models. Experimental and theoretical interaction force models that have been developed to date usually are of the form expressed by the equation in Figure 8.

The first two terms in the equation are associated with the static penetration resistance, while the last two terms are associated with the dynamic penetration resistance of the soil. This equation has been programmed into MSC's digital computer simulation of the LEM lunar touchdown dynamics. Work is underway to determine the lunar landing performance based on this theoretical foot-pad/soil interaction model. The next several figures summarize the results of our work to date. As experimental data becomes available, the results shown here will be updated as necessary.

The ground rules being used to determine soft surface landing performance are listed in Figure 9. Lateral motion of the foot pad was not allowed to get conservative answers for the vertical penetration.

The procedure for determining landing performance is outlined in Figure 10.

Figure 11 is an example of how "bad" stability cases are determined. The stability performance was computed for each set of initial conditions and plotted vs stability margin. Stability margin is the minimum distance between the LEM c.g. and a vertical plane passing through any two adjacent foot pads as shown in Figure 12.

The worst case is the point of least stability. Similar curves are being generated for a range of slopes, velocities, yaw angles, and interaction models.

The next step in landing performance evaluation is to determine foot pad penetrations.

Figure 13 shows foot pad penetrations for soil where penetration increases linearly with depth. Foot pad penetration is not too dependent on landing slope for these soil models. One of the study ground rules is that the spacecraft heat shield shall not contact the surface. Heat shield contact occurs when the foot pad penetration exceeds approximately 44 inches. Figure 13 shows that the surface strength should be greater than about 5 psi/ft to avoid penetrations greater than 44 inches.

Also shown on Figure 13 is the effect of adding a small amount of dynamic penetration resistance to the 5 psi/ft soil. As could be expected, the penetration decreases considerably when the dynamic terms are included. It will be shown later that stability also tends to improve as the dynamic penetration resistance increases.

Figure 14 shows foot pad penetration for a soil whose penetration resistance is constant with depth. A soil strength of about 8 psi is required to prevent foot pad penetration beyond 44 inches in this type of soil.

The results of the previous figures have been plotted together on Figure 15 to obtain a preliminary landing performance envelope.

The horizontal lines represent stability boundaries for the different soil models considered. It is important to note that soils that are acceptable from a penetration standpoint provide good landing areas from a stability standpoint. In fact, the stability performance on these soils is practically the same as the rigid surface performance.

Based on this preliminary data, it appears that the LEM can land safely on a surface slope of about 7 degrees plus 2-foot depressions, provided the soil strength is greater than 5 psi/ft or 8 psi depending on the type of soil.

This simplified theoretical soil/foot pad interaction model is also useful to approximate landing performance on particular landing areas as more information becomes available, e.g., the recent Surveyor 1 site. The limited penetration of the Surveyor foot pads into the lunar surface leaves the soil properties open to interpretation, but as far as LEM landing acceptability, it may be bounded using the findings presented in the preliminary report of the Surface Mechanical Properties working group. This report suggests that the bearing strength of the material in the area of the Surveyor foot pads photographed is approximately 5 psi. If we assume this to be a material that bearing strength varies linearly with depth, then the 5 psi at 2 inches could be represented by a 30 psi/ft interaction model which would result in LEM landing performance similar to that shown for a rigid surface.

If we assume the conservative interaction model of 5 psi constant, then the computed penetration of the LEM for a 10-fps straight down landing would be approximately 8 inches for all four foot pads. By introducing horizontal velocity component and vehicle maximum attitude and attitude rates, the greatest foot pad penetration is approximately 1.7 feet; therefore, for landing performance considerations, the LEM would land safely on a landing area similar to the Surveyor 1 site.

7.0 CONCLUSIONS

LEM landing performance meets the original specification design criteria.

The probe surface sensor allows manual engine cut-off consistent with design touchdown velocities and minimum engine operation height above the surface.

Study is continuing to determine minimum engine operating height above the surface that is required to establish maximum descent velocity and range of reaction times consistent with design touchdown velocities.

LEM landing performance on soft surfaces relies heavily on footpad/soil interaction forces.

Experimental work to determine interaction forces for several soil types is underway.

Preliminary data based on theoretical interaction forces indicate that the LEM can land safely on a slope of about 7 degrees plus 2-foot depressions if the soil strength is 5 psi/ft or 8 psi constant.

LEM landing would be successful on a lunar surface similar to the one indicated by the Surveyor 1 spacecraft.

Questions and Answers

TOUCHDOWN DYNAMICS

Speaker: Hugh Scott

1. Is it possible for LM to come to rest on three of the four foot pads?

   ANSWER - Yes. Simulation tests show that if the center of gravity is forward of the mid gears the LM will come to rest with one of the pads off the surface.

2. Assuming 7 ft/sec. vertical velocity, 4 ft/sec. horizontal velocity, and a 12 degree effective slope, then what coefficient of friction is required to overturn the vehicle?

   ANSWER - The performance boundaries shown in the presentation used infinite coefficient of friction, and the vehicle did not turn over.

NASA-S-66-5131 JUNE 7

LM LANDING CONDITIONS

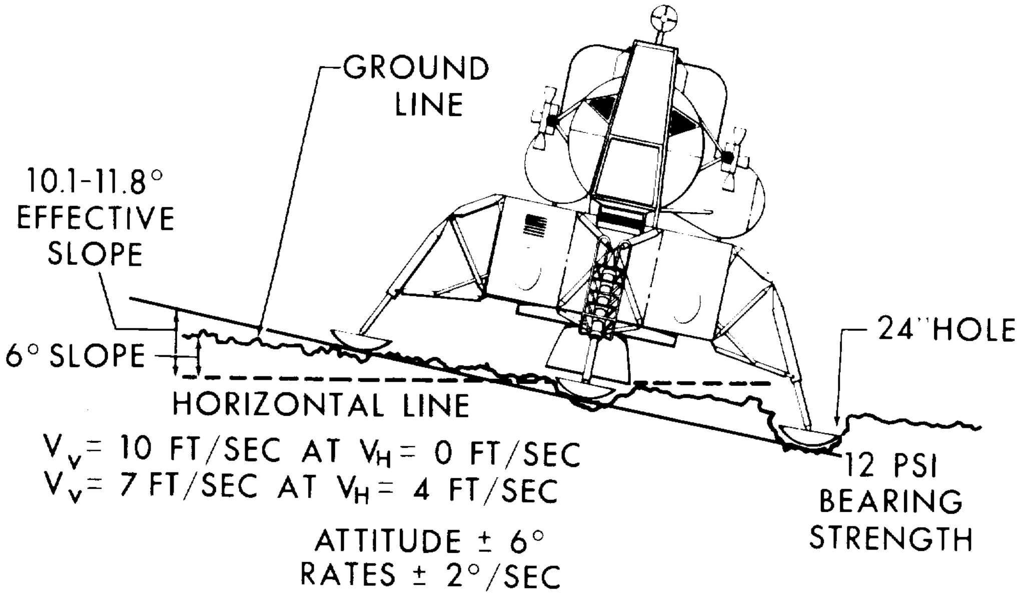

GROUND LINE
10.1-11.8° EFFECTIVE SLOPE
6° SLOPE
HORIZONTAL LINE
24" HOLE
12 PSI BEARING STRENGTH
V_V = 10 FT/SEC AT V_H = 0 FT/SEC
V_V = 7 FT/SEC AT V_H = 4 FT/SEC
ATTITUDE ± 6°
RATES ± 2°/SEC

Figure 1

NASA-S-66-5128 JUNE 7

LANDING CONDITIONS - LOAD STROKE

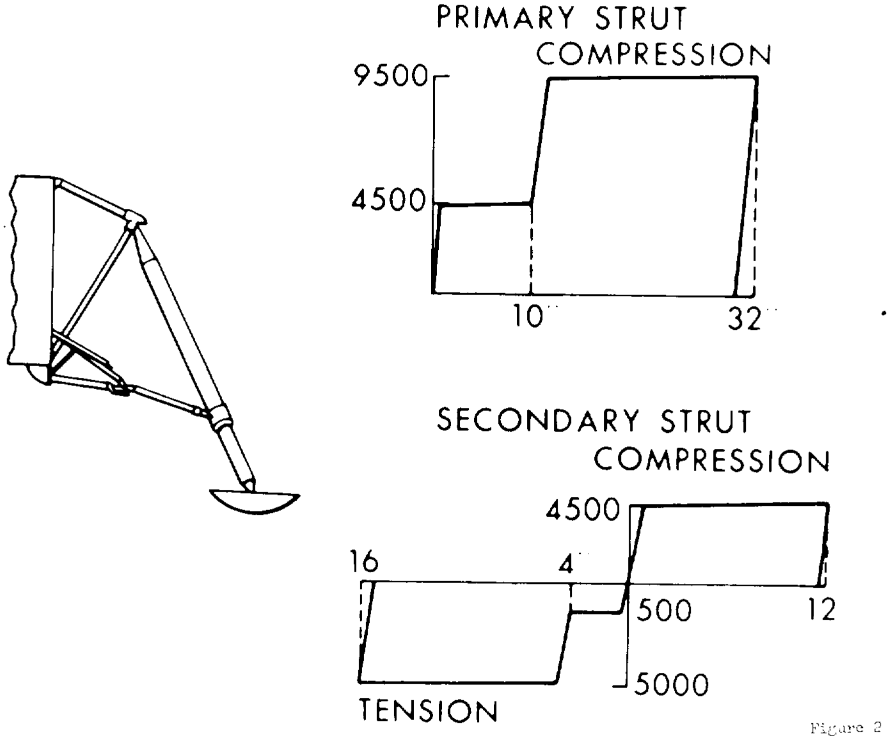


Figure 2

NASA-S-66-5130 JUNE 7

LANDING PERFORMANCE

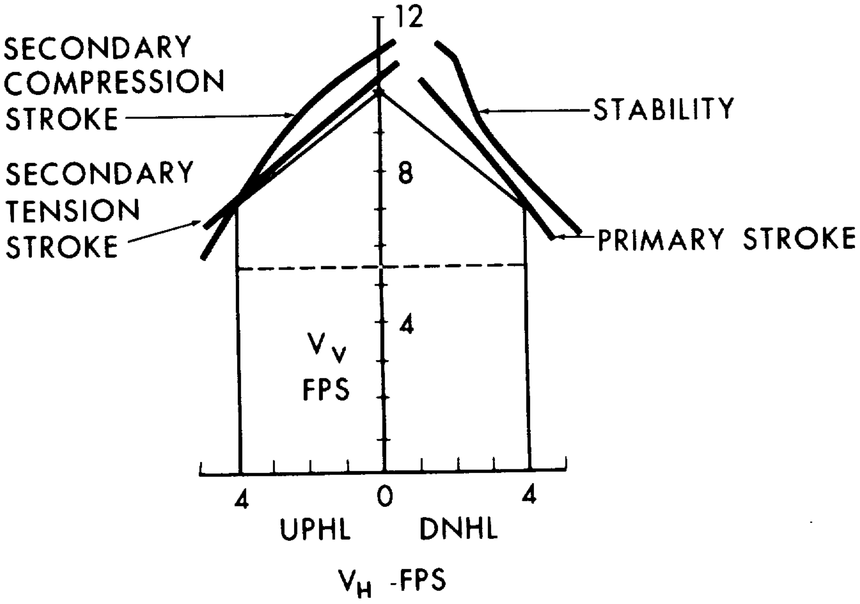
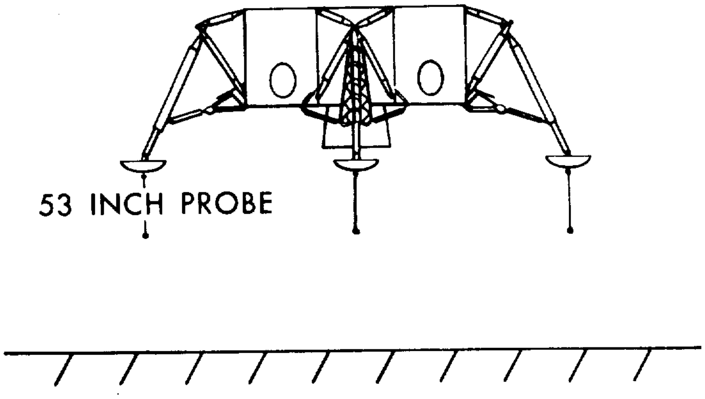


Figure 3

NASA-S-66-5129 JUNE

PROBE SENSOR


53 INCH PROBE

Figure

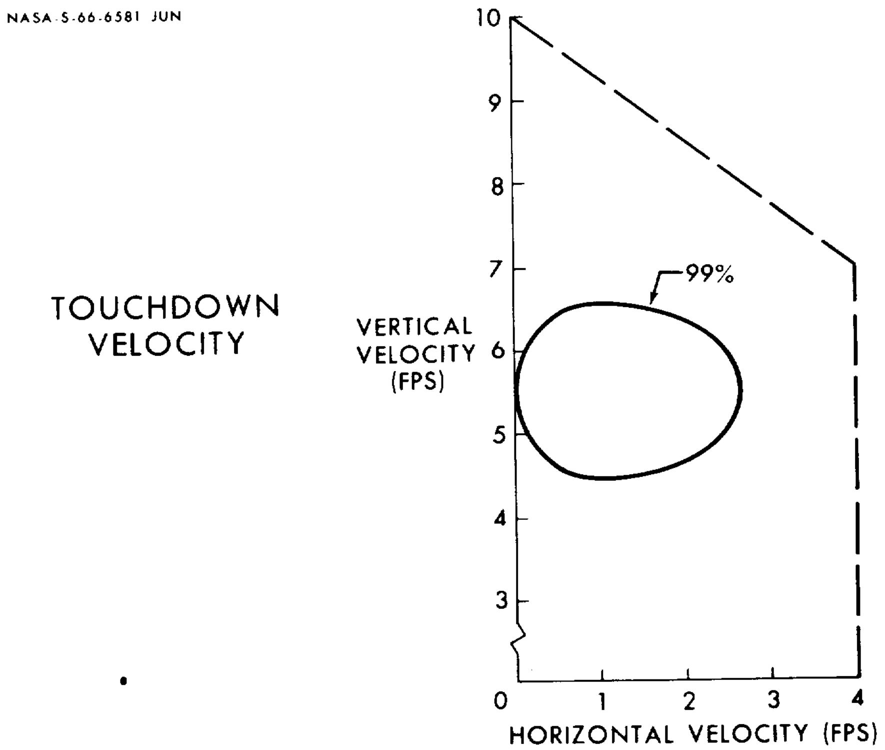


Figure 5

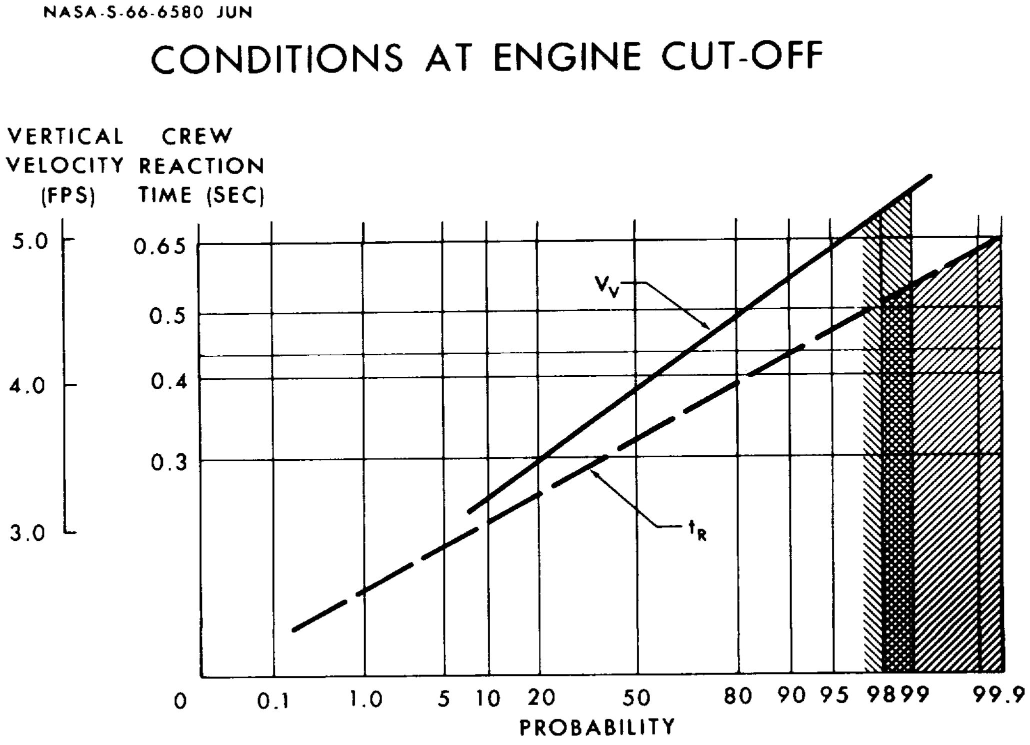


Figure 6

NASA S 66 6862 JUN

PLUME-SURFACE INTERACTION TEST IN LRC VACCUM CHAMBER

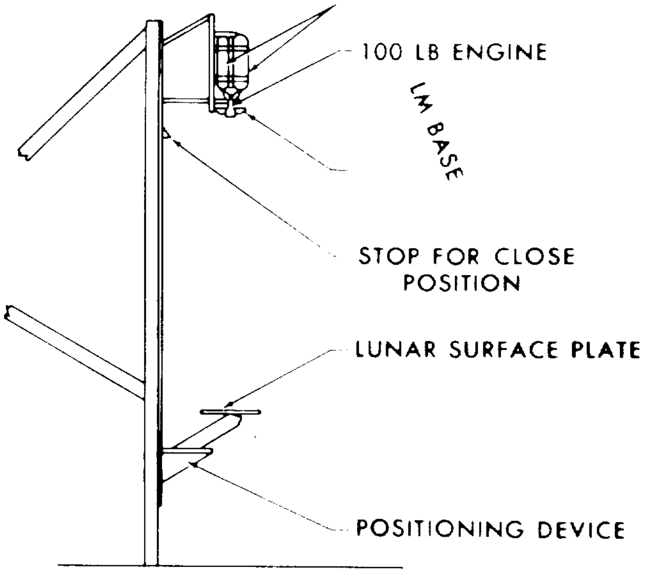

100 LB ENGINE
LM BASE
STOP FOR CLOSE POSITION
LUNAR SURFACE PLATE
POSITIONING DEVICE

Figure 7

NASA-S-66-5161 JUN

SOIL MODEL

$$F = (K_1 + K_2X + K_3\dot{X} + K_4\dot{X}^2)\ A_P$$

Figure 8

NASA-S-66-5162 JUN

GROUND RULES

- DESCENT STAGE HEAT SHIELD SHALL NOT CONTACT LANDING SURFACE
- 4-JET CONTROL SYSTEM ACTIVE
- NO LATERAL FOOTPAD MOTION
- SURFACE PROTUBERANCES & DEPRESSIONS $\leq$ 2 FT
- TOUCHDOWN STATE VECTOR-SAME AS DESIGN CRITERIA

Figure 9

NASA-S-66-6079 JUN

PERFORMANCE EVALUATION PROCEDURE

1) CHOOSE INTERACTION MODEL, E. G., $F = K_2X(A_P)$

2) DETERMINE "BAD" STABILITY & PENETRATION LANDINGS BY SIMULATING SERIES OF LANDINGS WHILE PARAMETICALLY VARYING TOUCHDOWN CONDITION & SLOPE

3) DETERMINE MAXIMUM FOOTPAD PENETRATION VS SLOPE

4) DETERMINE TOUCHDOWN STABILITY VS SLOPE

Figure 10

NASA-S-66-6081 JUN

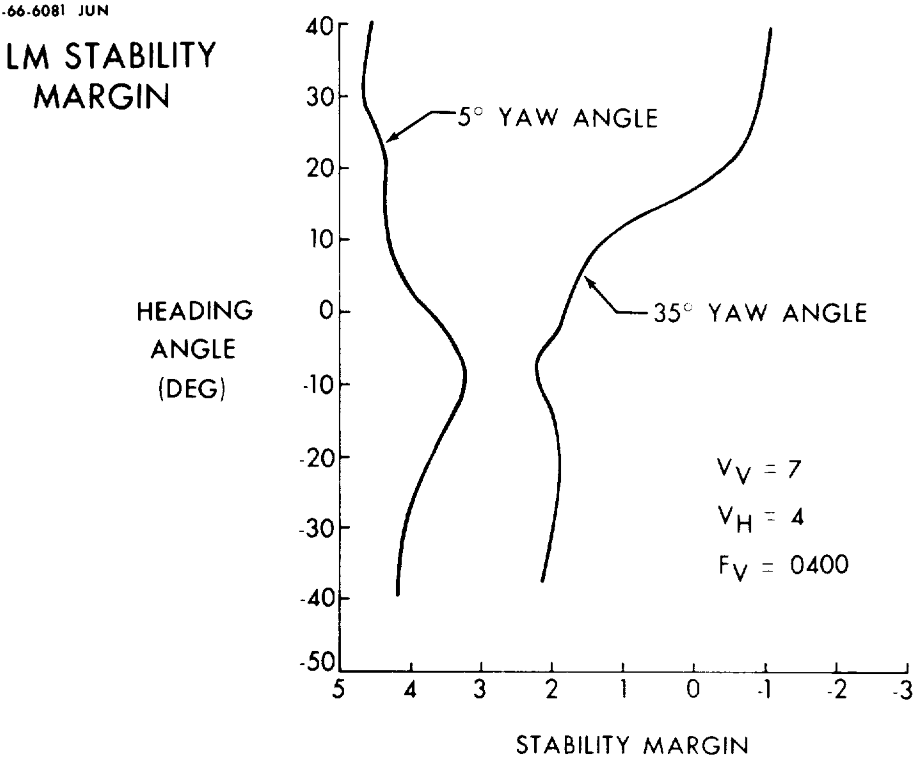


Figure 11

NASA-S-66-6857 JUN

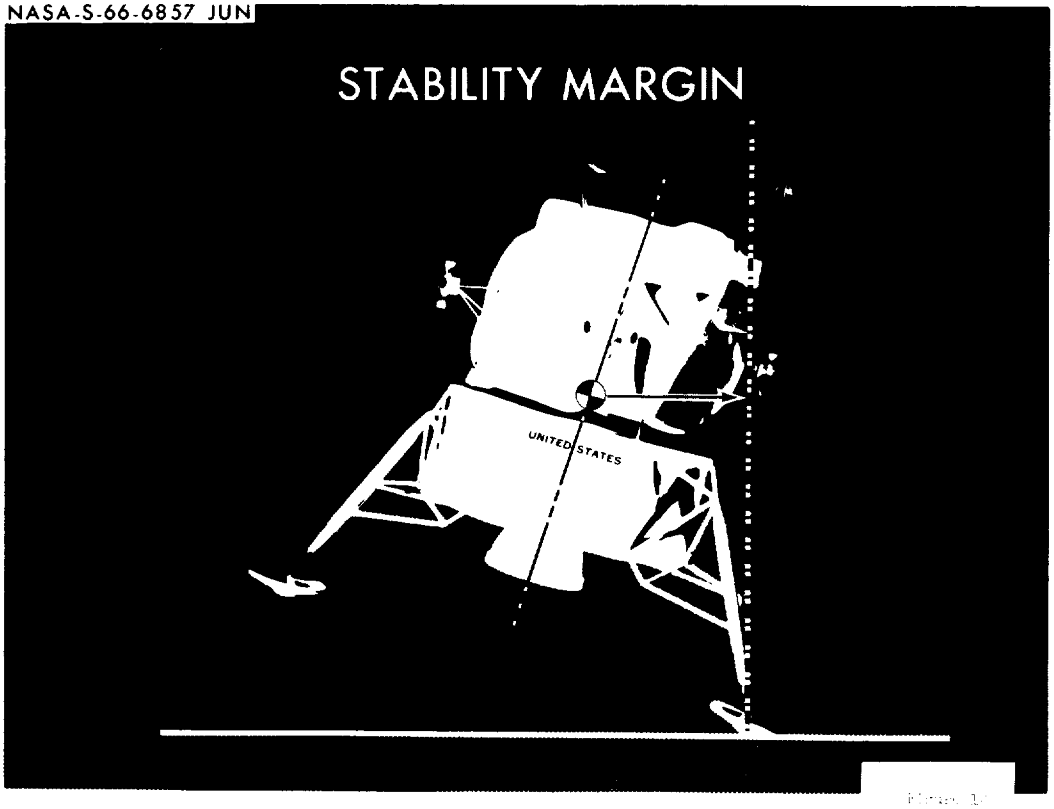

STABILITY MARGIN
UNITED STATES

Figure 12

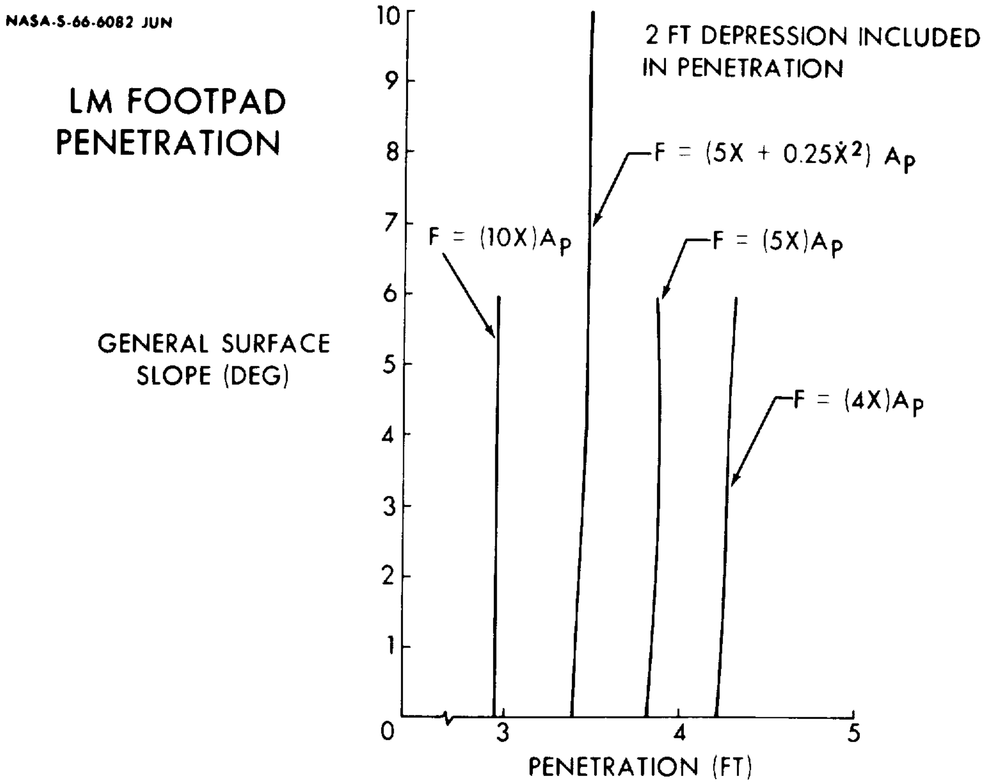


Figure 13

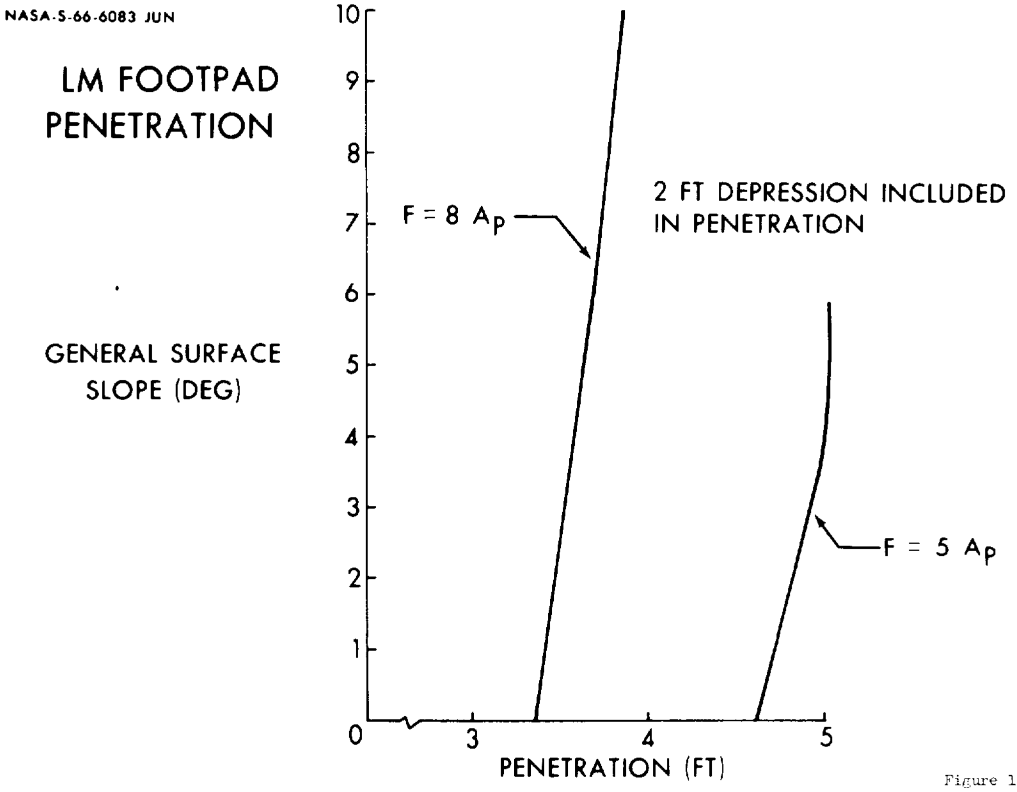


Figure 14

NASA-S-66-6080 JUN

PRELIMINARY LANDING PERFORMANCE ENVELOPE

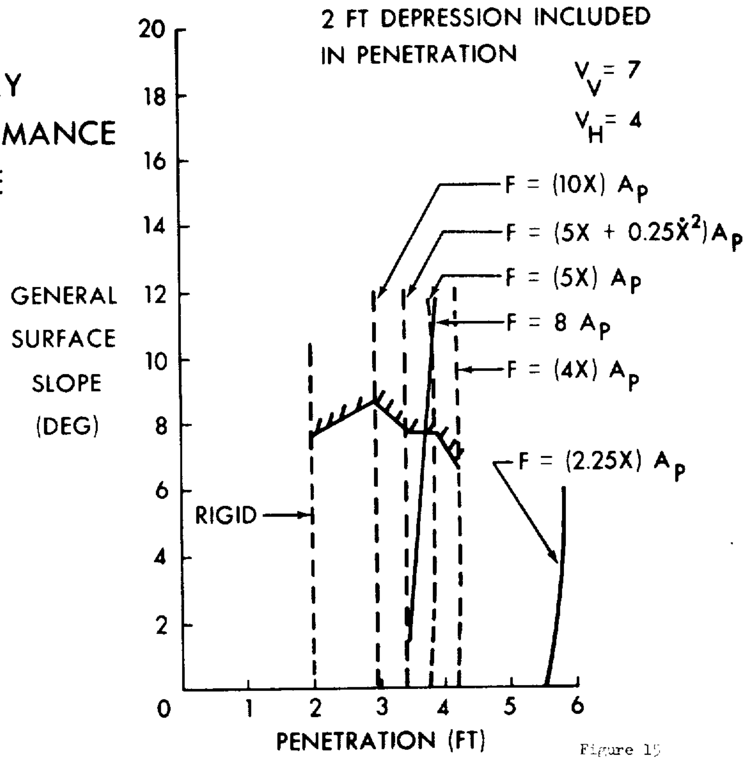


Figure 15

LANDING SITE SELECTION CRITERIA

by

A. V. Bernard, Jr.

LANDING SITE SELECTION CRITERIA

1.0 INTRODUCTION

The presentation on Site Selection is given in two parts: Site Selection Criteria and Site Selection Data Sources and Interpretation (Fig. 1). This document presents the Site Selection Criteria portion of the presentation and the associated charts.

The objectives of the site selection activities as shown on Fig. 2 are:

- To develop site selection criteria that will maximize the probability of successful LM landing
- To develop methods of ranking candidate sites
- To utilize lunar surface data in ranking candidate sites
- To select the landing sites

The site selection process must be completed not later than six months prior to launch to satisfy targeting requirements.

The capabilities and constraints of LM and its associated subsystems have been developed in previous papers. Similarly, the operational problems that must be considered have been developed in previous papers. It is these various considerations that constitute the site selection criteria. This criteria used in conjunction with the available lunar surface data provides the mechanism for candidate site selection as noted on Fig. 3.

The various considerations used to establish the site selection criteria have been divided into two categories: Operational considerations and spacecraft/surface interactions. These two categories will be discussed in subsequent sections.

2.0 OPERATIONAL CONSIDERATIONS

The operational considerations consist of lunar lighting, visibility and CSM performance as noted on Fig. 4. Lunar lighting and visibility will be discussed in some detail followed by a shorter discussion on the implications of lighting and CSM performance on the accessible lunar area and the required landing site spacing.

NASA-S-66-6117 JUN

LUNAR LANDING SITE SELECTION

- SITE SELECTION CRITERIA
- SITE SELECTION DATA SOURCES AND INTERPRETATION

Fig.

NASA-S-66-6112 JUN

OBJECTIVES

- DEVELOP CRITERIA TO SELECT SITES THAT WILL MAXIMIZE PROBABILITY OF SUCCESSFUL LM LANDING
- DEVELOP METHODS OF RANKING CANDIDATE SITES
- UTILIZE LUNAR SURFACE DATA IN RANKING CANDIDATE SITES
- SELECT LANDING SITES
- TARGETING REQUIRES SITE SELECTION NO LATER THAN SIX MONTHS PRIOR TO LAUNCH

Fig.

NASA-S-66-5399 MAY 31

BASIS FOR SITE SELECTION

- LIGHTING REQUIREMENTS
- CSM PERFORMANCE
- GN&C REQUIREMENTS
- LM LANDING REQUIREMENTS

SITE
SELECTION
CRITERIA

CANDIDATE
SITE
SELECTION

- EARTH-BASED OBSERVATIONAL DATA
- RANGER DATA
- LUNAR ORBITER DATA
- SURVEYOR DATA

LUNAR
SURFACE
DATA

Fig. 3

NASA-S-66-6116 JUN

OPERATIONAL CONSIDERATIONS

- LIGHTING CONSIDERATIONS
- CSM PERFORMANCE

Fig. 4

2.1 LUNAR LIGHTING AND VISIBILITY

An important crew task during LM descent is visual inspection of the lunar surface. This requirement for visual evaluation of the surface will affect site selection by the imposition of constraints on acceptable sun angles. The range of sun angles, in turn, influences the spacing of lunar sites and launch dates. In this section, the basis for the current choice of minimum and maximum sun angles is developed.

The visibility of obstacles is strongly dependent on the solar and viewing elevation angles. In general, above some minimum value of sun angle, visibility decreases as sun angle increases. Thus, for a given viewing angle, it is necessary to select a maximum sun angle which permits both acceptable visibility and allows sufficient operational flexibility.

The viewing angle to the nominal landing site is fixed by the descent trajectory and varies during the LM descent. The maximum acceptable sun angle for a given trajectory will, thus, depend on the point during descent at which visual interrogation of the surface is necessary.

In spite of the face that this point is not yet firmly determined it is possible to establish preliminary criteria for the minimum and maximum sun angles based on several sources of data. As indicated on Fig. 5, these are:

- Lunar reflectance characteristics.
- Detection range estimates.
- The effects of glare for forward sun angles.
- The need for shadowing to enhance site evaluation and detection.
- The amount of shadowing at low sun angles.

Correlation of these results with the viewing angle history resulted in the range of sun angles and lighting conditions currently used for site selection. Each set of results will now be described.

2.1.1 LUNAR REFLECTANCE PROPERTIES

Fig. 6 presents the variation in luminance of the horizontal lunar surface as it is viewed over a 180° range for three sun

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LUNAR LIGHTING & VISIBILITY

DATA SOURCES

- REFLECTANCE PROPERTIES
- ANALYTICAL STUDIES OF DETECTION
- EFFECTS OF GLARE
- SHADOWING REQUIREMENT FOR RECOGNITION
- SHADOWING AT LOW SUN ANGLES

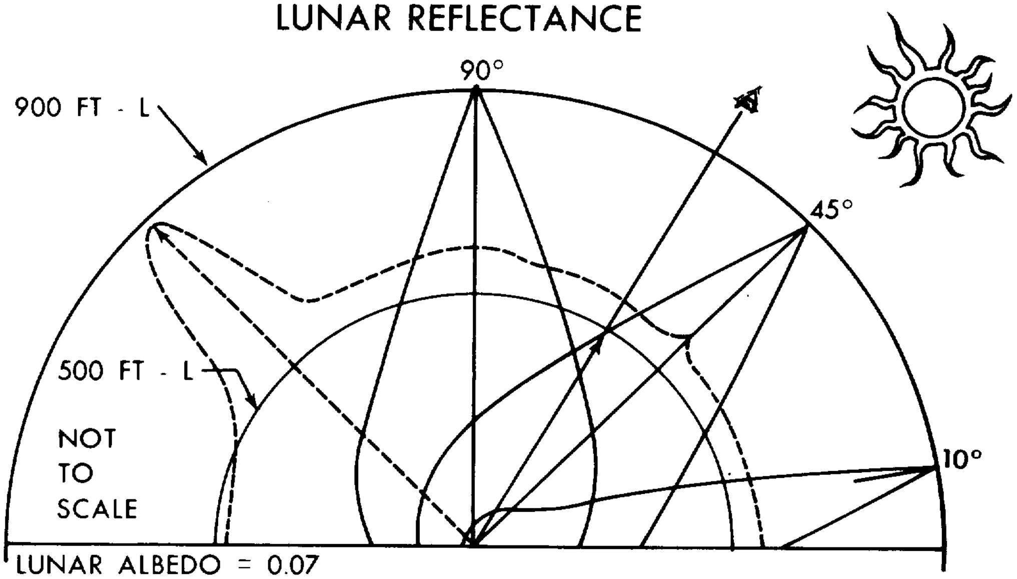
DETERMINATION OF SUN ANGLE LIMITS

- CORRELATION WITH VIEW ANGLE

Fig. 5

NASA-S-66-3878 JUN

LUNAR REFLECTANCE



LUNAR ALBEDO = 0.07

SOLAR ILLUMINATION = 13,000 LUMENS/FT^2

--- SEMI-SPECULAR EARTH MATERIAL

—— RETRO REFLECTIVE LUNAR SURFACE

REF: JPL TR NO. 32-664

Fig. 6

angles; 10°. 45°. and 90°. Two luminance contours are shown, 500 ft.-Lamberts and 900 ft.-Lamberts. Shown also, but not to scale, is a reflectance curve for a semi-specular material as an example of typical Earth terrain.

The important lunar reflectance characteristics are summarized on Fig. 6. These lunar reflectance properties were derived by JPL from Earth-based telescopic observations for use in the Ranger and Surveyor Programs (JPL TY 32-664, "The Lunar Reflectivity Model for Ranger Block III Analyses", by D. Willingham, November 1964). Although the telescopic data are of low resolution, the Ranger pictures have provided some confirmation at higher resolutions. Furthermore, studies of lunar surface models indicate that the surface structure responsible for such reflectance characteristics can be in the mm region. The possibility of deviations on the small scale from the currently used reflectance standard must be considered and it is hoped that analysis of the Surveyor pictures will provide additional information on this point.

The data presented on Figures 6 and 7 indicate that the lunar surface acts as a retroreflector; i.e., most of the incident illumination is reflected back along the direction of incidence. This is quite unlike typical Earth terrain for which peak reflectance occurs at a viewing angle opposite and equal to the incident angle and for which the reflectance is relatively uniform at other viewing angles. The lunar surface luminance reaches a maximum when the viewing angle equals the solar angle, that is, when the sun is directly behind the viewer. A rapid dropoff of luminance occurs on either side of this zero phase angle. For example, when the sun is at an angle of 45 degrees, the data in Fig. 6 indicates that luminance decreases from 900 ft.-Lamberts to 500 ft.-Lamberts when the viewing angle is increased 15 degrees beyond the zero phase angle. The luminance value at zero phase is equal for all sun angles. At that point, a variation in sun angle is equivalent to a variation in surface slope in terms of the amount of light reflected. Thus, a variation in luminance will not occur as slope is varied for this condition and the scene will appear homogeneous and "washed out". Except at the washout area, surface slope is the main contributor to visual contrast since the lunar albedo variation is small. However, for zero phase, even surface slope is ineffective. This is the reverse of the Earth situation in which albedo and color variations are most important and slope variations are of relatively minor significance. In addition to slope variations, shadows provide contrast areas and are the dominant visual features for low sun angles.

2.1.2 DETECTION RANGE ESTIMATES

Analytical studies of obstacle detection ranges have been conducted by MSC and by contractors under MSC direction. These studies are based on the lunar reflectance characteristics just described and on standard visual threshold data.

The principal results of such studies are parametric analyses of obstacle detection ranges as a function of solar angle and viewing angle. These results indicate the regions of optimal visibility and the relative degradation for deviations from the optimal. Less confidence can be attached to the absolute values of estimated detection range than can be given to the form of the functions; however, experimental studies are in progress to more firmly establish absolute values.

Typical results are shown in the next three figures. Fig. 8 presents detection range as a function of solar angle for model craters and protuberances. Two viewing angles are shown, 14° and 38°. Both obstacles were assumed to have depth (or height) to diameter ratios of 1:10 and to be 20 feet in diameter.

Note that the detection range for craters and protuberances decreases rapidly as sun angle increases. Protuberances show a greater detection range than craters at low sun angles, due to their large shadows. Note also that an increase in viewing angle results in an increase in detection range and that detection ranges approach zero for sun angles greater than the view angle. The loss of detection range is due to a severe decrease of contrast related to slope variations for sun angles higher than the viewing angle and to the loss of shadows.

Fig. 9 shows the effect of an azimuthal difference between the viewing angle and incidence angle. For low viewing angles, an azimuthal change is beneficial; however, for high viewing angles, an azimuthal change is somewhat detremental. Thus, objects will be more detectable when the viewing angle is low, if one looks to the side of a zero phase washout area.

With regard to the possibility of the sun being in front of the viewer, these data were extended for a complete range of sun angles. Results for a 14° viewing angle are shown is Fig. 10 for sun angles from 5° to 175°. The analysis carried out indicates that the detection range does not reach an appreciable magnitude, after the initial decrease to zero, until an angle of about 150° is reached; i.e., 30° above the horizon in front of LM. However, the values shown for the sun in front do not include the effect of glare, which seriously degrades visibility.

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LUNAR REFLECTANCE PROPERTIES

- LUNAR SURFACE ACTS AS A RETRO-REFLECTOR
  - SURFACE BRIGHTNESS DEPENDS ON VIEW & SUN ANGLES
  - LUMINANCE REACHES MAXIMUM AT SUN ANGLE
  - WASHOUT PHENOMENA OCCURS AT ZERO PHASE ANGLE
- ALBEDO VARIATION IS SMALL
  - FROM 0.065 (MARIA) TO 0.13 (BRIGHTEST RAYS)
- CONTRAST DEPENDS ON LOCAL SLOPE VARIATIONS AND SHADOWS
- MEASUREMENTS OF REFLECTANCE PROPERTIES ARE EARTH-BASED AND OF LOW RESOLUTION

Fig. 7

NASA-S-66-6125 JUN

DETECTION RANGE

PROTUBERANCES & CRATERS

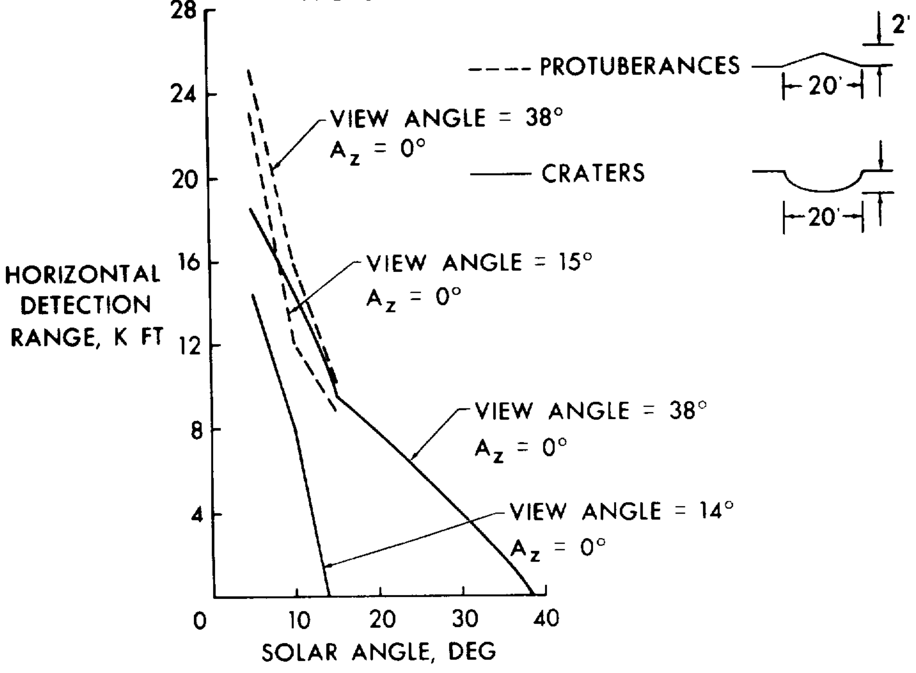

Fig. 8

2.1.3 GLARE AT FORWARD SUN ANGLES

When the sun shines directly in the LM window, a veiling luminance results which obscures the lunar surface (Fig. 11). The main component of the veiling luminance is the scattering at the window due to deposits from the adjacent RCS motor. A smaller component results from scattering in the eye. As can be seen from Fig. 12, the veiling luminance may be five to six times as great as the lunar background luminance. This results in a large reduction in scene contrast and precludes consideration of low forward sun angles.

2.1.4 SHADOWING AND RECOGNITION

Shadows are desirable for the enchancement of object recognition, as well as for detection. The maximum sun angle for shadowing is shown in Fig. 13 for typical crater configurations. The 10:1 diameter to depth ratio configuration is the most typical observed and corresponds to sun angles of 22° or less for shadowing. Sun angles of 22° or less would result in shadowing from at least 60% of the craters.

2.1.5 SHADOWING AT LOW SUN ANGLES

For very low sun angles, a substantial portion of each crater is covered by its shadow and a general decrease in luminance occurs. Acceptable landing areas with a slope of a few degrees could be completely contained within a shadow when the sun is at a low angle. Hence, a lower limit must be placed on the sun angle. This limit is presently set at 7° (Fig. 14).

2.1.6 VISIBILITY RESULTS

The results of the visibility studies insofar as they affect the relationship between the viewing angles and the sun angles are shown in Fig. 15.

2.2 SELECTION OF MAXIMUM SUN ANGLES

The visibility results have established the relationships between viewing angles and sun angles. The results must be related to the LM viewing angle history to arrive at a maximum and minimum sun angle. Fig. 16 shows viewing angle histories as a function of range from the landing site for the lunar descent trajectory developed in an earlier presentation.

The maximum view angle possible is defined by the lower window limit, this angle varies from 25° at high gate to approximately 50° in the vicinity of the landing site. The view angles to the landing site itself are seen to vary from 14° at high gate to approximately 38° in the vicinity of the landing site.

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DETECTION RANGE

CRATERS

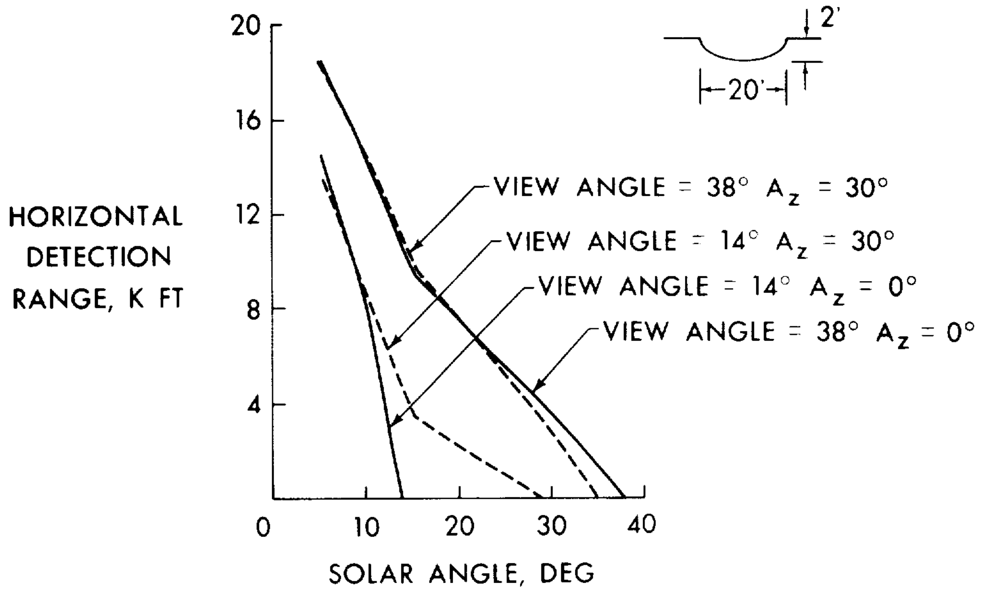


Fig. 9

NASA-S-66-3879 JUN

DETECTION RANGE AS A FUNCTION OF SOLAR ANGLE

(UNCORRECTED FOR GLARE EFFECT)

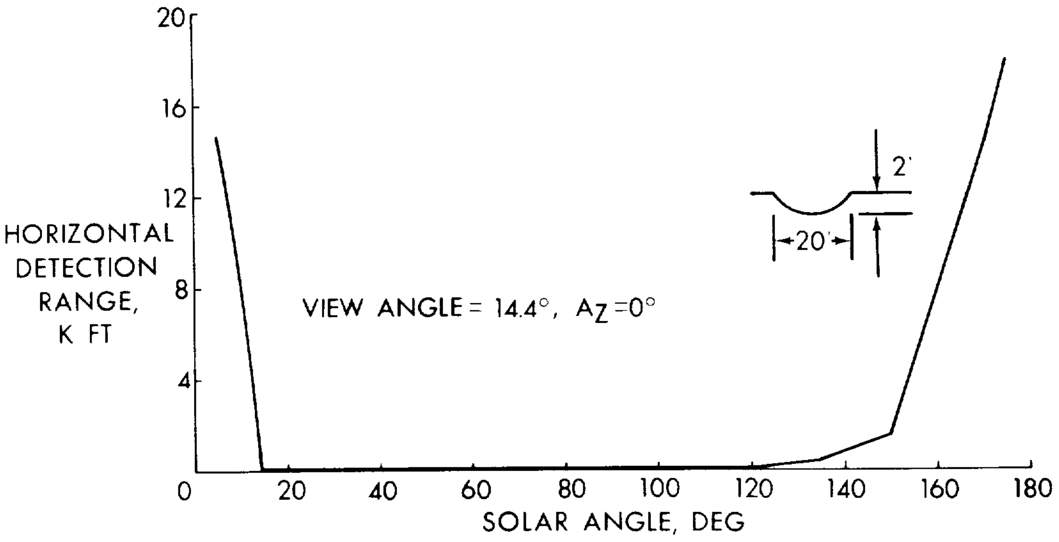
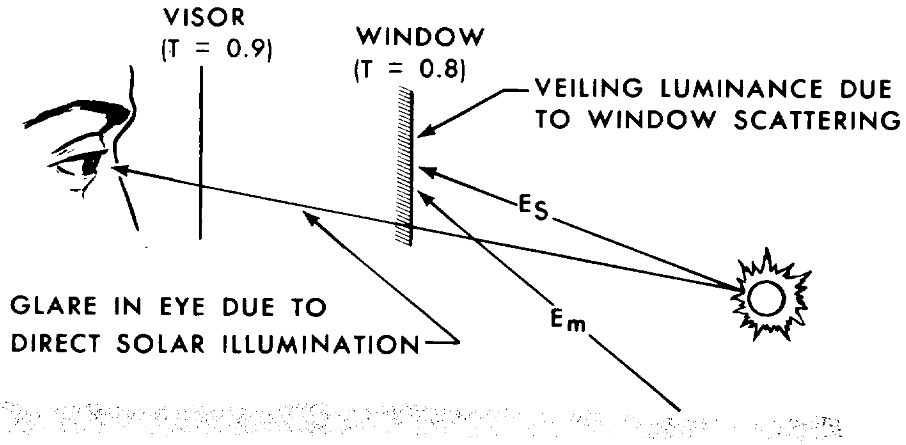


Fig. [illegible]

NASA-S-66-5946 JUL 1

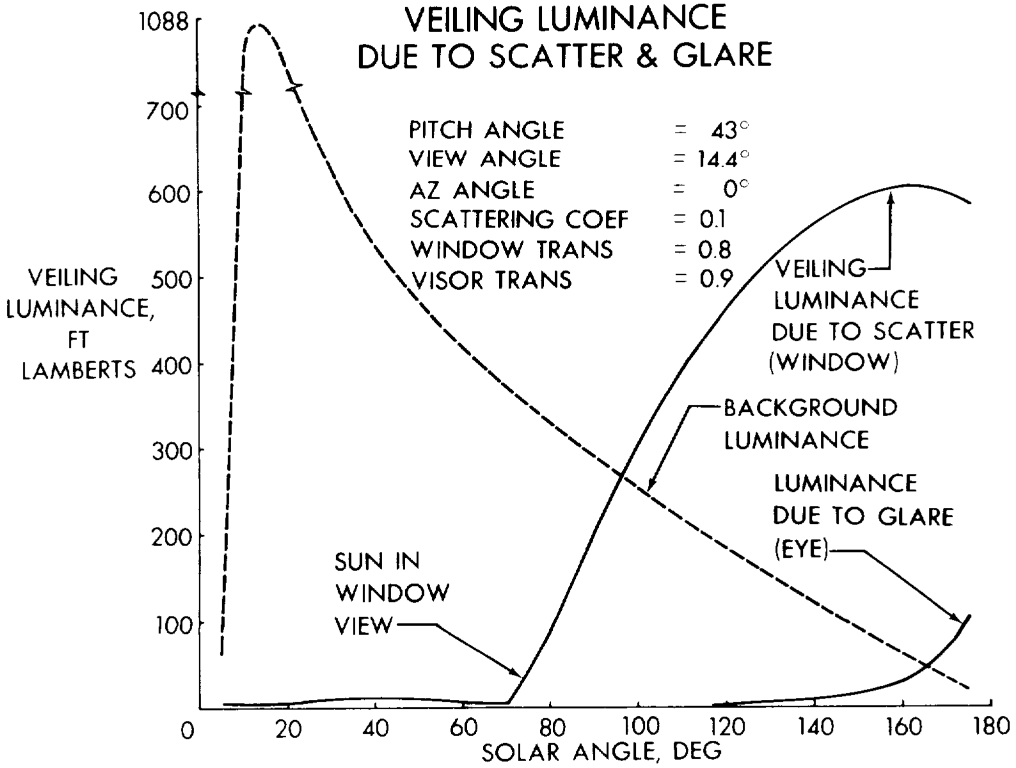
SOURCES OF VEILING LUMINANCE


VISOR
(T = 0.9)
WINDOW
(T = 0.8)
VEILING LUMINANCE DUE TO WINDOW SCATTERING
E_S
GLARE IN EYE DUE TO DIRECT SOLAR ILLUMINATION
E_m

E_S = SOLAR ILLUMINATION ON LM WINDOW
E_m = LUNAR ILLUMINATION ON LM WINDOW

Fig. 11

NASA-S-66-3882 JUN



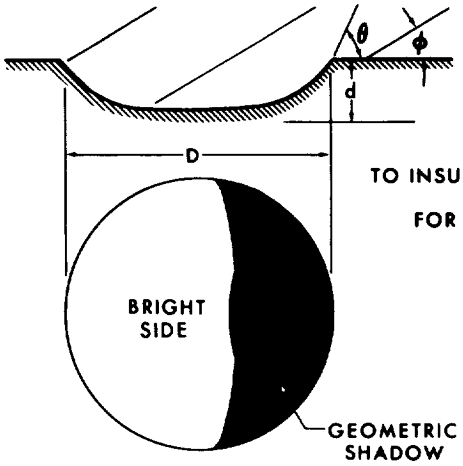
VEILING LUMINANCE
DUE TO SCATTER & GLARE

PITCH ANGLE = 43°
VIEW ANGLE = 14.4°
AZ ANGLE = 0°
SCATTERING COEF = 0.1
WINDOW TRANS = 0.8
VISOR TRANS = 0.9

Fig. [illegible]

NASA-S-66-5417 MAY 31

CRATERS



TO INSURE SHADOWS (SPHERICAL SEGMENTS)

FOR $\frac{D}{d} = \frac{12}{1}$, $\theta_{MAX} \approx 18.5° \geq \phi_{MAX} \leq 18.5°$

$\frac{D}{d} = \frac{10}{1}$, $\theta_{MAX} \approx 22.6° \geq \phi_{MAX} \leq 22.6°$

$\frac{D}{d} = \frac{8}{1}$, $\theta_{MAX} \approx 28.6° \geq \phi_{MAX} \leq 28.6°$

ϕ = SUN ANGLE
θ = CRATER SLOPE ANGLE

Fig. 13

NASA-S-66- 6812 JUN

MINIMUM SUN ANGLES

- EXTREME SHADOWING & LOW BRIGHTNESS OCCUR AT SUN ANGLES LESS THAN 5 - 7°
- VARIATIONS FROM EXPECTED SLOPE & TERRAIN MAY RESULT IN ENTIRE AREA COVERED BY SHADOWS

Fig. 14

NASA-S-66-6120 JUN

VISIBILITY RESULTS

- DETECTION RANGE DECREASES RAPIDLY AS SUN ANGLE INCREASES
- AT ZERO AZIMUTH, VIEW ANGLE MUST BE ABOVE SUN ANGLE FOR NON-ZERO DETECTION RANGE
- AZIMUTH CHANGE ON THE ORDER OF 30° IS BENEFICIAL ONLY FOR SUN ANGLES GREATER THAN THE VIEWING ANGLE
- FOR A FIXED SUN ANGLE, AN INCREASE IN VIEW ANGLE IS BENEFICIAL
- GLARE AT LOW SUN ANGLES IN FRONT OF THE LM PRECLUDES THAT CONDITION
- WASHOUT AT ZERO PHASE ANGLE RESULTS IN A DEAD BAND ABOUT THE VIEWING ANGLE

Fig. 15

NASA-S-66-6510 JUN

VIEW ANGLES

VISIBILITY PHASE

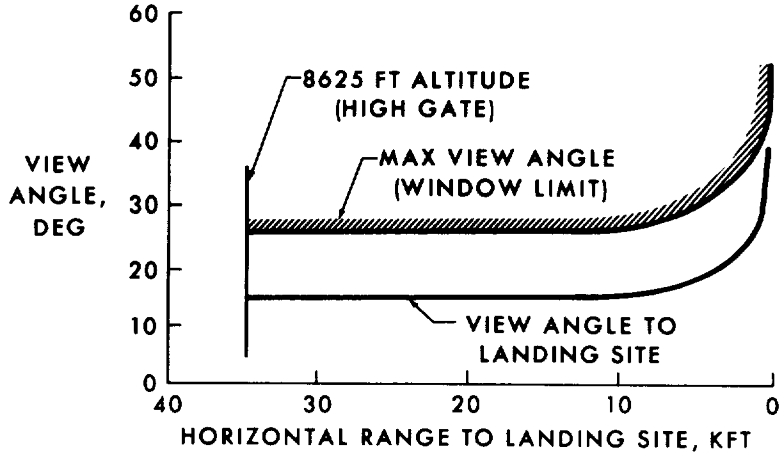


Fig. [illegible]

NASA-S-66-6533 JUN

MAXIMUM SUN ANGLES

VISIBILITY PHASE

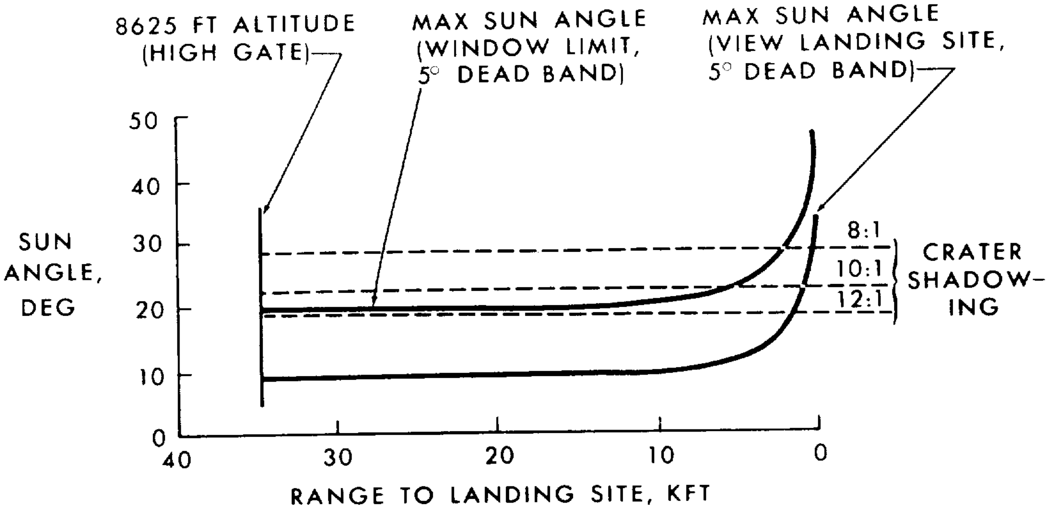


Fig. 17

As discussed earlier, the sun angle must be less than the viewing angle and also when the sun angle and the viewing angle are within a few degrees of being coincident, washout occurs. Thus, the maximum sun angles are taken to be 5° less than the viewing angle to accommodate the detection requirements and to accommodate an assumed 5° washout deadband.

Fig. 17 presents the sun angle histories corresponding to the maximum viewing angles and the view angles to the landing site. Also presented are the maximum sun angles for initiation of crater shadowing. Note that a maximum sun angle of 20° allows continuous viewing of the lunar surface from high gate, with favorable viewing conditions of:

1. View angle at least 5° greater than sun angle.

2. Crater shadows for most craters.

Also, the 20° maximum sun angle provides a view of the landing site under these favorable viewing conditions from a range of at least 1,000 ft., and at the phase in the landing trajectory of greatest importance; i.e., immediately prior to landing. Lower sun angles would increase the range of favorable viewing conditions even more.

For these reasons, a maximum value for the sun angle has been set at 20° and the minimum value set at 7° for the lunar landing site selection criteria. These values will change if newer information so indicate. The study program on which these data are based is not complete. Analysis of simulation studies on detection and recognition and of Surveyor and Orbiter photographs will assist in the evaluation of the current choice of lighting conditions.

2.3 CSM PERFORMANCE

2.3.1 ACCESSIBLE LUNAR REGION

The sun angle range of 7° to 20° will now be considered in conjunction with the CSM performance to establish the accessible lunar region and hence, establish the region which will be considered in the site selection process. The details relating to the CSM performance have been developed in a previous paper.

Fig. 18 shows the accessible lunar landing area when lighting conditions are correct for a typical month (February 1968). The accessible lunar area for the year of 1968 is shown on Fig. 19. Based on these typical data, the latitude boundaries for the lunar area of interest for site selection have been taken to be ±5°.

The longitude boundaries are extablished by tracking and communications considerations. The Eastern boundary is set at 45° longitude to satisfy powered descent requirements. The Western boundary is set -45° longitude to satisfy ascent requirements.

2.3.2 LANDING SITE SPACING

The sun angle range of 7° to 20° results in a one day launch opportunity for a given sity as noted on Fig. 20. This implies that the landing sites must be spaced in a manner which is dependent on the launch philosophy.

A consecutive day launch window will require, for example, separate sites which are spaced in longitude 10 ±2°. For an every-other-day launch philosophy, the sites must be spaced approximately 23 ±3° in longitude, and so on.

2.4 SUMMARY OF OPERATIONAL CONSIDERATIONS

The operational considerations which make up part of the site selection criteria are summarized in Fig. 21. The key items are the definition of the lunar region which will be considered in the site selection process, the fact that the sun angle range of 7° to 20° results in only a one day launch opportunity per landing site, and that landing sites must be spaced in longitude in a manner which is dependent on the launch philosophy.

3.0 SPACECRAFT/SURFACE INTERACTIONS

The second category of considerations which is included in the site selection criteria is the interaction between LM and the lunar surface. As noted in Fig. 22, the spacecraft/surface interactions apply to the landing approach and the landing.

3.1 LANDING APPROACH CONSIDERATIONS

The landing approach considerations relate to the interactions of the landing radar with the lunar surface. The portion of the descent trajectory where this interaction takes place is discussed below.

3.1.1 LM DESCENT TRAJECTORY

The portion of the descent trajectory where the landing radar interacts with the lunar surface is from initiation of landing radar altitude update to landing. As noted on Figures 23 and 24, this phase of the flight begins when the LM is 30 n.m. uprange of the landing site and at an altitude of approximately 25,000 feet.

NASA-S-66-5179 JUN 13

ACCESSIBLE LUNAR LANDING AREA IN FEB 1968 WHEN THE LIGHTING CONDITIONS ARE CORRECT

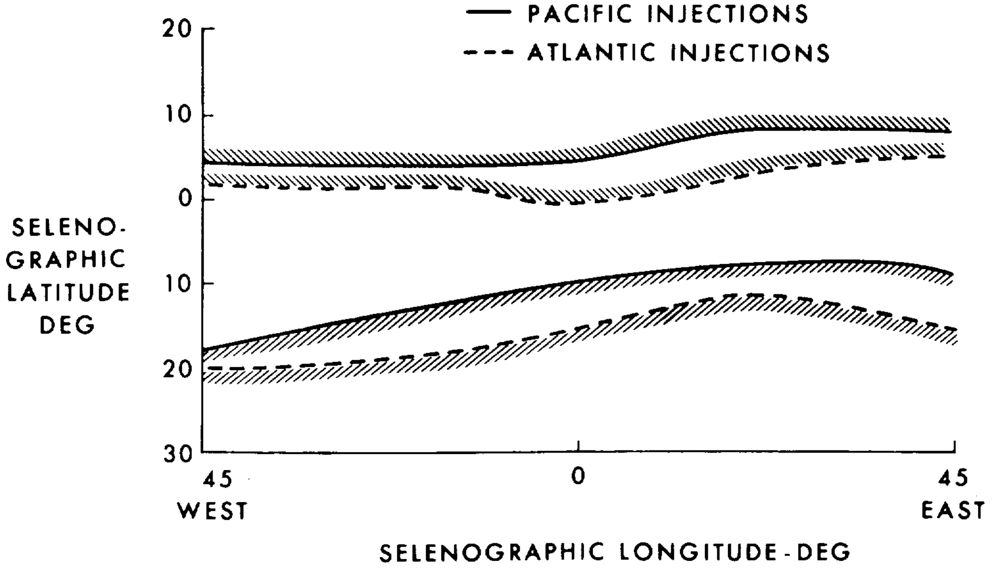


Fig. 18

NASA-S-66-5182 JUN

100% LUNAR SURFACE ACCESSIBILITY FOR 1968

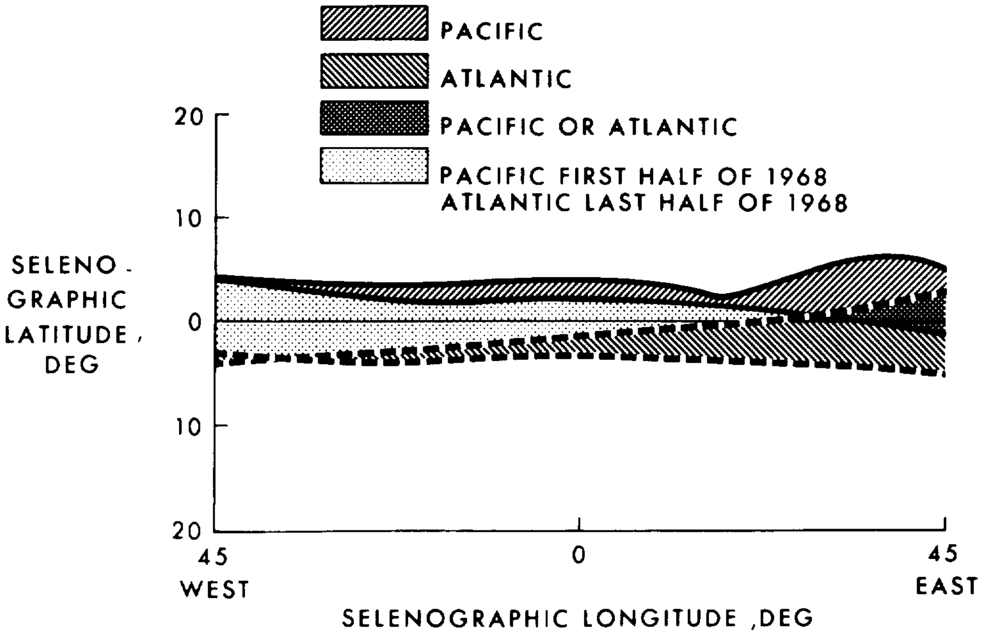


Fig. 19

NASA-S-66-5021 MAY 31

LUNAR LANDING OPPORTUNITIES

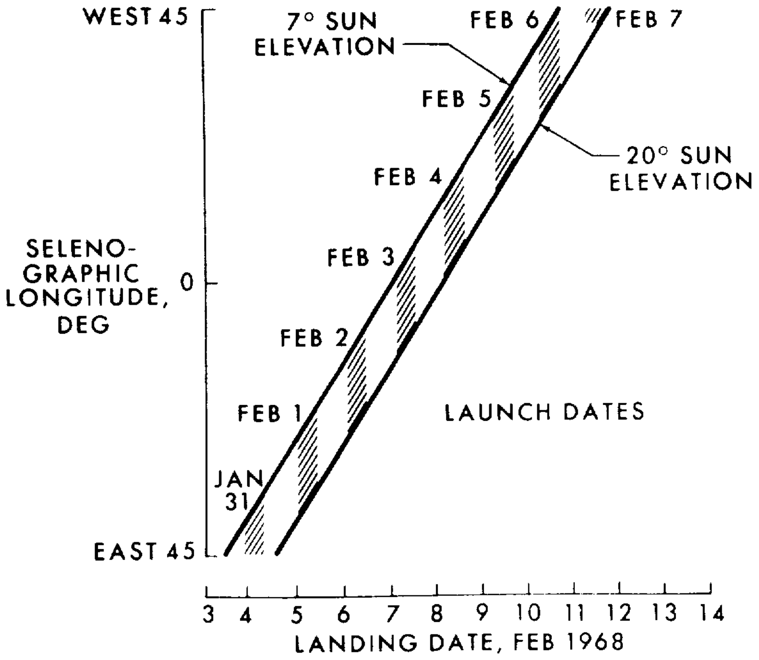


Fig. 20

NASA-S-66-5053 JUNE

SLIDE 21

SUMMARY OF OPERATIONAL CONSIDERATIONS

- DEFINITION OF ACCESSIBLE AREA
  - LONGITUDE: 45° W TO 45° E
  - LATITUDE: 5° N TO 5° S
  - LANDING SITES IN CLUSTERS WITH INDIVIDUAL SITES DISTRIBUTED LONGITUDINALLY WITH A 10° ± 2° OR 23° ± 3° INTERVAL BETWEEN SITES
- LIGHTING CONSIDERATIONS
  - SUN ANGLE IN THE RANGE 7° TO 20°
  - VIEW ANGLE GREATER THAN SUN ANGLE AT RANGE WHERE OBJECT DETECTION AND RECOGNITION CAPABILITY IS REQUIRED

Fig. 21

The use of the landing radar during the last 30 n.m. results in certain restrictions being imposed on the lunar surface in this region. These restrictions are discussed below.

3.1.2 TOPOGRAPHIC RESTRICTIONS IMPOSED BY LANDING RADAR

The restrictions imposed by the landing radar on the approach topography of the lunar surface are presented in Fig. 25. The restrictions are that the mean slope not exceed $\pm 2^\circ$ over the last 30 n.m. to touchdown. In addition, the deviation due to local surface effects about the mean slope must be less than or equal to ±5% of the nominal LM attitude. These topographic restrictions must be satisfied throughout an approach path ray which is 30 n.m. long and varies in width from 16,000 feet at the landing site to approximately 8.6 n.m. at a distance of 30 n.m. from the landing site. If the topography of the landing approach ray exceeds the restrictions described, then the commanded attitudes and commanded throttle ratios of LM can become excessive. This can result in excessive LM attitude excursions which would cause the landing radar to loose lock.

The inclination of the landing radar approach ray varies as a function of lunar latitude and longitude in such a manner that the angles η_1 and η_2 noted on the chart, possess the following properties:

$$\eta_1 + \eta_2 = 20^\circ$$

$$4^\circ \leq \eta_1 \leq 16^\circ$$

$$4^\circ \leq \eta_2 \leq 16^\circ$$

This variation is due to the change in spacecraft orbital inclination as a function of landing site position.

3.2 LANDING CONSIDERATIONS

The landing considerations relate to the size of the landing area and the requirements imposed on the lunar topography and the soil characteristics within the landing area. The requirements on topography and soil characteristics are defined by LM landing dynamics and stability considerations.

3.2.1 LANDING SITE SIZE

The landing area, for site selection purposes, corresponds to the 3-sigma dispersion ellipse of the LM at landing. The

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SPACECRAFT/SURFACE INTERACTION

- APPROACH CONSIDERATIONS
- LANDING CONSIDERATIONS

Fig. [illegible]

NASA-S-66-5418 MAY 31

LM POWERED DESCENT

ENGINE IGNITION
h=50,000 FT
θ=86°
T=0 SEC
V=5500 FT/SEC
γ=0°

MAXIMUM THROTTLE
h=50,000 FT
θ=88°
T=28 SEC
V=5564 FT/SEC
γ=−.2°

TARGET SWITCHOVER
h=43,000 FT
θ=80°
T=228 SEC
V=3385
γ=−1.4°

LR ALTITUDE UPDATE
h=25,000 FT
θ=71°
T=328 SEC
V=2164 FT/SEC
γ=−4.0°

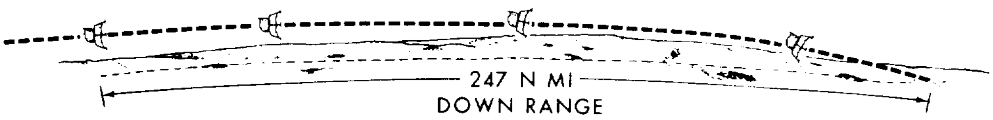


Fig. [illegible]

NASA-S-66-5414 MAY 31

LM POWERED DESCENT (CONT)

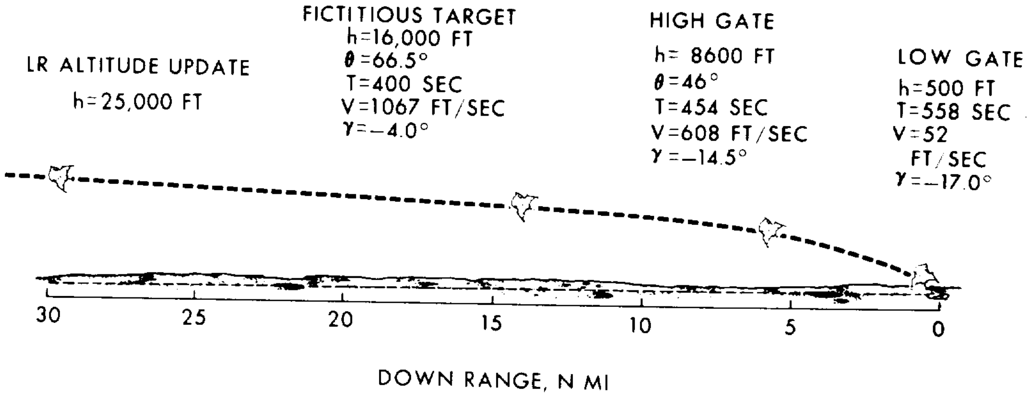

FICTITIOUS TARGET
h=16,000 FT
θ=66.5°
T=400 SEC
V=1067 FT/SEC
γ=–4.0°
HIGH GATE
h= 8600 FT
θ=46°
T=454 SEC
V=608 FT/SEC
γ=–14.5°
LR ALTITUDE UPDATE
h=25,000 FT
LOW GATE
h=500 FT
T=558 SEC
V=52 FT/SEC
γ=–17.0°
30
25
20
15
10
5
0
DOWN RANGE, N MI

Fig. 24

NASA-S-66-5233 JUN

LM APPROACH RAY

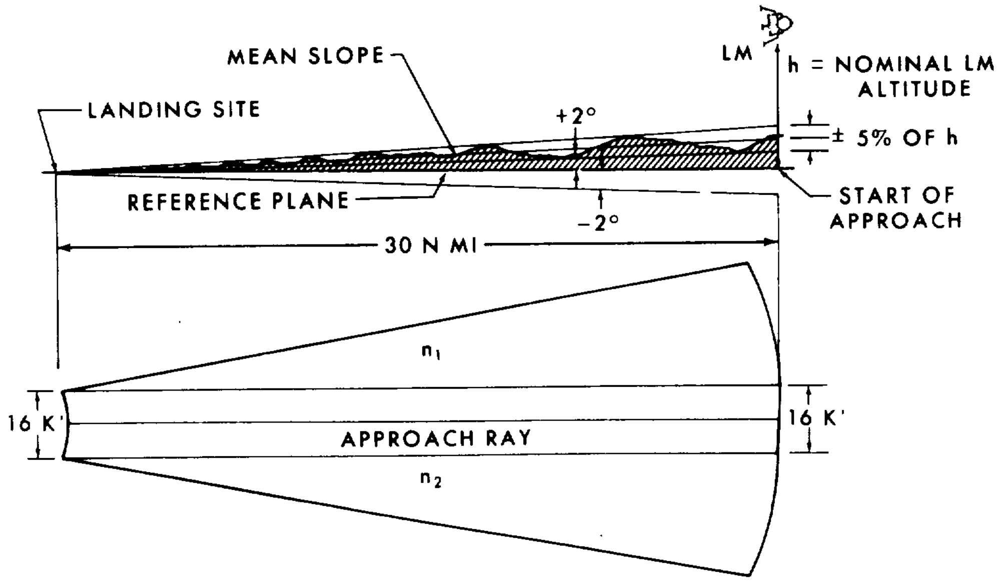

MEAN SLOPE
LM
h = NOMINAL LM ALTITUDE
LANDING SITE
+2°
± 5% OF h
REFERENCE PLANE
-2°
START OF APPROACH
30 N MI
n_1
16 K'
APPROACH RAY
16 K'
n_2

Fig. 5

dimensions of the landing dispersion ellipse were developed in a previous paper and are repeated on Fig. 26. As noted, the landing area has a major axis of 52,000 feet and a minor axis of 10,800 feet.

This landing dispersion ellipse is based on MSFN tracking for three orbits and does not include site location uncertainties. This ellipse corresponds to the guidance philosophy which allows range to be a free parameter.

3.2.2 LANDING SITE RESTRICTIONS IMPOSED BY LANDING DYNAMICS/STABILITY CONSIDERATIONS

The characteristics of the lunar topography and the soil within the landing area (3-sigma dispersion ellipse) are defined by landing dynamics and landing stability considerations. A previous paper described the studies carried out by MSC in this area and established surface requirements.

The results of the analysis previously described indicated that the maximum allowable penetration of the lunar surface by LM is approximately 44 inches. Based upon this value, it has been determined that for a surface with 24 inch deep holes the surface slope must not exceed 7°. The corresponding static bearing stress must be greater than 8 psi or equivalently, the bearing stress must be greater than 5 psi per foot of penetration. For a surface devoid of holes, the surface slope must not exceed 13 degrees and the bearing stress must be greater than 4 psi. In addition, the soil coefficient of friction must be ≤ 0.4 to prevent excessive sliding.

Fig. 27 summarizes these slope and soil requirements.

4.0 SITE SELECTION CRITERIA

The site selection criteria is summarized on Figures 28 and 29. This criteria consists of all the operational considerations and spacecraft/surface considerations discussed herein. This criteria represents the most realistic requirements as presently understood and will be updated as the requirements change.

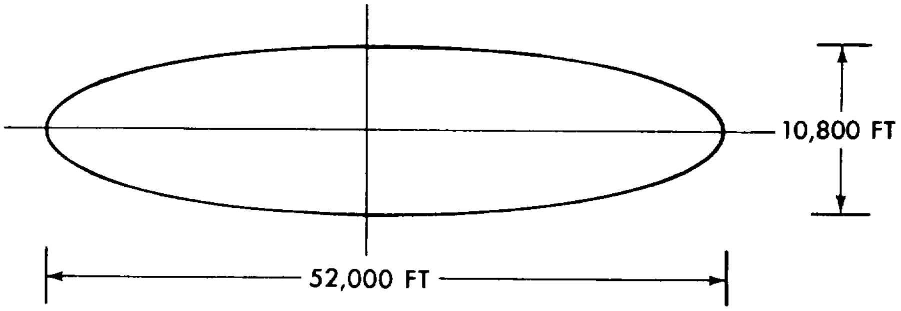
As stated in the Introduction, the site selection criteria will be used in conjunction with the available lunar surface data to provide the mechanism for selecting candidate landing sites.

The next discussion entitled, "Site Selection Data Sources and Interpretation" will elaborate on the lunar surface data and how it will be used for selecting candidate landing sites.

NASA-S-66-5945 JUL

LANDING SITE SIZE

3-SIGMA LANDING DISPERSION ELLIPSE



- RANGE FREE
- MSFN ONLY - 3 ORBITS
- NO SITE UNCERTAINTIES

Fig. [illegible]

NASA-S-66-6864 JUNE

LANDING DYNAMICS/STABILITY CONSIDERATIONS

TO PRECLUDE EXCEEDING MAXIMUM ALLOWABLE SURFACE PENETRATION OF 44 INCHES

- FOR SURFACE WITH 24 INCH DEEP HOLES
  - SURFACE SLOPE MUST BE ≤ 7 DEGREES
  - SOIL BEARING STRESS MUST BE:
    1. ≥ 8 PSI (CONSTANT) OR
    2. ≥ 5 PSI PER FOOT OF PENETRATION
- FOR SURFACE DEVOID OF HOLES
  - SURFACE SLOPE MUST BE ≤ 13 DEGREES
  - SOIL BEARING STRESS MUST BE ≥ 4 PSI (CONSTANT)

TO PREVENT EXCESSIVE SLIDING, SOIL COEFFICIENT OF FRICTION MUST BE ≥ 0.4

Fig. [illegible]

NASA-S-66-6811 JUN

SITE SELECTION CRITERIA

DEFINITION OF ACCESSIBLE AREA

- LONGITUDE: 45° W TO 45° E
- LATITUDE: 5° N TO 5° S
- LANDING SITES IN CLUSTERS WITH INDIVIDUAL SITES DISTRIBUTED LONGITUDINALLY WITH A 10° ± 2° OR 23° ± 3° INTERVAL BETWEEN SITES

LIGHTING CONSIDERATIONS

- SUN ANGLE IN THE RANGE 7° TO 20°
- VIEW ANGLE GREATER THAN SUN ANGLE AT RANGE WHERE OBJECT DETECTION & RECOGNITION CAPABILITY IS REQUIRED

Fig. 28

NASA-S-66-5404 MAY 31

SITE SELECTION CRITERIA (CONT)

TOPOGRAPHY

- LANDING RADAR AND GUIDANCE CONSIDERATIONS
  - MEAN SLOPE OF ≤ ±2° OVER LAST 30 N MI
  - LOCAL TERRAIN DEVIATIONS (ABOUT MEAN SLOPE) ≤ ±5% OF NOMINAL LM ALTITUDE
  - APPROACH RAY: 16,000 FT WIDE AND INCLINATION VARIES AS A FUNCTION OF LONGITUDE AND LATITUDE
- LANDING ELLIPSE GEOMETRY
  - BASED ON MSFN WITH NO SIGHTINGS, THE 3σ DISPERSION ELLIPSE DIMENSIONS ARE: 52 K FT BY 10.8 K FT WITH RANGE FREE
- LM STABILITY
  - WITH HIGH CONFIDENCE, 95% OF THE POSSIBLE RANDOM LM LANDINGS ON THE TOPOGRAPHIC SURFACE WITHIN THE 3σ DISPERSION ELLIPSE CAUSE THE LM TO TILT ≤7° WITH RESPECT TO THE LOCAL VERTICAL. THE REGION SHALL ALLOW AVOIDANCE OF BOTTOMING HAZARDS DUE TO PROTUBERANCES > 24'' IN HEIGHT

SOIL MECHANICS

- COEFFICIENT OF FRICTION: ≥0.4
- SOIL BEARING STRENGTH ≥8 PSI TO PRECLUDE LM SINKAGE OF >44 INCHES

Fig. 29

Questions and Comments

SITE SELECTION CRITERIA

Speaker: A. V. Bernard

1. Comment by Mr. Milwitzky - Although the photometric function (as reviewed in this presentation) indicates that a certain range of sun angle is highly desirable (and the Surveyor data generally confirm this), the Surveyor data have shown that the degradation of visibility is not as bad at the higher sun angles as was indicated in the presentation. Further study will be made on this subject; however, there is no question that the low sun angle is best.

2. Dr. Shoemaker - What were the main contributors to scattering, the window or the helmet? How were the scattering data established?

   ANSWER - Only the window scattering was considered, not the helmet. A rather pessimistic scattering coefficient was used, assuming that the RCS had left deposits on the LM window. The scattering coefficient assumed was 0.1.

3. E. Stern - Why does the landing radar limit the acceptable slope to 2 degrees?

   ANSWER - The consideration is one of excursions in LM attitude. As the slope and the local deviations exceed those presented in the criteria, it would probably result in excessive attitude rates or excursions from the nominal attitude.

4. How did the assumption for the window degradation compare with the experience in Gemini?

   ANSWER - The GT-9 flight crew observed that particles from the RCS from the other vehicle came back and impacted the windows like raindrops. The actual numerical value for the degradation is not known at this time.

5. Comment by Dr. Von Braun - The shadow of the LM with the sun behind it could be used as an aid to the crew in surface contour identification, for general observations, and for use as an altimeter during the final landing phase.

   ANSWER (O. Maynard) - Mr. Loftus has been looking into the potential value of the shadow as an aid to the crew. A cursory review of the analysis to date indicates that the shadow is not of much use to the pilot for assessment of the terrain contour or altitude until the last portion of the descent at which time there are some useful altitude cues. The study will be available early in August. We do plan to perform flight tests which seem promising and use the phenomenon if appropriate.

LUNAR LANDING SITE DATA SOURCES AND ANALYSIS

by

John E. Dornbach

LUNAR LANDING SITE DATA SOURCES AND ANALYSIS

1.0 INTRODUCTION

Information from earth-based studies and from Surveyor and Luna spacecraft indicates that Apollo landing sites can be found within areas to be photographed by Lunar Orbiter. Analysis of lunar data in relation to Apollo Spacecraft and operational constraints has shown that favorable sites can be located, so as to provide for maximum mission flexibility. The favorable areas are generally clustered around 30° East longitude, near 0°, and 35° West longitude.

This portion of the program will concentrate on lunar surface data as gathered from Earth or from unmanned spacecraft and how it is being used in the Apollo landing site analysis program at MSC.

2.0 EARTH-BASED STUDIES

FIGURE 1 - LAC 1:1,000,000 SCALE COVERAGE

At the present time most of the visible face of the Moon has been mapped for the NASA by the U.S. Air Force Aeronautical Chart and Information Center. These charts at sixteen miles to the inch are about the same scale as the best you could get today for covering parts of Antarctica, Australia, and Africa. By carefully measuring on observatory photographs, the lengths of shadows cast by craters at low sun angles, we are able to estimate one thousand foot contours on the lunar surface. This slide illustrates that at this scale charts of the lunar equitorial region, or the region of interest for early Apollo missions, have already been completed. The area covered by one of these charts is shown by the outline of the Kepler chart.

FIGURE 2 - USGS PHOTOGEOLOGICAL MAP

At the present time professional geologists of the U.S. Geological Survey are using the Air Force charts as base maps on which they are plotting terrain and stratigraphic relationships. These geologists headquartered at the USGS Center of Astrogeology in Flagstaff, Arizona, spend many hours at the telescope to provide such maps as this one of the region surrounding the crater, Kepler. The various colors indicate materials such as mare, ray, impact ejecta, and volcanic. The Apollo area of interest has already been geologically mapped at this scale.

FIGURE 3 - AIC 1:500,000 SCALE COVERAGE

Within the past year, the Air Force Chart Center has also been asked by the NASA to produce charts at a scale of eight miles to the inch concentrating on the equatorial area of immediate Apollo interest. These charts carry the maximum amount of information that can be gained visually or photographically through the finest earth-based observatory telescopes. They are concentrating their viewing efforts in detailing the flatter mare areas, where landing sites may most probably be found. These charts are keyed to the 1:1,000,000 scale series as may be seen on the Kepler sheet.

FIGURE 4 - THERMAL ANOMALIES

The cartographers and geologists have mapped from Earth, how the Moon varies from place to place in the visible spectrum. As the wavelength of instrumental observation is increased into the infrared, we see that around 10 microns wavelength, the Moon begins to demonstrate anomalous characteristics. This slide by Shorthill and Saari illustrates that during an eclipse, as the source of energy - the sun - is quickly removed, certain portions of the lunar surface retain heat longer than others. From this we might infer that the areas retaining heat are rocky or covered by more dense material, whereas those which quickly cool could be covered with a less dense or highly insulating material.

FIGURE 5 - RADAR MAPPING

The NASA is also conducting studies with the Lincoln Laboratories of Massachusetts Institute of Technology in another portion of the spectrum. By examining the Moon in the centimeter and meter radar wavelength regions it should also be possible to infer differences from place to place as small as several square miles. From an analysis of the returned signals, scientists expect to learn something of the density of materials making up the upper lunar matle and the average roughness of the surface from place to place. By combining all data gathered from Earth with knowledge gained from discrete samples from Surveyor and Luna, we feel confident that satisfactory landing sites can be located on the Orbiter photography.

FIGURE 6 - APOLLO ZONE OF INTEREST

The Apollo Area of interest has been defined as forty-five degrees East to forty-five degrees West longitude and from five degrees North to five degrees South latitude. NASA has already photographically explored part of this region as a result of the Ranger Project and soon the unmanned Orbiter A spacecraft will supply detailed photographic information for Apollo landing site selection. The surface has been photographed at two locations - by Surveyor I and Luna IX, but only Surveyor was in the Apollo Zone of Interest. Data from Surveyor has indicated that the soil bearing strength would be acceptable for the Lunar Module in this area. Future Surveyors will sample some of the other areas indicated by circles on this chart.

3.0 RANGER DATA ANALYSIS

FIGURES 7 AND 8 - RANGER IX AND LUNAR SLOPES AT LM SCALE

The highest resolution photographs taken during the Ranger flights came from Ranger IX. The last few P frames as seen on this figure provided surface resolution less than one foot. MSC has conducted a computer analysis of photometrically derived slopes on these photographs. Data from these programs indicates that more than 90 percent of the slopes within the area of the last few photographs were less than 10°. It was only on those photographs where the large two hundred foot crater was located, that a cumulative slope frequency diagram would indicate a potential problem in landing. With the astronauts performing the lunar landing, a crater such as this could easily be avoided.

FIGURES 9 AND 10 - RANGER VIII APOLLO SITE ANALYSIS AND LM LANDABILITY

These two slides illustrate how our computer analysis of lunar photography is being used to determine Lunar Module topographic landability. The Jet Propulsion Laboratory has provided a magnetic tape containing the digital information as received from Ranger VIII and a computer program to remove extraneous noise from the photography and to rectify each image. On this figure, the photo on the left illustrates a composite map of slope and protuberance values derived from the photograph.

The graphs on figure 10 were computed from the Ranger VIII photography and illustrate numerically that which is shown on the map on the previous figure. We can see that from our analysis of this photograph, the Lunar Module would have encountered slopes of more than six degrees over only twenty-five percent of the area and protuberances of more than two feet over less than five percent of the area.

FIGURES 11 AND 12 - RANGER IX PHOTOGRAPH OF FLOOR OF ALPHONSUS AND GEOLOGICAL MAP

This photograph and map cover an area of about 20 x 30 km (about 240 sq. miles) on the north-east floor of the crater Alphonsus. Very evident on the photograph are geologic features such as young craters, old or worn-down craters, rilles, chain craters, and others. As our knowledge of the lunar surface improves, we will be able to draw much more detailed maps. The geological map was prepared by the USGS and is included in the JPL report on Rangers VIII and IX. We anticipate that as a result of Surveyor and Orbiter, much more detailed and precise geologic maps will be prepared for use in Apollo landing site analysis. Large scale geologic maps will also aid in mission planning and in lunar surface exploration.

FIGURE 13 - KUIPER BEARING STRENGTH ANALYSIS

By formulating concepts for the evolution and morphology of features seen on the Ranger photographs, it is possible to estimate some physical properties. Dr. Gerard Kuiper of the University of Arizona and Principal Investigator for Ranger photography established several basic assumptions and from them, estimated the bearing strength of the cratered surface. He examined many protuberances, indicated on this Ranger IX photograph by small arrows, and surmised that they were materials ejected from the 150 ft. primary crater at the top of the photo. The materials measured at the 10° sun angle appeared to be about 3 feet across and about a foot high. Assuming they were ejected from the primary crater to where they are found and that their bulk density was about 2, he concluded that most of the protuberances were probably half squashed and half buried. From this he concluded the bearing strength to be over 14 pounds per square inch averaged for the upper one to two feet of this part of the lunar surface.

4.0 LUNAR SURFACE DATA

A. LUNA IX

FIGURES 14 AND 15 - LUNA IX PANORAMIC SKETCH AND ANALYSIS OF CLOSE-UP PHOTOGRAPHY

The soft landing on the lunar surface by Luna IX provided the first substantive information that a man-made vehicle could land, transmit information to Earth, and not slowly sink into a morass of dust. In fact, the spacecraft and optical surfaces showed a remarkable absence of fine particles or dust. This sketch is a composite of much of the information which has been interpreted from analysis of the photography. By locating conjugate images on two mirrors and a direct photo image, we were able to determine the distance to the materials photographed. From this distance we could determine that the size of particles 3 to 5 feet from the spacecraft were about 2/10" in size. The larger objects are 10" - 12". The crater on the left horizon is approximately 50 feet in diameter.

I have included this figure to illustrate some of the problems we encounter when working with photography taken close to the surface. We assume the Luna IX spacecraft camera to have been only about 22 inches or less above the lunar surface. Due to the small mirror base of about 12", reliable measurements are limited to a distance between about 3 to 20 feet.

The Surveyor TV camera provided photographs from about five and a half feet above the lunar surface. Also, a more precise knowledge of the Surveyor spacecraft, landing dynamics, camera calibration, and orientation, should provide much more quantative information than Luna IX.

In this Luna photograph we have indicated the location of the mirror images and some of the scaling measurements we have made.

B. SURVEYOR I

FIGURES 16 AND 17 - SURVEYOR I SPACECRAFT SHADOW

The mosaic figure illustrates lunar lighting conditions with a 10° sun angle looking toward the 0° phase angle point (directly along the sun line, away from the sun). The spacecraft shadow is about 35' long. The detailed texture of the lunar surface is evident below the camera shadow and to each side.

The side-angle view of the camera shadow taken at about the same sun angle covers a 25° field of view. It may be noted that even from the Surveyor camera height of about 5.5 feet above the lunar surface, there is a very strong back-scattering of light above and around 5° below and on either side of the camera shadow. As discussed by previous speakers, this factor is being considered in the mission planning for Apollo by utilizing sun angles below the viewing angle. This would be the region covered by the spacecraft below the top of the camera.

FIGURE 18 - SURVEYOR I FOOT-PAD PHOTOGRAPH

This is the computer enhanced photograph of Footpad 2 released by JPL at the June 16 press conference in Washington, D.C. It clearly shows the honeycomb structure of the pad and lunar surface materials in the millimeter size range (.04 inches). On the basis of photographs such as this and other touchdown dynamics measurements, JPL scientists and investigators have estimated that the upper surface materials have a static bearing strength of 5 pounds per square inch at a penetration depth of one inch.

FIGURES 19 AND 20 - SURVEYOR I PANORAMAS - 20° AND 10° SUN ANGLES

These two panoramas are looking to the south of the Surveyor spacecraft as the sun was setting in the West. Figure 19 was made with the sun approximately 20° above the horizon. On figure 20, the sun was about 10° or less than twenty-four hours before sunset. These panoramas illustrate dramatically how visible detail on the lunar surface is directly related to the sun angle.

The large crater just beyond the rock in the left foreground is about 35 feet from the spacecraft, and has been estimated to be about ten feet in diameter and 1 to 2 feet deep. Even at the 20° sun angle, it is difficult to locate this crater. At higher sun angles it would become even more difficult.

The 10° sun angle mosiac shows a pock-marked surface composed of fragmented materials, with the majority being much less than one-sixteenth inch in size. This type of surface has been postulated by many scientists working on NASA programs. In investigating surface and sub-surface explosions on Earth, scientists of the USGS and Ames Research Center found a direct relationship to the cohesion of the material and the type of crater formed. In cohesive material such as basalt they would expect a rocky ejecta with no apparent lip or rim. In loosely cohesive material they would expect little apparent ejecta, but a well-formed and raised lip. JPL investigators have surmised that the 10 foot crater was in loosely cohesive material, whereas the much larger crater in the center of the mosaic is deep enough to have reached a more cohesive or rocky layer.

MOVIE - MARE CARBORUNDUM (AMES RESEARCH CENTER)

Researchers at the NASA Ames Research Center and USGS - Menlo Park have studied high and low velocity impacts in their laboratory through use of special light gas guns and other devices. By relating their results to primary and secondary craters formed by atomic and explosive projectile explosions at various test grounds; Gault, Moore, and others have deduced that the principal small-scale crater forming process on the Ranger photographs resulted from primary and secondary cratering in loosely cohesive materials. The following short movie called Mare Carborundum by Don Gault illustrates how much of the surface layer of the lunar landscape could have been formed by low velocity secondary impacts in loosely cohesive material such as dry sand.

Other than the possibility of using a wider range of projectile sizes, Gault believes this simulation to be characteristic of the major process responsible for the particulate material making up the upper few feet of surface as seen on the Surveyor photographs.

5.0 ORBITER MISSIONS AND DATA ANALYSIS PROCEDURES

FIGURE 21 - ORBITER MISSION A AND B

From an altitude of about 30 miles, the Orbiter will photograph areas about 22 x 56 miles in 8 meter resolution and 10 x 38 miles in 1 meter resolution on each photographic orbit. The A mission has ten photographic sites and the B mission has eleven. On the A mission Sites 2, 4, and 6 are in upland areas and the remaining in mare terrain. On the B mission, Sites 4 and 5 are in upland areas, Sites 7 and 9 are mixed, and the remainder generally in mare terrain. The low resolution photos will provide for stereoscopic analysis. Monoscopic or photometric slope analysis will be performed on the high resolution photos.

It may be noted that two consecutive passes will be used on Mission A in order to insure that the area in which Surveyor landed will be included in the photographs. Langley Research Center, Lunar Orbiter Project Office, has stated that the two consecutive passes may provide side-lap stereo on the high resolution photography of the landed Surveyor I.

FIGURE 22 - DETERMINATION OF POTENTIAL AREAS OF LANDING INTEREST

Due to the large amount of photographs to be returned from each Orbiter mission (about 2,000, 9" x 14" photos) it will be necessary to concentrate the search for acceptable landing sites in areas most accessible according to operational mission and spacecraft constraints. We have initiated a program to analyze each Orbiter Site utilizing as much data as we have available at this time. This chart shows that based on application of current landing site selection criteria certain portions of the Orbiter areas may not be available for use, regardless of the detailed topographic or geologic data obtained from the Orbiter photography.

FIGURE 23 - ORBITER A MISSION - SITE NO.5 AND SLOPES AND CRATERS

This site near the center of the visible side of the Moon will be used as a sample to illustrate some of our current data analysis procedures. The base map illustrates all of the surface topographic detail which can be seen through telescopic observation from Earth. The areas shaded on the base chart illustrate those slopes too steep to be acceptable for landing or those craters several thousand feet or more across which would not be acceptable as a touchdown point.

FIGURE 24 - ORBITER A MISSION - SITE NO.5 - ACCEPTABLE AREA FOR LANDING ELLIPSE

This figure shows areas rejected on the previous figure and the area remaining which could be reached when using a 30 mile radar approach path which did not exceed the site selection criteria by passing over a large crater or steep slope. We have also concluded that to provide the best slope data along the landing radar descent path, the entire approach should be included within the area of low resolution Orbiter photography.

In using the 5.3 x 18.5 kilometer (3 x 11 mile) 3δ landing dispersion ellipse, it is evident that only one landing area can be found in which there are only topographic features below the resolution which can be seen from Earth. If we were to use a 20 mile approach path, there would be five available ellipses of this size and with a ten mile approach path the number increases to eight. If the landing ellipse size were reduced to 3 x 7 kilometers (2 x 4 miles) there would be 24 available landing sites with a ten mile radar approach path in Orbiter Site 5.

FIGURE 25 - ORBITER A MISSION - SITE NO.5 TERRAIN FEATURES

This figure illustrates geologic data prepared by the USGS for the Orbiter Sites. Most of the area is covered by what is considered "undifferentiated mare". This is similar to the type of terrain in which Surveyor I landed. In our analysis of areas available for LM landings, we did not attempt to classify areas topographically or geologically at the scale of the LM landing gear. This information can only be supplied after receipt of the Orbiter photos and after the USGS extrapolates geological information from Surveyor on to the Orbiter photography at the scale of the LM landing gear.

If at this time we consider the ridges, domes, rilles, and escarpments (indicated by the Roman Numeral IV) to be unacceptable within the landing ellipse, it would be necessary to reduce the radar approach path to 10 or 20 miles to get within the large IA (undifferentiated Mare) areas between 0° and 2° W. longitude.

FIGURES 26 AND 27 - PARAMETRIC ANALYSIS OF ORBITER A AND B SITES

These figures illustrate that on the basis of today's knowledge of the Orbiter sites, favorable Apollo landing sites may be found on most of the mare locations, even by taking the most conservative operational approach.

If the lunar surface proves to be exceedingly rough or very heterogeneous insofar as physical properties are concerned, it may be necessary to alter operational plans to reduce the size of the landing ellipse or the length of the radar approach path. Many more sites are available for analysis by utilizing the small landing ellipse and the ten mile radar approach path.

FIGURES 28 AND 29 - RANKING OF MORE FAVORABLE ORBITER SITES AND

CLUSTERED LOCATION IN APOLLO ZONE OF INTEREST

These figures illustrate the Orbiter Sites from Missions A and B which possess favorable areas for Apollo landing site analysis. As a result of application of operational site selection criteria and the changes resulting from month to month due simply to celestial motions, it is fortuitous that by selecting alternate days for three launch opportunities, favorably clustered sites are located at about 30° East longitude, near the 0° meridian, and about 35° West longitude.

6.0 SOURCES OF DATA FOR LANDING SITE SELECTION

FIGURE 30

Other NASA Centers, the USGS, DoD Mapping Agencies, JPL and Surveyor Scientific Evaluation Teams, and MSC will participate in providing data for the selection of candidate Apollo landing sites for the first and subsequent lunar missions. The final selection of the lunar sites to be used for each Apollo mission will be made by the Associate Administrator for Manned Space Flight from the recommendations of the Apollo Site Selection Board, chaired by Major General Samuel Phillips, Office of Manned Spaceflight.

Questions and Answers

SITE SELECTION DATA SOURCES AND INTERPRETATION

Speaker: John E. Dornbach

1. Why was site A5 used in the discussion?

   ANSWER - It was purely an example.

2. Comment from Audience: The Orbiter sidelap percentages are quite small, and it would be difficult to include a Surveyor site in stereo coverage.

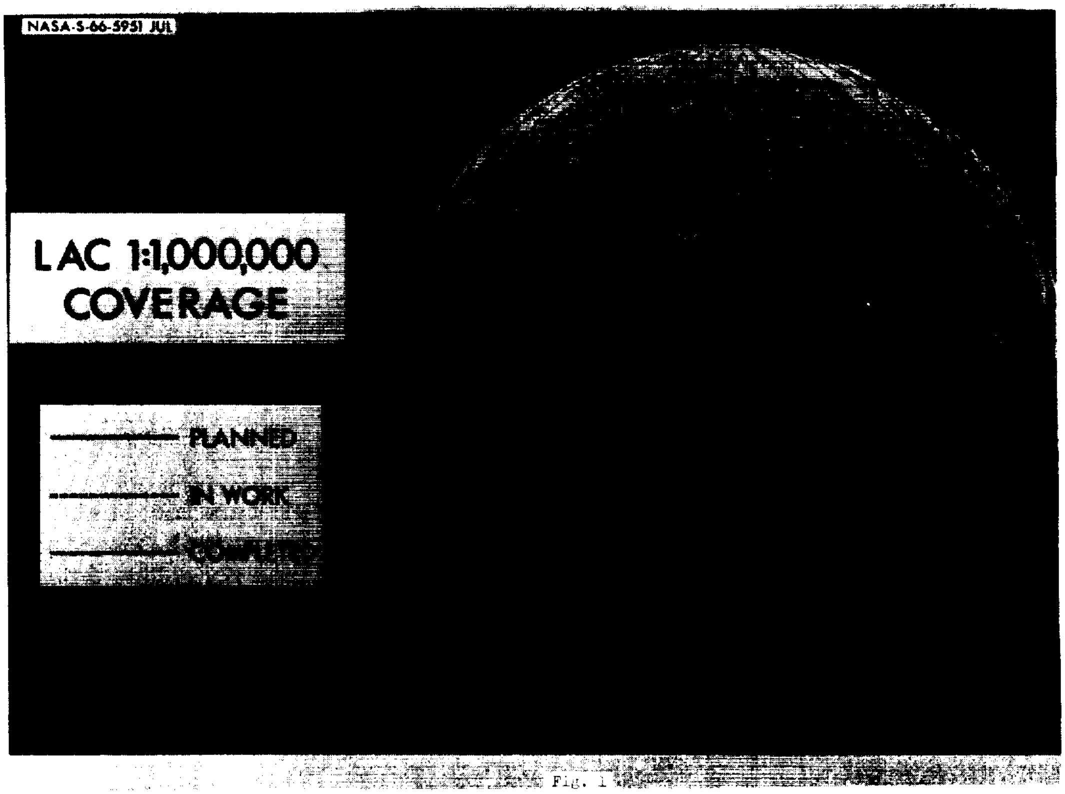
   Comment from Mr. Dornbach: It is our understanding that Orbiter will have a larger sidelap than the normal 5%. This could be done by tilting the spacecraft; however, the actual method to be used is not known.

3. Were there filters on the Surveyor and could the details [of the surface features] be increased by the use of these filters?

   ANSWER - There were filters, but their purpose was for colorimetry. Since the lunar surface is basically grey, the filters had little effect on the imagery received.

4. Has any consideration been given to observing the LM landing by another spacecraft in orbit, such as Lunar Orbiter?

   ANSWER (Dr. Shea) - The answer is basically "no", primarily because it would represent an extra constraint [to get into phase with some other spacecraft which was already in lunar orbit] out of which one would expect to get very little additional real value to the program.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LAC 1:1,000,000
COVERAGE
PLANNED
IN WORK
COMPLETED

Fig. 1

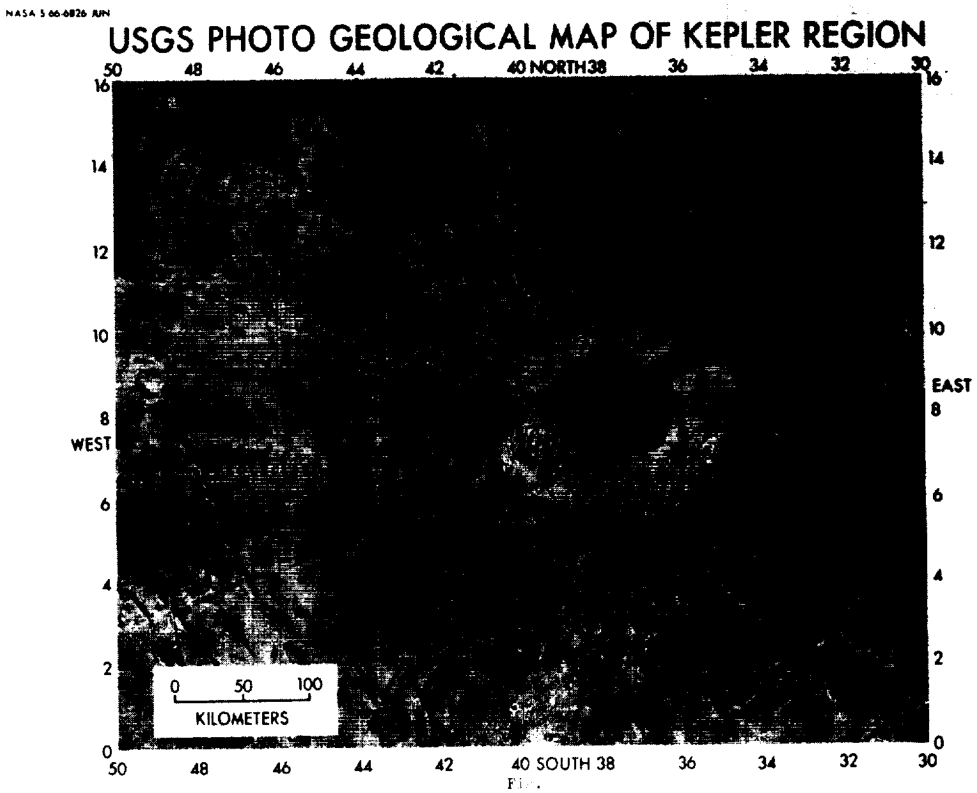
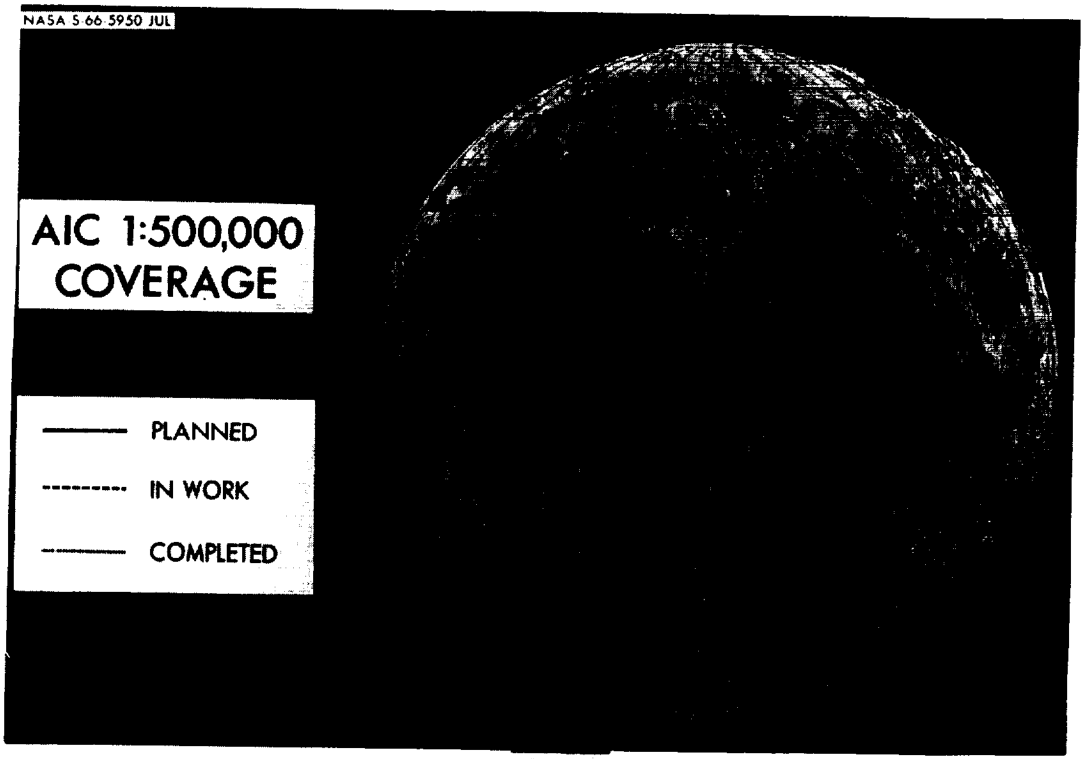

NASA S 66-6826 JUN
USGS PHOTO GEOLOGICAL MAP OF KEPLER REGION
50
48
46
44
42
40 NORTH 38
36
34
32
30
16
14
12
10
8
WEST
6
4
2
0
EAST
0
50
100
KILOMETERS
40 SOUTH 38

Fig.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AIC 1:500,000 COVERAGE
PLANNED
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Fig. 3

NASA-S-66-6024 MAY

THERMAL ENHANCEMENTS (LINES) AND HOT SPOTS (DOTS) NEAR THE END OF TOTALITY

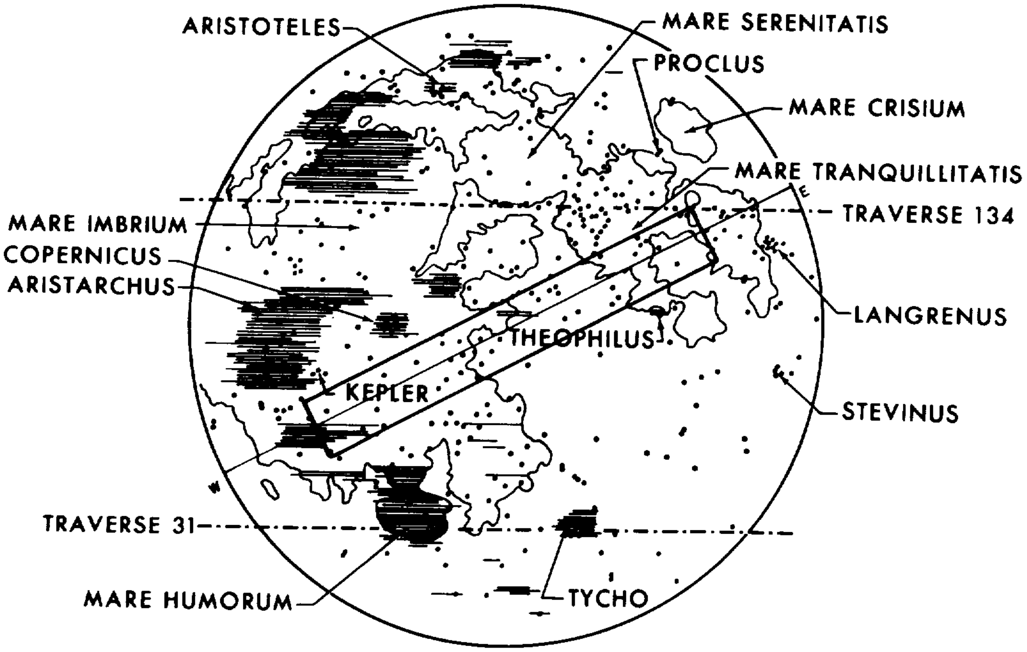
ARISTOTELES
MARE SERENITATIS
PROCLUS
MARE CRISIUM
MARE TRANQUILLITATIS
TRAVERSE 134
E
MARE IMBRIUM
COPERNICUS
ARISTARCHUS
THEOPHILUS
LANGRENUS
KEPLER
STEVINUS
W
TRAVERSE 31
MARE HUMORUM
TYCHO

Fig. 4

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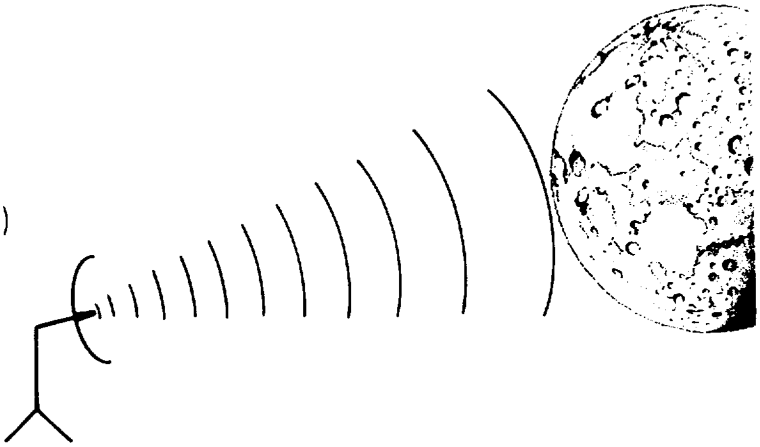
RADAR MAPPING



MIT LINCOLN LABS
ARECIBO (CORNELL)

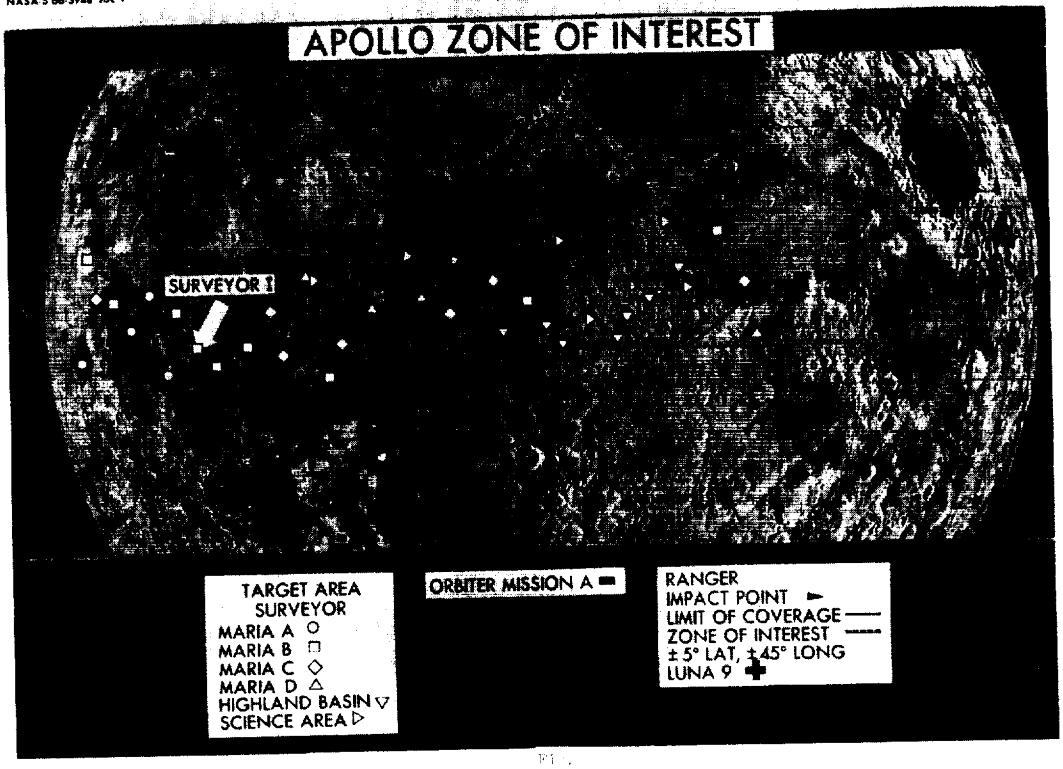
0.86 CM
3.8 CM
23.0 CM
70.0 CM
7.5 M

LOCALIZED AREAS ON MOON
(<10 SQ KM IN SIZE)

1. ESTIMATE BULK DENSITY VERSUS DEPTH
2. SURFACE ROUGHNESS-RMS SLOPE VERSUS SPACING

FIG. 5

NASA-S-66-5948 JUL 1



APOLLO ZONE OF INTEREST

TARGET AREA
SURVEYOR
MARIA A ○
MARIA B □
MARIA C ◇
MARIA D △
HIGHLAND BASIN ▽
SCIENCE AREA ▷

ORBITER MISSION A ▬

RANGER
IMPACT POINT ►
LIMIT OF COVERAGE ——
ZONE OF INTEREST ----
± 5° LAT, ± 45° LONG
LUNA 9 ✚

FIG.

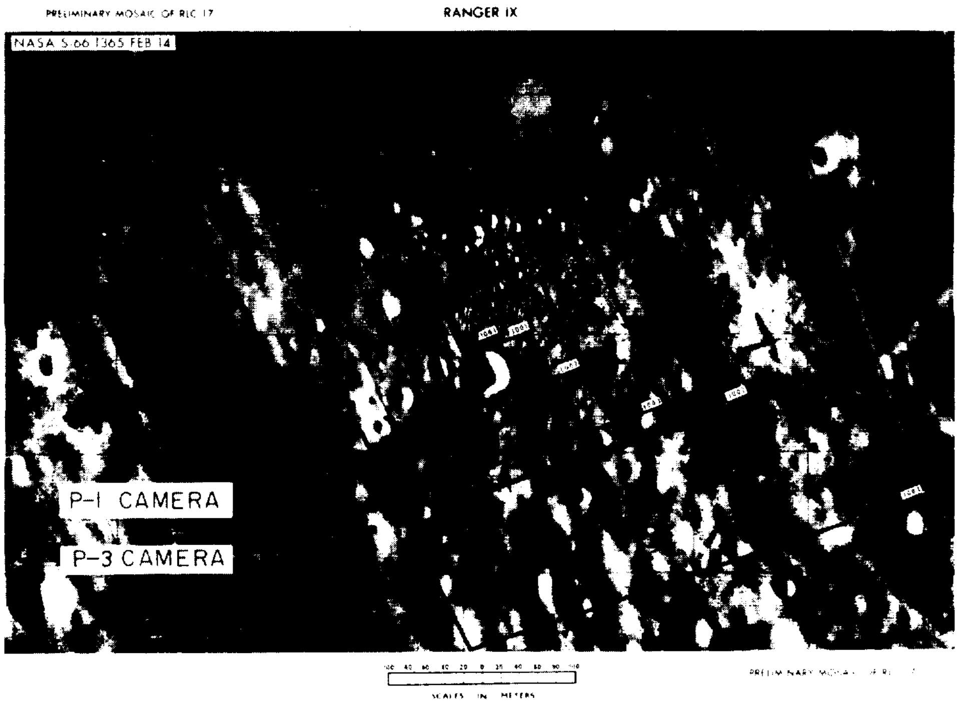

PRELIMINARY MOSAIC OF RLC 17
RANGER IX
NASA S-66 1365 FEB 14
P-1 CAMERA
P-3 CAMERA
100 80 60 40 20 0 20 40 60 80 100
SCALES IN METERS

Fig. 7

NASA-S-66-6049-JUN

LUNAR SLOPES AT LM SCALE

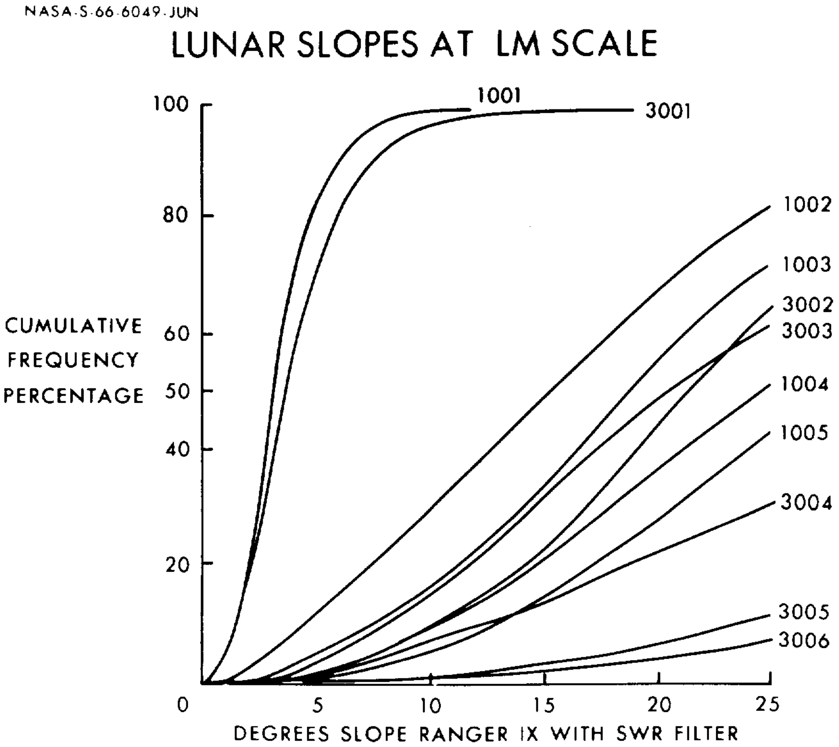


Fig. 8

NASA-S-66-6056 JUNE

RANGER 8 APOLLO SITE ANALYSIS

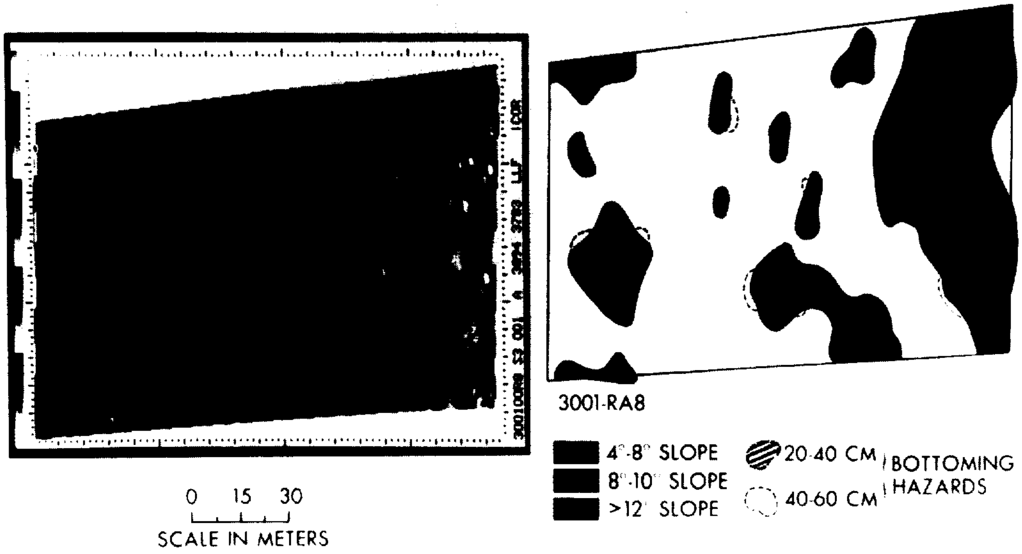
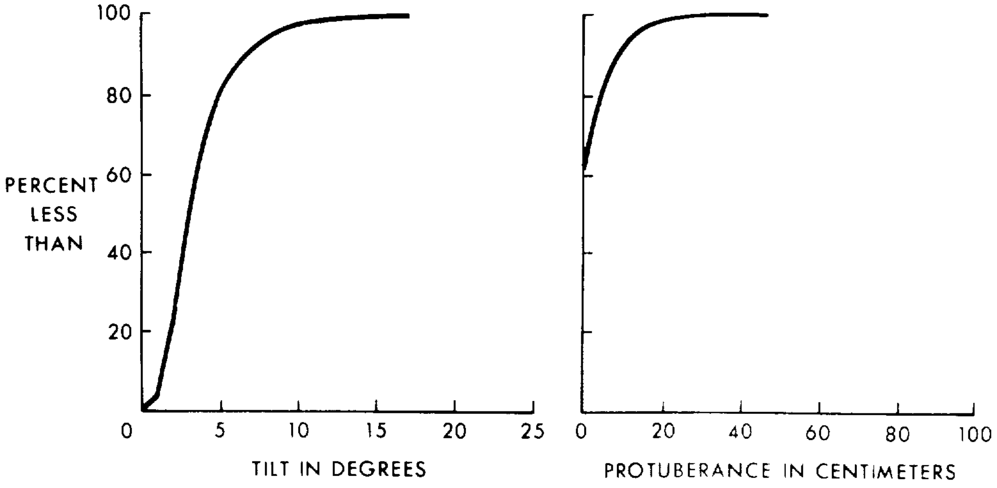

3001-RA8
4°-8° SLOPE
8°-10° SLOPE
>12° SLOPE
20-40 CM
40-60 CM
BOTTOMING HAZARDS
0 15 30
SCALE IN METERS

Fig.

NASA-S-66-6055 JUN

RANGER 8-LM LANDABILITY



RA-8-3001 WITH SWR FILTER

Fig.

RANGER IX PHOTOGRAPH OF FLOOR OF ALPHONSUS

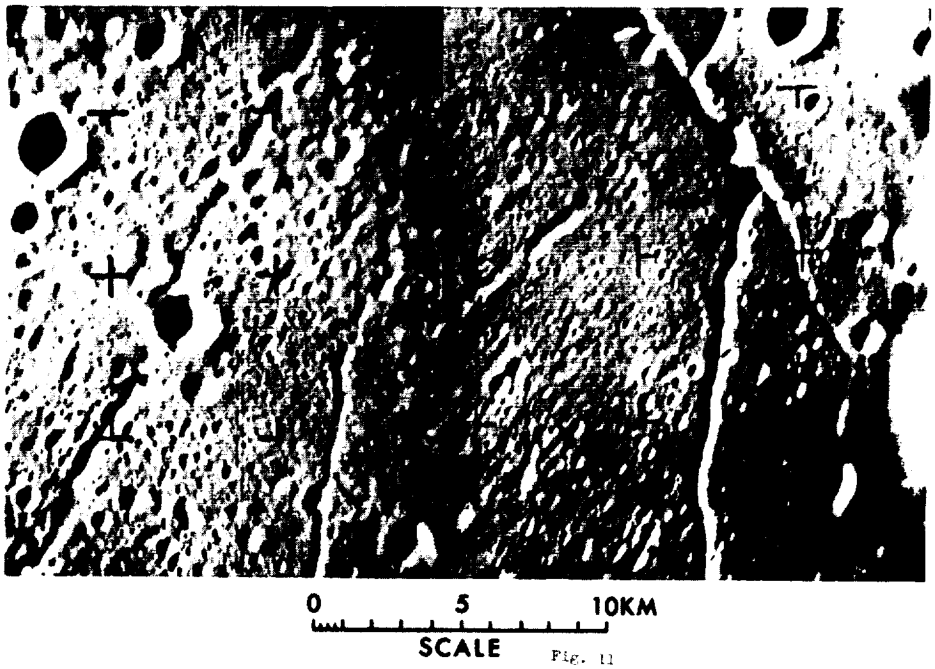
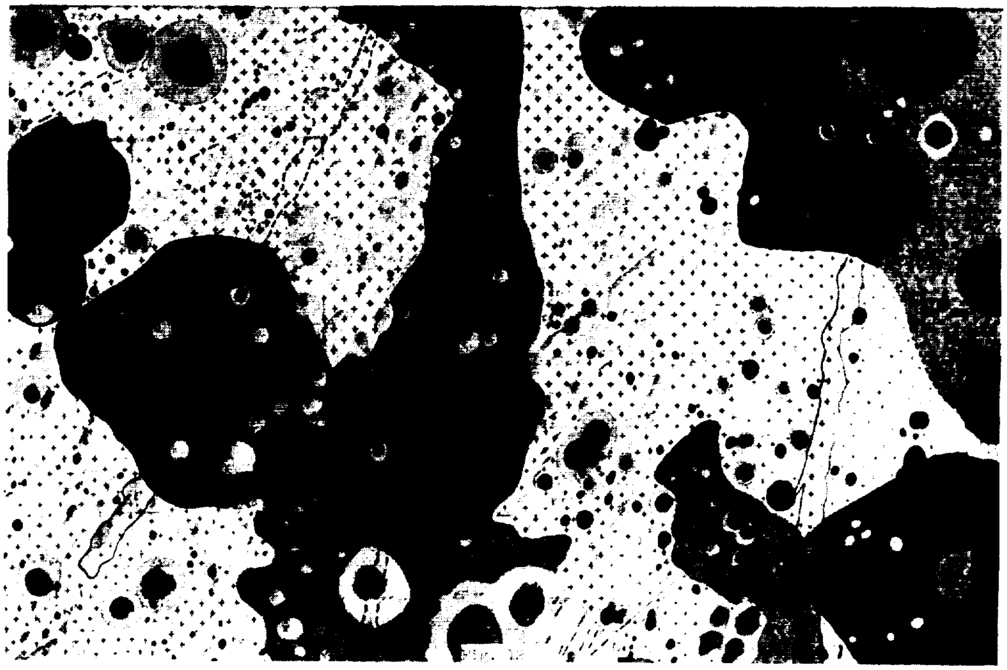

0
5
10KM
SCALE

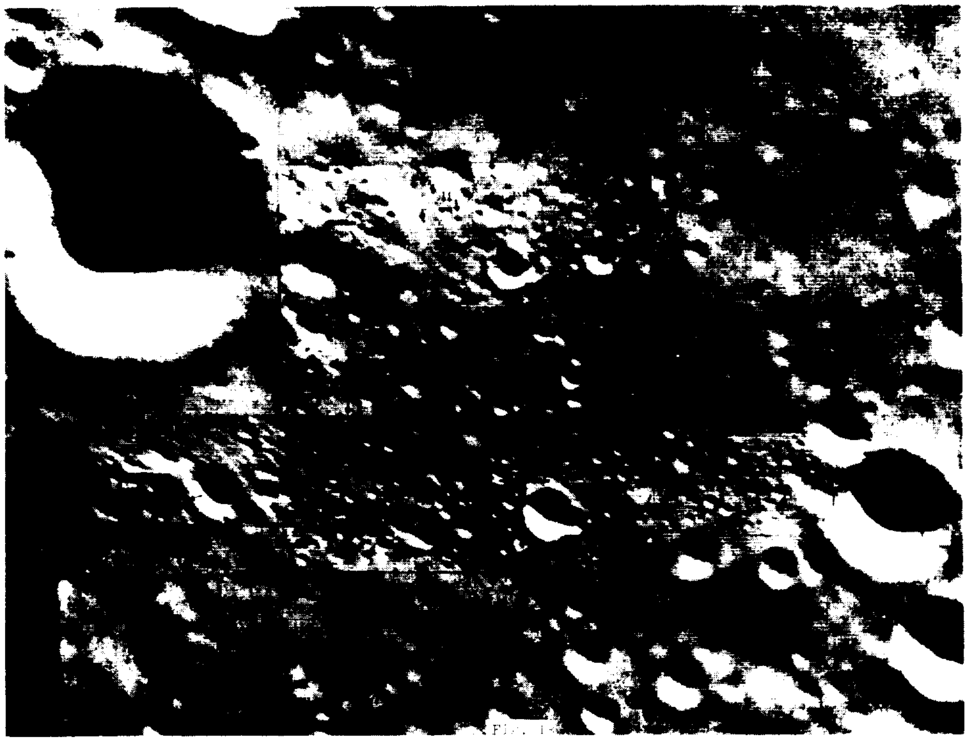
Fig. 11

GEOLOGICAL MAP OF THE FLOOR OF THE ALPHONSUS



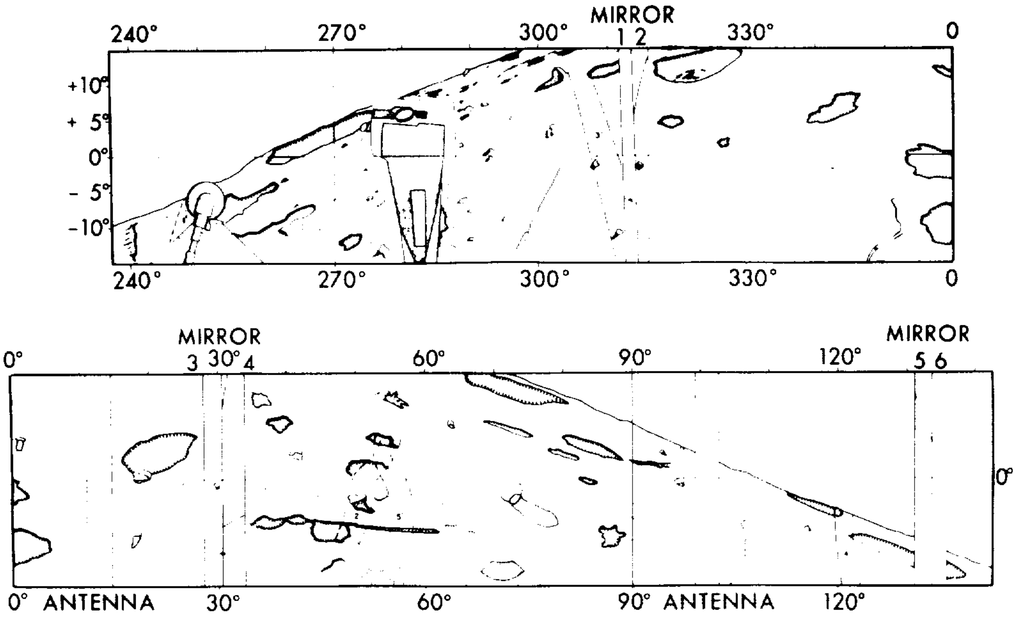
NASA S 66 5056 JUNE 1

KUIPER BEARING STRENGTH ANALYSIS



NASA-S-66-5160 JUN 8

SKETCH OF PORTION OF LUNA IX PANORAMA TAKEN FEBRUARY 5, 1966 TILT 21 , SUN ELEVATION 27


MIRROR
1 2
240°
270°
300°
330°
0
+10°
+ 5°
0°
- 5°
-10°
240°
270°
300°
330°
0
MIRROR
3 30° 4
MIRROR
5 6
0°
60°
90°
120°
0°
0° ANTENNA
30°
60°
90° ANTENNA
120°

LUNAR IX PHOTOGRAPHIC INTERPRETATION

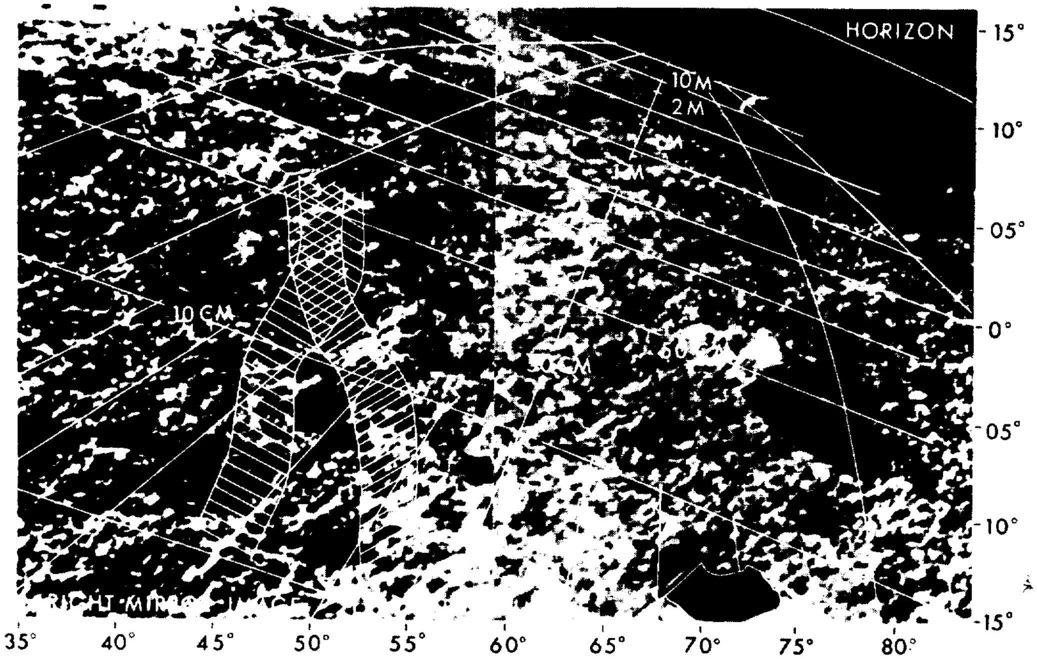
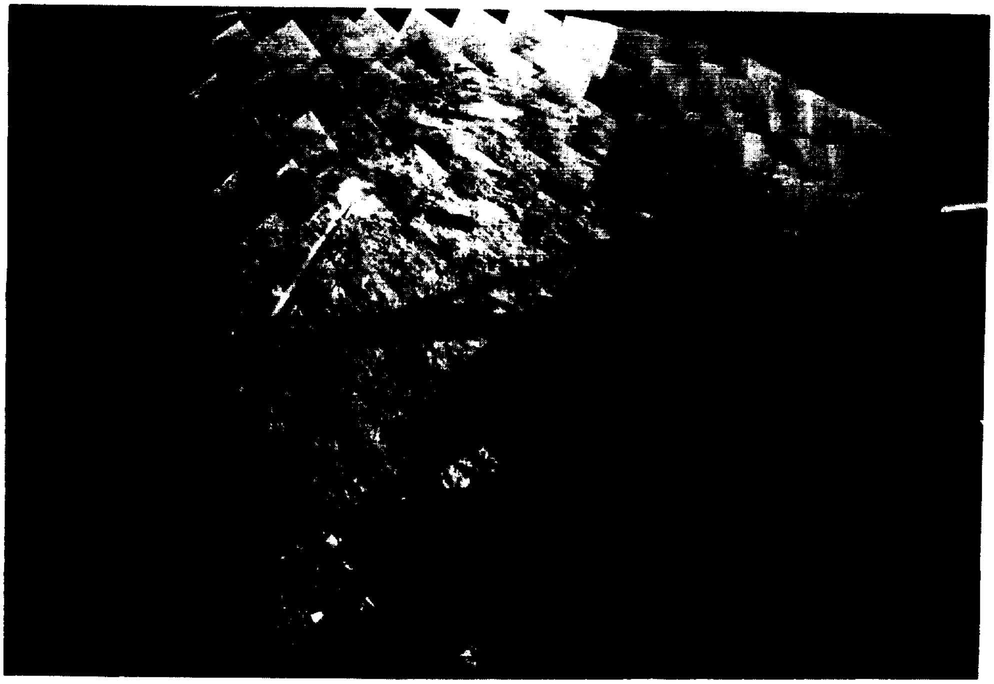

HORIZON
10M
2M
10 CM
RIGHT MIRROR IMAGE
15°
10°
05°
0°
05°
10°
15°
35°
40°
45°
50°
55°
60°
65°
70°
75°
80°

Fig. 15

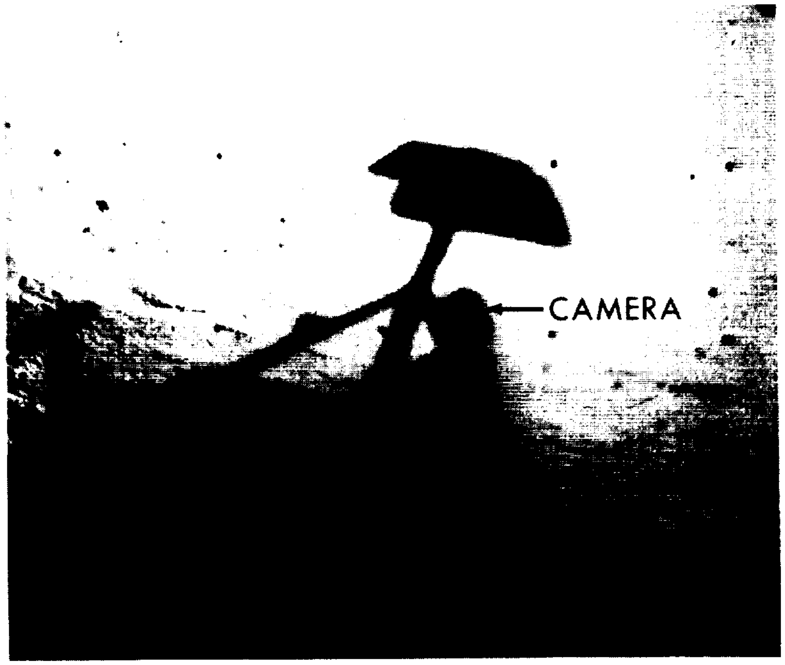
SURVEYOR I



SUN ANGLE ~10°

Fig. 16

SURVEYOR I



SUN ANGLE ~10°
FOV 25°

SURVEYOR I

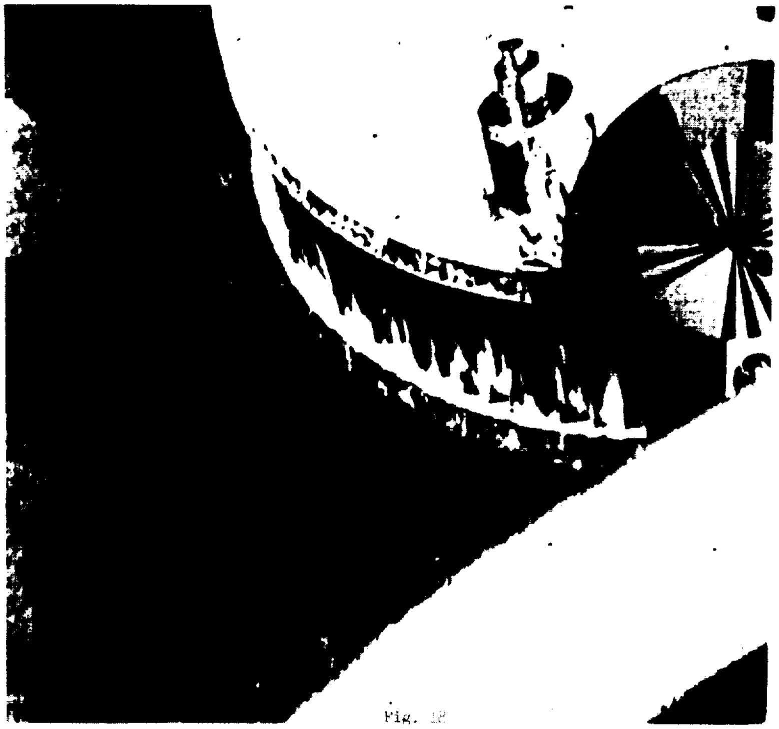
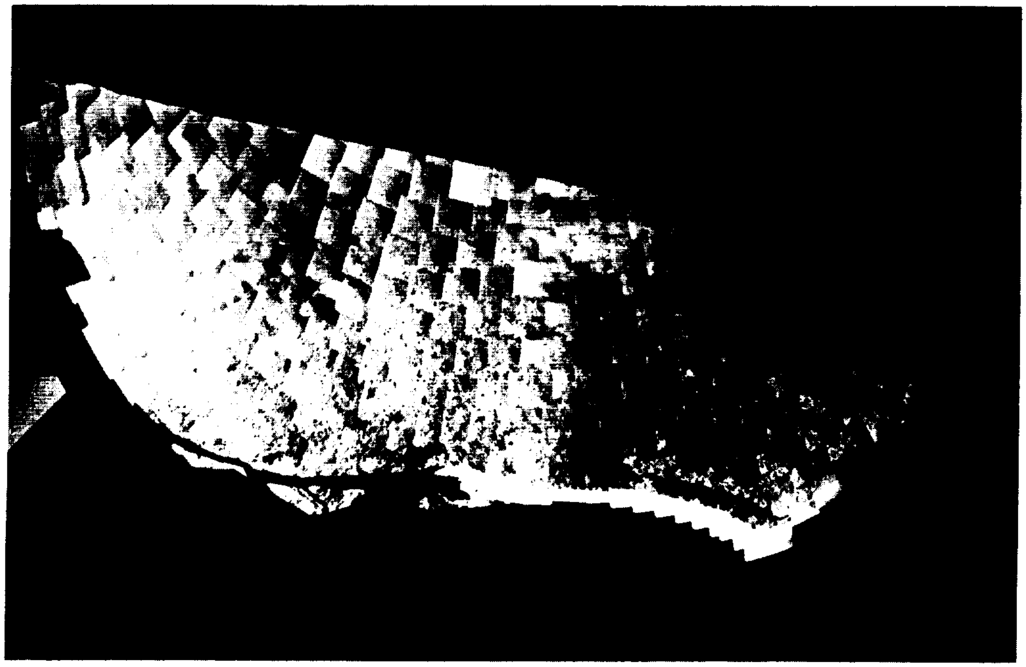


Fig. 18

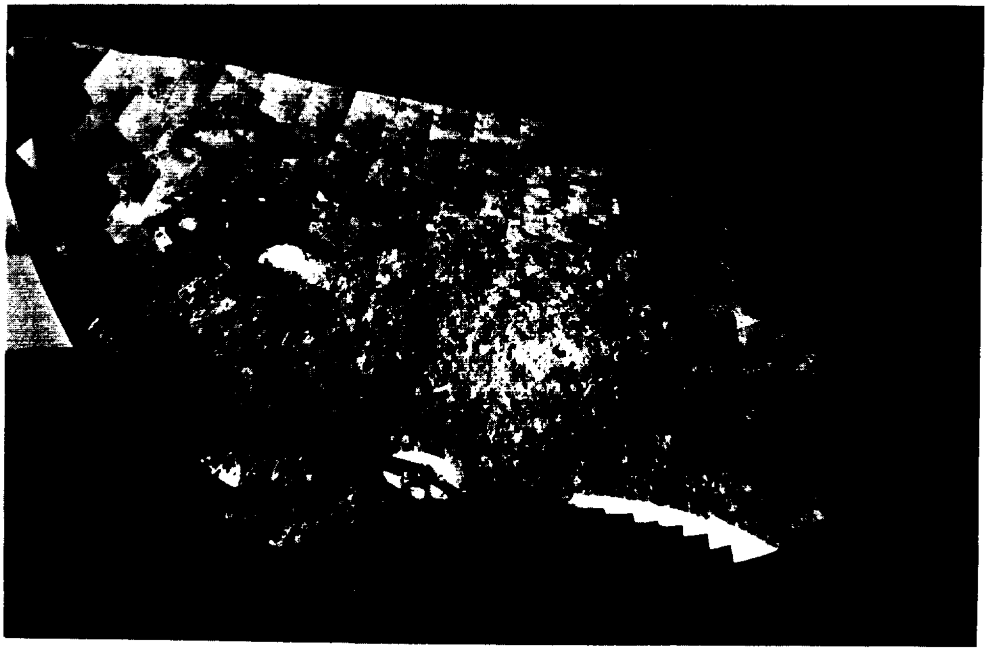
SURVEYOR I



SUN ANGLE ~20°

Fig. 19

SURVEYOR I



SUN ANGLE ~10°

Fig. 20

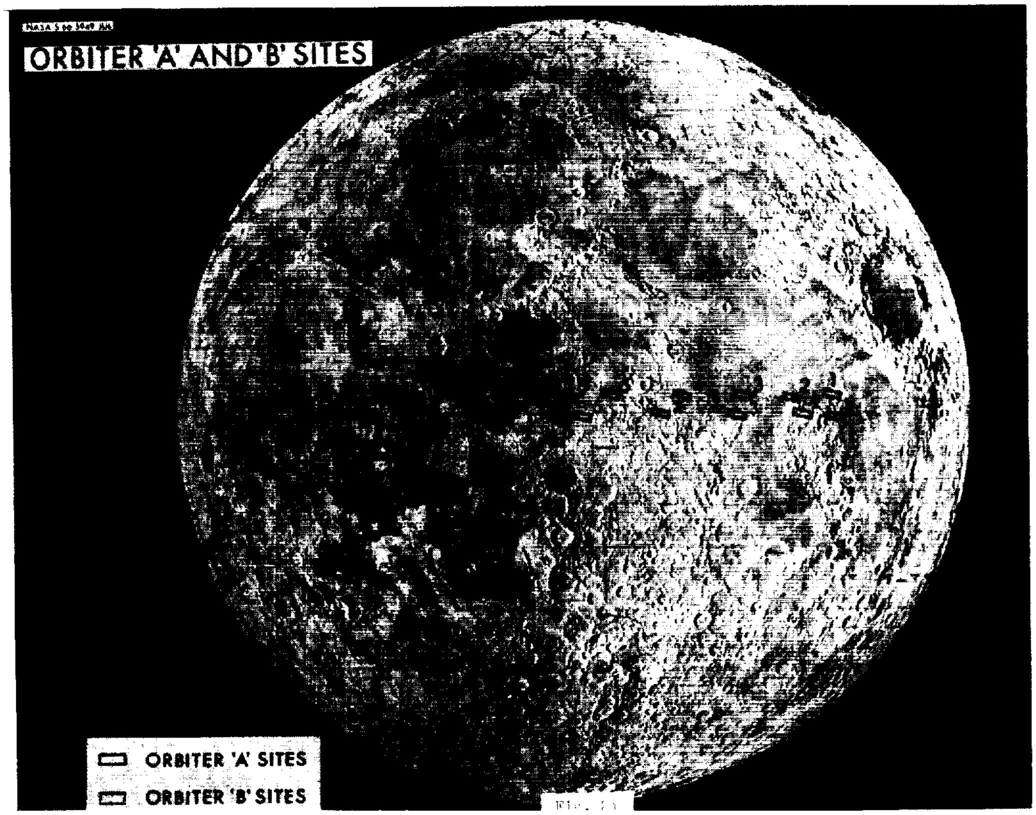

NASA S 66 3449 JUN
ORBITER 'A' AND 'B' SITES
ORBITER 'A' SITES
ORBITER 'B' SITES

Fig. [illegible]

NASA-S-66-6443 JUN

DETERMINATION OF POTENTIAL AREAS OF LANDING INTEREST

BASED ON CURRENT LANDING SITE SELECTION CRITERIA

↓

REJECT IN SEQUENCE

1. SLOPES > 10°
2. SLOPES 6 - 10°
3. CRATERS 0.5 - 10 KM DIA
4. AREAS TOO SMALL FOR LM 3σ DISPERSION ELLIPSE
5. UNACCEPTABLE RADAR APPROACH PATHS

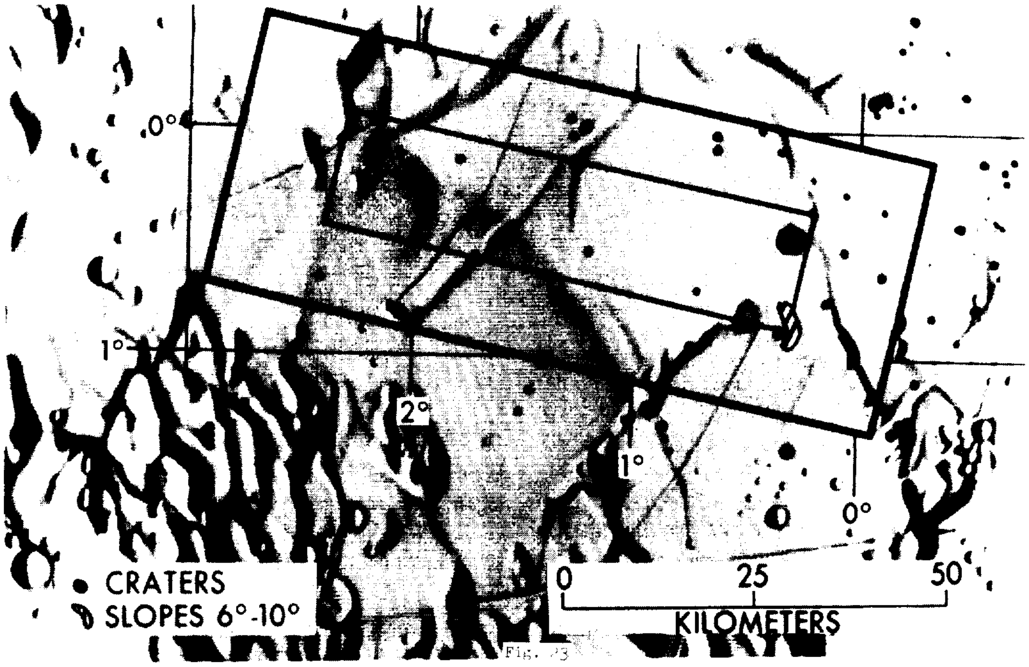
↓

POTENTIAL TARGETS FOR ORBITER AND SURVEYOR

Fig. [illegible]

ORBITER "A" MISSION-SITE NO. 5

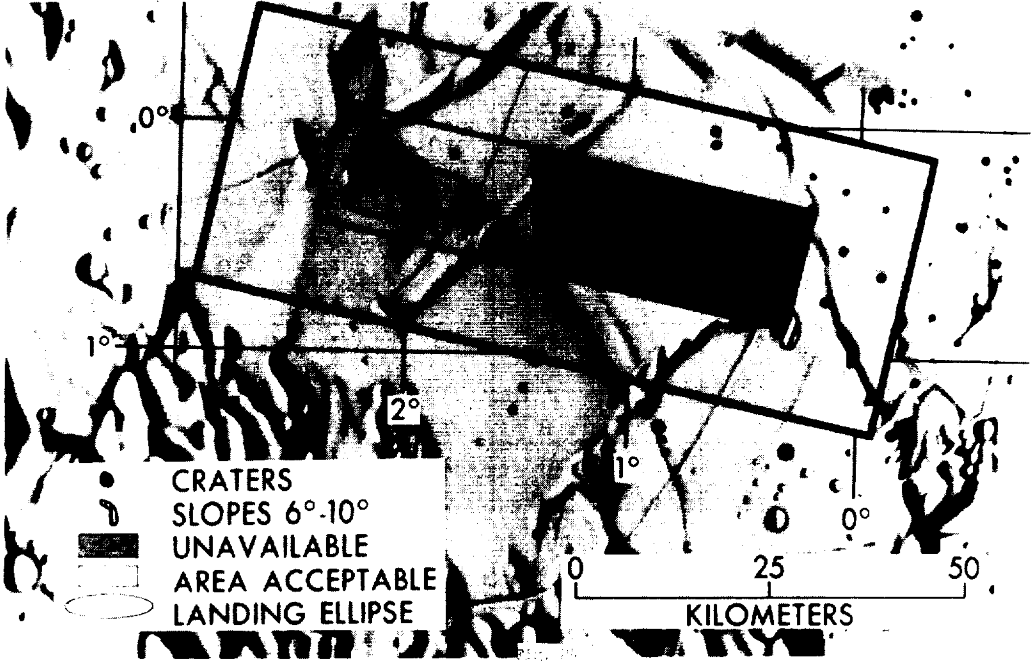
CRATER AND SLOPE DELINIATION



0°
1°
2°
1°
0°
● CRATERS
SLOPES 6°-10°
0
25
50
KILOMETERS

ORBITER "A" MISSION-SITE NO. 5

ANALYSIS BASED ON 5.3 × 18.5 Km LANDING ELLIPSE WITH 55 Km (30 N M) RADAR APPROACH PATH

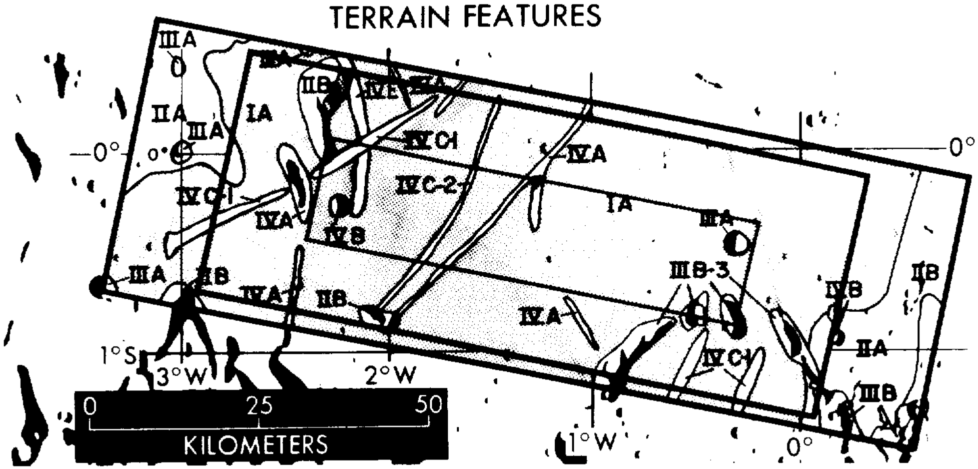


0°
1°
2°
1°
0°
● CRATERS
SLOPES 6°-10°
UNAVAILABLE
AREA ACCEPTABLE
LANDING ELLIPSE
0
25
50
KILOMETERS

NASA-S-66-3891 JUNE

ORBITER 'A' MISSION - SITE NO. 5

TERRAIN FEATURES


IIIA
IIA
IIIA
IA
0°
IIB
IVE
IVA
IVC-1
IVC-2
IVA
IVB
IA
IIIA
IIIB-3
IIB
IVB
IIA
IIIB
IVC-1
IIIA
IIB
IVA
1°S
3°W
2°W
1°W
0°
0
25
50
KILOMETERS

I A	MARE, UNDIFFERENTIATED	III C	CRATERS FIELDS AND CRATERS
II A	UPLAND, PLAINS AREAS	IV A	RIDGES
II B	UPLAND (SUBDUED)	IV B	DOMES
III A	WELL FORMED CRATERS	IV C-1	RILLS
III B	MODIFIED CRATERS	IV C-2	SINUOUS RILLS
III B-3	RIM OR SMALL CRATERS	IV E	ESCARPMENTS

REF D H DAHLEM ETAL 1966 - PRELIMINARY TERRAIN MAP OF THE LUNAR EQUATORIAL BELT

Fig. 25

NASA-S-66-5208 JUN

HIGH RESOLUTION AREAS

ORBITER A   ELLIPSE 5.3 × 18.5 Km

SITE NO.	MAX* AREA	RADAR		APPROACH		PATH (N MI)	
		10		20		30	
	%	%	#	%	#	%	#
1	95	80	8	40	5	20	2
2	90	20	2	0	0	0	0
3	95	60	8	60	7	57	7
4	90	27	3	15	1	10	1
5	95	70	8	42	5	7	1
6	55	0	0	0	0	0	0
7	95	60	7	33	4	0	0
8	99	85	8	85	8	85	8
9	90	55	6	45	4	45	4
10	98	90	8	90	8	90	8

*MAX ACCEPTABLE AREA (EXCLUDING CRATERS AND SLOPES OVER 6°)

Fig.

NASA-S-66-5207 JUN

HIGH RESOLUTION AREAS

ORBITER B

ELLIPSE 5.3×18.5 KM

SITE NO.	MAX* AREA	RADAR APPROACH PATH (N MI)					
		10		20		30	
	%	%	#	%	#	%	#
1	100	75	6	45	5	20	2
2	95	70	7	45	5	30	3
3	95	33	5	33	5	30	4
4	85	22	2	7	1	0	0
5	70	9	1	0	0	0	0
6	90	43	4	10	1	10	1
7	60	17	2	10	1	10	1
8	90	42	5	23	3	23	3
9	90	20	2	0	0	0	0
10	98	95	9	95	9	95	9
11	97	85	8	85	8	77	7

*MAX ACCEPTABLE AREA (EXCLUDING CRATER, AND SLOPES OVER 6°)

Fig. 27

NASA-S-66-5183 JUN

AVAILABLE ORBITER SITES FROM MISSION A & B

(5.3 × 18.5 KM ELLIPSE)

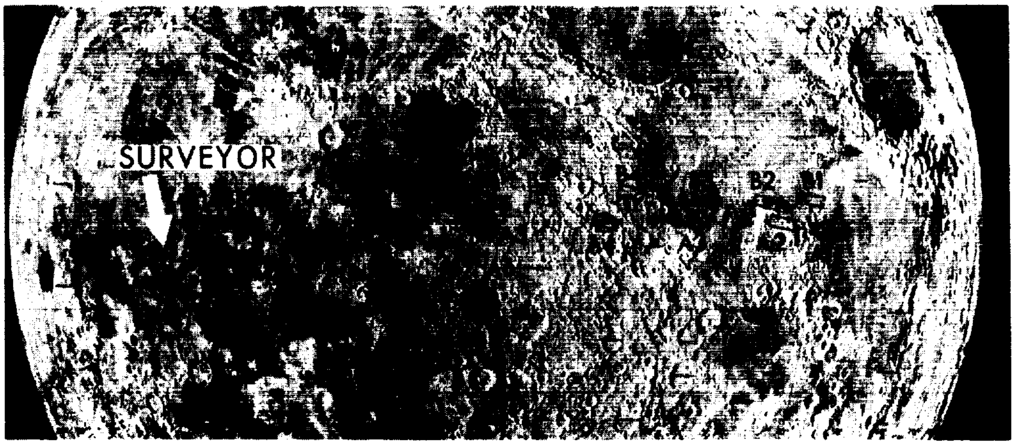
(30 N MI RADAR APPROACH)

DESCENDING ORDER OF SITES AVAILABLE	ORBITER SITE (NO. OF ELLIPSES)
1	B-10 (9)
2	A-8 (8)
3	B-11 (7)
4	A-3 (7)
5	A-9 (4)
6	B-3 (4)
7	B-2 (3)
8	B-8 (3)
9	B-1 (2)
10	A-1 (2)
11	B-6 (1)
12	A-5 (1)

Fig. 28

NASA-S-66-5947 JUL

AREAS FOR APOLLO LANDING SITE ANALYSIS



▭ ORBITER A AND B SITES
⬚ FAVORED ORBITER A AND B SITES
○ SITES NOW USED IN MISSION PLANNING
X LUNAR LANDING AREA – NATURAL ENVIRONMENT AND PHYSICAL STANDARDS FOR THE APOLLO PROGRAM, APRIL 1965

Fig. 29

NASA-S-66-6052-JUN

SOURCES OF DATA FOR LANDING SITE SELECTION

DATA	SOURCE
LEM SLOPE FREQUENCY DISTRIBUTION	MSC
LEM BOTTOMING HAZARD FREQUENCY DISTRIBUTION	MSC
ANALYSIS OF TERRAIN FOR LANDING RADAR APPROACH	MSC
TOPOGRAPHIC MAPS	DOD, ACIC & AMS
PHOTO-GEOLOGIC MAPS	USGS
LUNAR SOIL MECHANICS PROPERTIES	JPL SURVEYOR–WORKING GROUPS USGS MSC
ANALYSIS OF SITES FOR SCIENTIFIC INTEREST	USGS
ORBITER DATA SCREENING	LRC, MSC & USGS

Fi . ·

CONTROL OF LUNAR SURFACE CONTAMINATION
AND
BACK CONTAMINATION

BY

RICHARD H. KOHRS

THE PURPOSE OF THIS PRESENTATION IS TO:

Show the current status of hardware design and procedures that are being developed to control or minimize lunar surface and back contamination.

Subsequent charts will show:

a. For each crew biological contamination source, the hardware or procedure being developed to minimize lunar surface contamination.

b. Preliminary contour plots showing the level and area of concentration of the descent propellant contaminants.

c. The current design criteria for the prevention of back contamination plus the spacecraft and recovery procedure postulated to meet this criteria.

d. The current design requirements and concepts of the Lunar Receiving Lab.

BIOLOGICAL LUNAR SURFACE CONTAMINATION

Figures 1, 2, and 3 summarize for each crew biological contamination source the hardware or procedure design currently being implemented. The prevention of exposure of crew biological excretions is accomplished primarily when there is still lunar surface sampling and exploration time remaining. All storable excretions are kept in the LM up to the completion of the last excursion. At this time the storage containers are offloaded onto the lunar surface and then stored in the LM descent stage by the LM crewman. Similarly, after the LM crewman enters the LM after the final excursion, certain EMU components are offloaded onto the lunar surface. Offloading of these storage containers and equipment accounts for approximately a 100 pound inert weight savings which is equivalent to approximately 200 pounds of total ascent stage launch weight.

SUPPORTING DATA

Sweat Filter	95% Effective
Cabin Air	80% Effective
LiOh Cannisters	Tests scheduled this month at Ft. Dietrich
Germicide	SETOL

NASA-S-66-5740 JUN 16

LUNAR SURFACE CONTAMINATION

BIOLOGICAL CONTAMINATION SOURCE	DISPOSITION
FECES URINE	STORED WITHIN LM UNTIL LUNAR SAMPLING AND EXPLORATION COMPLETED
LiOH CANISTERS	TRANSFERED TO THE LUNAR SURFACE AND STORED IN LM DESCENT STAGE PRIOR TO LAST EVA INGRESS
PLSS CONDENSATE	FECES, URINE, AND PLSS CONDENSATE CONTAINERS CONTAIN GERMICIDE
SWEAT	PASSED THROUGH BACTERIA FILTER PRIOR TO BEING VENTED ONTO LUNAR SURFACE
CABIN AIR	PASSES CONTINUOUSLY THROUGH LiOH CANISTER DURING CABIN DEPRESSURIZATION PASSES THROUGH FILTER INSTALLED ON CABIN VENT

Fig. 1

NASA-S-66-5133 JUN 8

LUNAR SURFACE CONTAMINATION

BIOLOGICAL CONTAMINATION SOURCE	DISPOSITION
EXTRAVEHICULAR ASTRONAUT	
FECES	DEFECATION NOT ANTICIPATED. IF REQUIRED, STORED IN SUIT UNTIL RETURN TO LM
URINE	STORED IN SUIT AND THEN DUMPED INTO STORAGE CONTAINERS UPON RETURN TO LM CABIN
LiOH CANNISTERS	REMOVED FROM BACK PACK WITHIN LM, STORED, AND THEN OFFLOADED INTO DESCENT STAGE
SWEAT	COLLECTED AS PLSS CONDENSATE. DUMPED INTO STORAGE CONTAINERS UPON RETURN TO LM
SUIT AIR	PASSES THROUGH LiOH CANNISTER IN BACK PACK. .04 LB/HR SUIT LEAK ONTO SURFACE

Fig. [illegible]

NASA-S-66-5134 JUN

LUNAR SURFACE CONTAMINATION

BIOLOGICAL CONTAMINATION SOURCE	DISPOSITION
EXTRAVEHICULAR MOBILITY UNIT COMPONENTS 1 PLSS 2 PR LUNAR BOOTS	LEFT ON LUNAR SURFACE AFTER COMPLETION OF LAST LUNAR EXPLORATION ONE PLSS RETAINED DURING LM ASCENT FOR BACKUP EXTRAVEHICULAR TRANSFER

Fig. 3

DESCENT ENGINE CONTAMINATION

Figures 4 through 7 show the amounts of the principal LM exhaust gas constituents that will be adsorbed on the lunar surface in the vicinity of the LM touchdown site. These charts are results of an initial study of GAEC and show preliminary estimates. The maps are oriented with respect to the landing trajectory so that the LM approaches the touchdown point from the top of the map. It is apparent that the most heavily contaminated regions lie along the trajectory. Maps of this type could serve as guides to locations from which lunar surface samples containing minimum amounts of contamination can be collected. They could also indicate to scientists the amounts of contamination that will exist in samples gathered at various locations with respect to the touchdown point.

Contours of constant adsorption in units of micrograms of adsorbed gas per square centimeter of lunar surface are plotted for H_2O, OH, NO, O, and O_2. CO, CO_2, H, and H_2 are also present in the exhaust; however, no appreciable amounts of these adsorbed contaminants should be found in lunar samples.

Maps of this type including information on the depth of contamination could be used as an important tool in compensating for contamination. Such maps would indicate that the astronauts need to collect samples at distances of only 1000 or less feet from the LM to insure samples that are relatively uncontaminated.

Similarly, if the location, time, and depth at which every lunar sample is collected is recorded, a comparison can be made between the amounts and distribution of actual versus predicted contaminants during postflight evaluation. This data would be useful in the development of distribution maps for subsequent flights.

SUPPORTING DATA

It is currently estimated that approximately 1/3 of exhaust by-products will be H_2O, 1/3 N_2, and 1/3 H_2, CO, CO_2, H and O.

Propellant UDMH and N_2H_4 + N_2O_4 (50/50 unsymdimethydrazine/hydrazine + nitrogen tetroxide).

NASA-S-66-6086 JUN

CONTOURS OF EQUAL ADSORPTION OF H_2O IN UNITS OF $\mu g/cm^2$

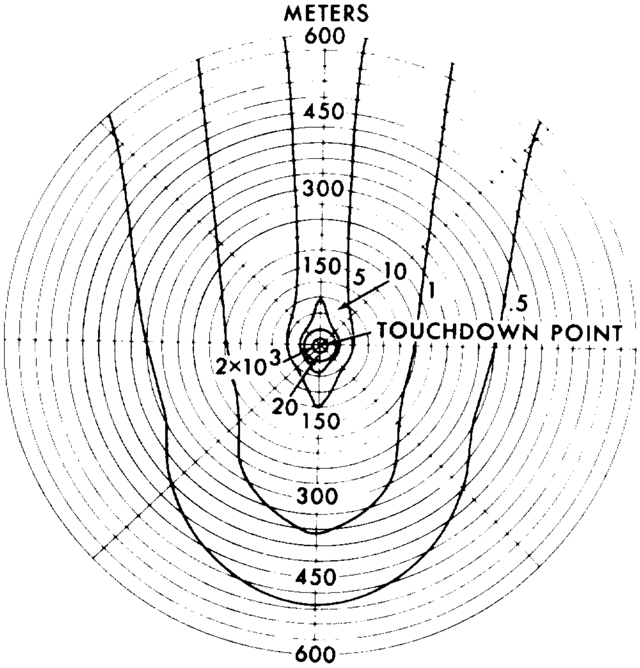


Fig. 4

NASA-S-66-6087 JUN

CONTOURS OF EQUAL ADSORPTION OF OH IN UNITS OF $\mu g/cm^2$

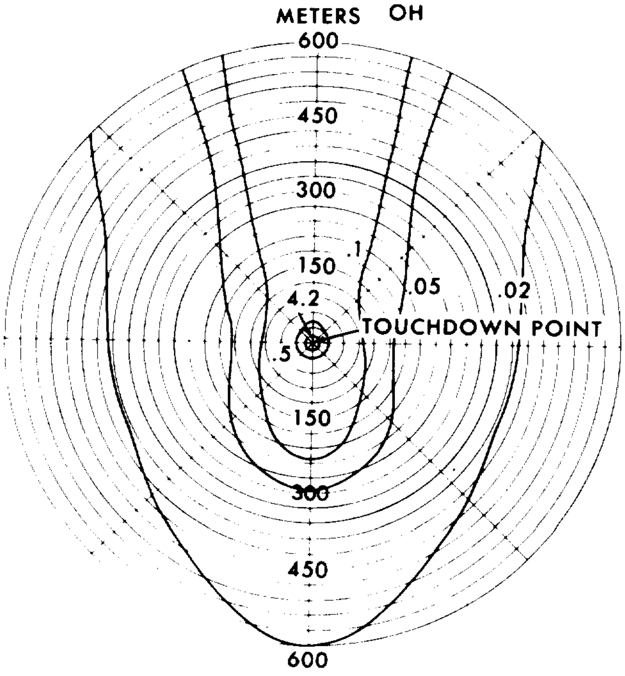


Fig. 5

NASA-S-66-6088 JUN

CONTOURS OF EQUAL ADSORPTION OF NO IN UNITS OF $\mu g/cm^2$

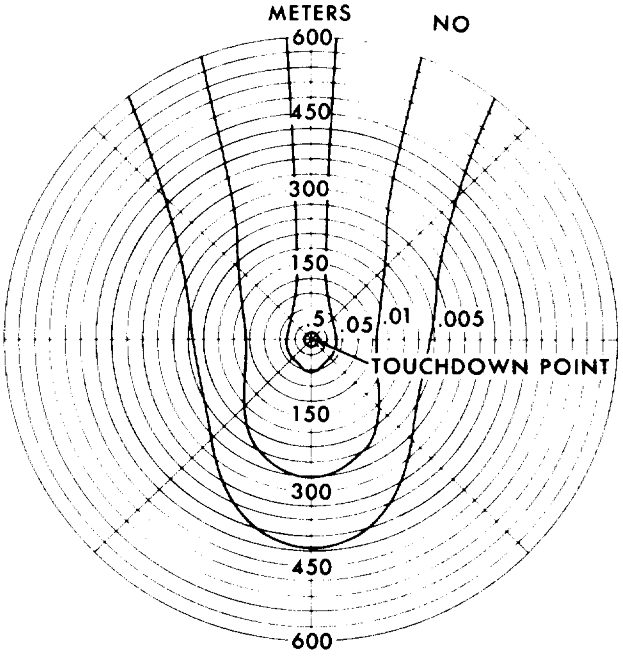


Fig.

NASA-S-66-6089 JUN

CONTOURS OF EQUAL ADSORPTION OF O AND O_2 IN UNITS OF $\mu g/cm^2$

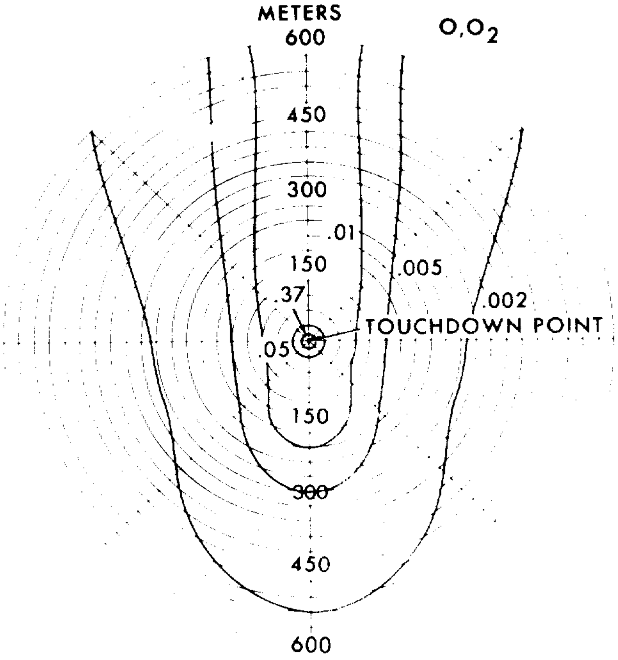


Fig.

BACK CONTAMINATION

The remaining charts will show the back contamination criteria, the postulated inflight and recovery procedures to meet this criteria, plus the LRL (Lunar Receiving Lab) design requirements and concepts.

Hardware and procedures to minimize back contamination can be divided into three specific phases. Namely:

a. The time from lunar landing through postearth landing recovery.

b. The time from recovery through transportation to the Lunar Receiving Lab.

c. The post mission confinement in the Lunar Receiving Lab.

Figures 8 and 9 show the spacecraft and recovery criteria recommended by the interagency committee.

Until recently no specific requirement has existed to provide hardware or develop procedures that minimize the sources of back contamination. However, the interagency committee for back contamination has recently recommended spacecraft and recovery procedures as well as approved the design requirements and functions of the Lunar Receiving Lab.

This criteria has recently been approved by NASA Headquarters and currently is under study by MSC.

This criteria was established as a result of a briefing at MSC on April 13, 1966, at which MSC outlined the spacecraft, recovery, and lab current capabilities and requirements.

The committee gave its approval to the lab requirements and recommended the spacecraft and recovery criteria shown on the next charts.

NASA-S-66-5135 JUN 8

SPACECRAFT AND RECOVERY RECOMMENDATIONS BY THE INTERAGENCY COMMITTEE

SPACECRAFT CRITERIA

- DEVISE AND PROTOTYPE TEST ANY COMBINATION OF THE LM CM RETURN PROCEDURES, ALONG WITH ANY DESIRED CONTAMINATION CONTROL EQUIPMENT, THAT WILL CUMMULATIVELY PRODUCE A DEGREE OF EARTH PROTECTION GENERALLY EQUIVALENT TO THAT ACHIEVED WITH AN EXHAUST FILTER

Fig. 8

NASA-S-66-5136 JUN 8

SPACECRAFT AND RECOVERY RECOMMENDATIONS BY THE INTERAGENCY COMMITTEE

RECOVERY CRITERIA

- MSC SHOULD PERFORM A STUDY OF ALTERNATIVES IN THE RECOVERY MODES SUCH AS SUBSTITUTING GOOD SANITATION, ISOLATION AND QUARANTINE TECHNIQUES IN PLACE OF SPECIAL ISOLATION EQUIPMENT

Fig. 9

Based on the interagency criteria the spacecraft procedures shown on figures 10 through 14 are being considered as ways of minimizing the source of back contamination.

The procedures have been subdivided into the various mission phases.

SUPPORTING DATA

a. Forward hatch open only 7% of time.

b. Helmets normally off for eating only 4% of time.

NASA-S-66-5137 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

LUNAR SURFACE PHASE

- THE LM FORWARD HATCH WILL BE SECURED EXCEPT FOR CREW EGRESS AND INGRESS
- PRIOR TO LM INGRESS THE CREW WILL WIPE OFF EXTERNAL GARMENTS AND BOOTS
- THE CREW WILL MINIMIZE THE HELMET OFF TIME TO REDUCE DIRECT EXPOSURE TO CABIN ENVIRONMENT
- LM LiOH CANNISTERS WILL FILTER CABIN/SUIT ENVIRONMENT
- SCIENTIFIC CONTAINERS WILL BE WIPED OFF PRIOR TO LM INGRESS

Fig. 10

NASA-S-66-5138 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

LM TO CSM TRANSFER PHASE

- AFTER HARD DOCK THE CSM WILL PRESSURIZE THE DOCKING TUNNEL, THE LM ECS WILL BE DEACTIVATED, THE LM RELIEF VALVE WILL BE MANUALLY VENTED TO CREATE A FLOW FROM CM THROUGH LM. PRESSURE HATCH REMOVED AND FLOW OF AIR WILL COME FROM CM THROUGH LM
- EXTERNAL TMG'S REMAIN IN LM ASCENT STAGE
- EQUIPMENT TRANSFERRED TO CSM WILL BE "WIPED OFF" IN THE LM PRIOR TO TRANSFER

TRANSEARTH COAST

- LiOH CANNISTERS FILTER CABIN/SUIT ENVIRONMENT

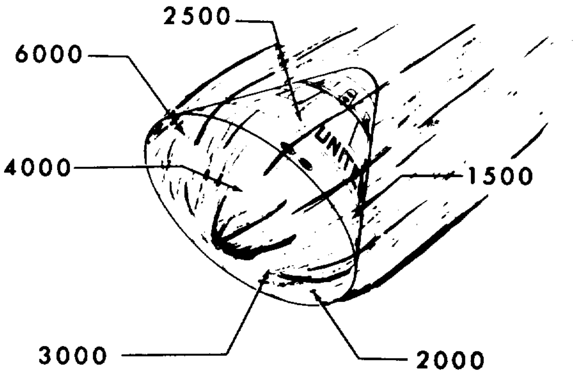
[illegible]

NASA-S-66-5235 JUNE

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

ENTRY PHASE

- LiOH CANISTERS FILTER CABIN/SUIT ENVIRONMENT
- EXTERNAL REENTRY TEMPERATURES WILL DECONTAMINATE THE EXTERNAL SURFACE OF THE SPACECRAFT



SURFACE TEMPERATURE °F

90% SURFACE LOST

600°F BOND LINE AT CHUTE DEPLOYMENT

300°F BOND LINE AT TOUCHDOWN

Fig. 12

NASA-S-66-5139 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

POSTLANDING

- VENTILATION SYSTEM PROVIDES 100 - 150 CFM AIR CIRCULATION. STUDY CURRENTLY BEING PERFORMED TO SHOW IMPACT OF PROVIDING POSTLANDING BIOLOGICAL FILTRATION.

Fig. 13

NASA-S-66-5234 JUN

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION (CONT)

	PRESENT SYSTEM WITH FILTER	REDESIGNED SYSTEM		
		NEW FAN WITH FILTER	PRESENT FAN WITH FILTER PLUS WATER COOLED SUITS	NEW FAN WITH FILTER PLUS WATER COOLED SUITS
△ INERT WEIGHT	4 LBS	210 LBS	35 LBS	43 LBS
EFFECTIVE CSM WEIGHT	9 LBS	450 LBS	75 LBS	93 LBS
LIMITS	THERMAL CONTROL OF CREW INADEQUATE		THERMAL CONTROL OF CREW MARGINAL	
COST & SCHEDULE	TBD	TBD	TBD	TBD

Fig. 14

Figures 15 through 20 show the postulated recovery procedures being considered from time of initial pararescue contact through shipment to the Lunar Receiving Lab.

The procedures have been subdivided for various situations. That is, crew egress prior to retrieval: Crew in CM at ship retrieval; Crew transfer to LRL; Spacecraft transfer to LRL.

NASA-S-66-5140 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

CREW EGRESS PRIOR TO RETRIEVAL

- PARARESCUE PERSONNEL JUMP WITH FLOTATION COLLAR AND BIOLOGICAL ISOLATION SUITS
- PARARESCUE PERSONNEL INSTALL FLOTATION COLLAR, PUT BIOLOGICAL ISOLATION SUITS IN RAFT OUTSIDE OF CM
- CREW EGRESS FROM CM, GET INTO RAFT , AND DON BIOLOGICAL ISOLATION SUITS
- PARARESCUE PERSONNEL ASSIST FLIGHT CREW ONLY IF NEEDED; OTHERWISE MOVE SOME DISTANCE AWAY UNTIL THE CREW DONS SUITS
- RECOVERY SHIP PROCEEDS TO LANDING POINT; RETRIEVES SPACECRAFT, FLIGHT CREW, AND PARARESCUE PERSONNEL

Fig. 19

NASA-S-66-5141 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

CREW EGRESS PRIOR TO RETRIEVAL (CONTINUED)

- AFTER CM IS ABOARD SHIP, ONE MEMBER OF FLIGHT CREW REENTERS CM IN ORDER TO REMOVE LUNAR SAMPLES, FLIGHT TAPES,ETC. ITEMS ARE PLACED IN AN APPROPRIATE CONTAINER, CREW MEMBER EGRESSES, AND DONS A FRESH BIOLOGICAL ISOLATION SUIT.
- FLIGHT CREW PROCEED TO QUARANTINE QUARTERS OR FACILITY FOR MEDICAL AND MISSION DEBRIEFING. (MEDICAL PERSONNEL ARE THEN QUARANTINED ALONG WITH THE CREW.)
- FLIGHT CREW AND MEDICAL PERSONNEL STAY IN QUARANTINE UNTIL SHIP REACHES NEAREST APPROPRIATE DOCK.
- ALL WASTE PRODUCTS OF QUARANTINED PERSONNEL ARE COLLECTED AND TREATED.

Fig. 20

NASA-S-66-5142 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

CREW IN CM AT SHIP RETRIEVAL

- PARARESCUE PERSONNEL JUMP WITH FLOTATION COLLAR, BIOLOGICAL ISOLATION SUITS, AND CABIN FILTRATION DEVICE *
- PARARESCUE PERSONNEL INSTALL FLOTATION COLLAR AND REMOVE CM HATCH AND INSTALL FILTRATION DEVICE
- PARARESCUE PERSONNEL ASSIST FLIGHT CREW ONLY IF NEEDED; OTHERWISE MOVE SOME DISTANCE AWAY
- RECOVERY SHIP PROCEEDS TO LANDING POINT; RETRIEVES SPACECRAFT/CREW, AND PARARESCUE PERSONNEL
- AFTER CM IS RETRIEVED AND PLACED ON DECK, FLIGHT CREW EGRESS WITH LUNAR SAMPLES, FLIGHT TAPES, ETC, DON BIOLOGICAL ISOLATION SUITS, AND PROCEED TO QUARANTINE QUARTERS OR FACILITY FOR MEDICAL AND MISSION DEBRIEFING

*CABIN FILTRATION DEVICE CURRENTLY UNDER STUDY. IF SPACECRAFT HAS FILTER, THIS DEVICE MAY NOT BE REQUIRED

Fig. 17

NASA-S-66-5143 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

CREW IN CM AT SHIP RETRIEVAL (CONT)

- FLIGHT CREW PROCEED TO QUARANTINE QUARTERS OR FACILITY FOR MEDICAL AND MISSION DEBRIEFING. (MEDICAL PERSONNEL ARE THEN QUARANTINED ALONG WITH THE CREW)

- FLIGHT CREW AND MEDICAL PERSONNEL STAY IN QUARANTINE UNTIL SHIP REACHES NEAREST APPROPRIATE DOCK

- ALL WASTE PRODUCTS OF QUARANTINED PERSONNEL ARE COLLECTED AND TREATED

Fi. [illegible]

NASA-S-66-5144 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

CREW TRANSFER TO AIRCRAFT FOR FLIGHT TO LRL, HOUSTON

- FLIGHT CREW AND MEDICAL PERSONNEL DON BIOLOGICAL ISOLATION SUITS, LEAVE QUARANTINE QUARTERS, AND TRANSFER BY APPROPRIATE TRANSPORTATION TO AIRCRAFT. LUNAR SAMPLES, TAPES, ETC., ARE TRANSFERRED
- FLY TO ELLINGTON, TRANSFER BY APPROPRIATE TRANSPORTATION TO LRL

Fig. 19

NASA-S-66-5145 JUN 8

PROCEDURES POSTULATED TO MINIMIZE SOURCES OF BACK CONTAMINATION

QUARANTINE SPACECRAFT

- ABOARD SHIP THE HATCH IS REPLACED
- EXTERNAL SURFACES OF SPACECRAFT ARE CHEMICALLY TREATED
- AT DOCKSIDE SM/RCS IS DEACTIVATED AND THEN TRANSFERRED TO LSRL BY AIR AND SURFACE TRANSPORTATION

Fi. .

Figure 21 shows the general flow sequence from recovery of spacecraft, crew, lunar samples, and miscellaneous equipment from the recovery ship through distribution and return of scientific samples.

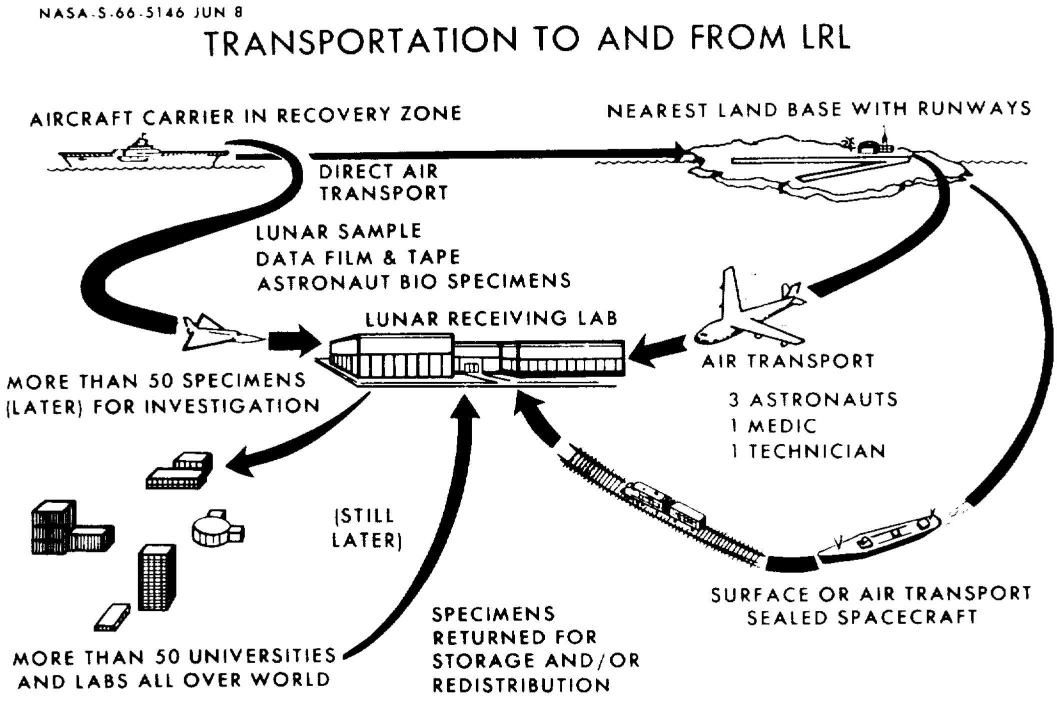

NASA-S-66-5146 JUN 8
TRANSPORTATION TO AND FROM LRL
AIRCRAFT CARRIER IN RECOVERY ZONE
NEAREST LAND BASE WITH RUNWAYS
DIRECT AIR TRANSPORT
LUNAR SAMPLE
DATA FILM & TAPE
ASTRONAUT BIO SPECIMENS
LUNAR RECEIVING LAB
AIR TRANSPORT
3 ASTRONAUTS
1 MEDIC
1 TECHNICIAN
MORE THAN 50 SPECIMENS (LATER) FOR INVESTIGATION
(STILL LATER)
SURFACE OR AIR TRANSPORT SEALED SPACECRAFT
MORE THAN 50 UNIVERSITIES AND LABS ALL OVER WORLD
SPECIMENS RETURNED FOR STORAGE AND/OR REDISTRIBUTION

Fig. [illegible]

Figures 22 through 26 show the Lunar Receiving Lab functions during quarantine plus the containment concepts currently planned for the spacecraft, crew, and lunar samples.

NASA-S-66-5151 JUN 8

LUNAR RECEIVING LABORATORY OPERATIONS

LUNAR RECEIVING LAB FUNCTIONS DURING QUARANTINE

- ASTRONAUT ISOLATION
  - POST FLIGHT MEDICAL EXAMINATION
  - TECHNICAL DEBRIEFING
  - PARTICIPATION IN SAMPLE IDENTIFICATION AND SCIENTIFIC DEBRIEFING
  - EXPERIENCE INPUT TO NEXT FLIGHT
- SAMPLE ISOLATION
  - OPEN CONTAINERS, IDENTIFY & CATALOG
  - REMOVE SPECIMENS FOR QUARANTINE CLEARANCE TESTS
  - PERFORM TIME DEPENDENT SCIENTIFIC EXPERIMENTS
  - PREPARE FIFTY OR MORE SPECIMENS TO BE SENT TO OUTSIDE LABS

Fig. [illegible]

NASA-S-66-5150 JUN 8

LUNAR RECEIVING LABORATORY OPERATIONS (CONT)

LUNAR RECEIVING LAB FUNCTIONS DURING QUARANTINE

- QUARANTINE CLEARANCE TEST
  - PERFORM MINIMUM BIO-TESTS IN CONFORMANCE WITH INTERAGENCY REQUIREMENTS TO CERTIFY SAFE RELEASE OF ISOLATED PERSONNEL AND EQUIPMENT
- DATA FILM AND TAPE ISOLATION
  - PLAY TAPES THROUGH BIOLOGICAL BARRIER FOR OUTSIDE PROCESSING
  - DEVELOP FILM AND PRINT THROUGH OPTICAL PRINTER FOR OUTSIDE USE
- SPACECRAFT COMMAND MODULE ISOLATION
  - AVAILABLE FOR ESSENTIAL TECHNICAL INSPECTION
  - AVAILABLE FOR ADDITIONAL BIO SAMPLING

Fig. 3

NASA-S-66-5149 JUN 8

LUNAR BIOLOGY PROGRAM BASIC CONTAINMENT CONCEPTS

SPACECRAFT

- NO ENTRANCE PERMITTED TO SPACECRAFT INTERIOR
- SPACECRAFT STORED IN NON-QUARANTINE AREA
- IF REQUIRED, SPACECRAFT STORAGE AREA MAY BE INCORPORATED INTO CREW RECEPTION AREA
- POSSIBLE REQUIREMENTS INCLUDE
  - COMPONENT ANALYSIS OR REMOVAL
  - FURTHER BIOLOGIC ASSESSMENT OF SPACECRAFT
- REMAINS QUARANTINED FOR SAME PERIOD AS CREW (21 DAYS)

Fig. 24

NASA-S-66-5148 JUN 8

LUNAR BIOLOGY PROGRAM BASIC CONTAINMENT CONCEPTS

CREW RECEPTION AREA

- FACILITY FEATURES
  - SINGLE BARRIER
  - ALL WASTE HEAT TREATED
  - ALL EFFLUENT AIR FILTERED THROUGH 'BIOLOGICAL' FILTERS
  - NEGATIVE PRESSURE MAINTAINED INSIDE FACILITY
- NO PERSONNEL CAN LEAVE FACILITY DURING QUARANTINE PERIOD
- AS REQUIRED, PERSONNEL CAN ENTER FACILITY
- POSSIBLE REQUIREMENTS INCLUDE
  - MEDICAL ASSISTANCE
  - TECHNICAL PERSONNEL TO EXAMINE SPACECRAFT
  - FACILITIES PERSONNEL TO REPAIR EQUIPMENT
- CREW REMAINS ISOLATED FOR 21 DAYS, UNLESS SAMPLE ASSAY POSITIVE

Fig. 25

NASA-S-66-5147 JUN 8

LUNAR BIOLOGY PROGRAM BASIC CONTAINMENT CONCEPTS

LUNAR SAMPLE

- FACILITY FEATURES
  - DOUBLE BARRIER CONCEPT
  - SECONDARY BARRIER - BUILDING WALLS
  - PRIMARY BARRIER - CABINET SYSTEM
  - OPTIMUM AIR PRESSURE DIFFERENTIALS MAINTAINED
  - LIQUID EFFLUENTS HEAT STERILIZED
  - EFFLUENT AIR FROM CABINET INCINERATED
  - EFFLUENT AIR FROM ROOM FH FILTERED THROUGH 'BIOLOGICAL' FILTERS
  - 100% MAKE-UP AIR
  - ALL MAKE-UP AIR FILTERED THROUGH BIOLOGICAL FILTERS
- ALL PERSONNEL ENTER AND EXIT THROUGH CHANGE ROOM - SHOWER

Fig. [illegible]

THERMODYNAMIC CONSTRAINTS ON LUNAR MISSION CAPABILITY

by

Jerry W. Craig

THERMODYNAMIC CONSTRAINTS ON LUNAR MISSION CAPABILITY

1.0 MISSION CONCEPT FOR APOLLO THERMAL DESIGN

Thermal design of the Apollo spacecraft was initially based upon the concept of steady state worst case environmental conditions. In other words, each element of the spacecraft was to be designed to exist in the worst cold or hot condition for an indefinite time period. The primary incentive for this criteria was our desire to eliminate any potential mission constraints.

Early calculations revealed cost, complexity and weight penalties would be large if this criteria was adhered to. For instance, a cooling loop was required for maintaining temperature control of propellants for both Reaction Control System (RCS) and Service Propulsion System (SPS).

Because of this large impact, it was decided that thermal design mission conditions should be established. This criteria has been used to desigh the spacecraft thermally and establishes a certain boundary within which the spacecraft missions must be planned. Figure 1 indicates salient features of this design mission. This does not mean that we will fly this mission. Sufficient flexibility results from designing to this criteria that a large variety of mission conditions can be accommodated.

We are presently conducting parametric analysis to better define this operating envelope of the thermal design. We expect to provide mission planning with sufficient data by the end of the summer to more realistically include thermal constraints in the planning of the lunar mission.

To illustrate the sensitivity of the spacecraft to mission conditions, let us examine the response characteristics of the Command Module heat shield. Figure 2 shows the response of the heat shield to the space environment and indicates the relaxation in temperature requirements afforded by the mission constraints.

Virtually all components in the spacecraft have temperature limitations which could be exceeded under some conceivable mission condition. However, certain components are more sensitive. Figure 3 shows those components which analysis has revealed to be most sensitive.

In summary, we have designed for well defined transient mission conditions with a capability for limited steady-state (worst case) conditions, and normal communication, guidance, experimental thrusting, or contingency type mission requirements. We are actively engaged in analysis to provide parametric data to completely define the operating envelope of the thermal design of the spacecraft. Figure 4 summarizes these primary points.

2.0 THERMAL COATING CONTAMINATION DATA

The greatest uncertainty in establishing the capability of the spacecraft thermal control system to effect the required mission evolves from a lack of understanding of the effect of self-induced environments on the spacecraft thermal coating. Figure 5 shows a layout of the upper stages of the lunar vehicle. During boost, the solid propellant rocket motors for jettison of the Launch Escape System (LES) and Saturn II stage impinge directly on the spacecraft. In addition, cork located on both the Service Module and Command Module for thermal protection of the structure during boost emits ablation products which can impinge on spacecraft thermal control surfaces.

The thermal control surfaces which concern us most are also indicated on Figure 5. The effect of degraded coating performance will be discussed in greater detail a little later. However, let me briefly summarize the functions affected. The Environmental Control System (ECS) radiator heat rejection capability is reduced as the solar absorptance of the radiator surface is increased. The result is a requirement for increased water boiling. A similar reduction in Electrical Power System (EPS) radiator capability results from coating degradation. Such a reduction here means that the allowable fuel cell power level is restricted. We have already seen an indication of the response characteristics of the heat shield and how they vary with changing coating properties. The primary effect is that our original mission flexibility is further restricted.

The first information regarding the effect of the boost environment on thermal coatings was collected by the Marshall Space Flight Center (MSFC) during the SA-8, -9, and -10 missions. Several thermocouples were located on the Service Module Adapter (SMA) between the Saturn IV stage and the payload. These temperatures have been recorded for many months and correlated with the SMA coating properties. Figure 6 shows these data. Note that the α/ϵ ratio (ratio of solar absorptance to infrared emittance) has increased by a factor of two over the initial value.

Several calorimeters were shrouded so as to protect them from the boost environment. These measurements indicated no change in coating properties in the shrouded areas. Figure 7 shows the configuration of these vehicles compared to the Saturn V lunar vehicles.

Additional data were collected on a ground test at the Arnold Engineering Development Center (AEDC). The test setup is shown in Figure 8. The test consisted of firing a 3,500-pound thrust engine at altitude and measuring the effects for thermal coating properties. Two Apollo coatings were tested. The results of the test are shown in Figure 9. Note that the α/ϵ increase of the radiator coating is somewhat less than that experienced in the SA-8, -9, and -10 flights but is still of sufficient magnitude to significantly affect thermal performance.

The Service Module coating α&ϵ each increased, but at nearly the same rate, so that the α/ϵ ratio remained approximately constant.

Figure 10 shows results of analyses conducted to relate test data to the lunar mission design situations. This analysis consists of a prediction of particle impingement in the test environments and in the lunar mission. The analysis indicates a more severe environment for the lunar mission than the test conditions; therefore, some extrapolation is required.

A second type of test has also been conducted to further define expected degradation. An emission spectrographic analysis of deposits from the Spacecraft 009 and 002 windows and the test samples from the AEDC ground test are shown in Figure 11. Note the similarity of deposits on all three samples; they would not be expected to be exactly the same due to the slightly different environment, handling procedures, etc. Most of the metal oxides contained in the solid propellants are found on all three samples. These deposits have been laboratory synthesized and deposited on coatings in varying densities on the optical properties listed. The results of this investigation are shown on Figure 12. Preliminary indications are that this data correlated with the previous tests.

We are also concerned about the effect of these deposits on the transmittance of the spacecraft windows. Figure 13 shows the effect of the AS-201 mission environment on Spacecraft 009 window performance. Spacecraft 002 data shows a similar degradation. We are not certain about the relative effect of boost, entry, and recovery; however, we strongly suspect that most of the degradation occurs during boost. A camera will be mounted in Spacecraft 011 to record the change in window clarity during the AS-202 boost phase.

The effect of RCS plume impingement on coatings will be determined by scaled ground tests at a simulated altitude of 300,000 feet. These tests will be completed in July and will include the effect of ablating cork.

No applicable test data is available to quantitate cork ablation biproducts effect on coating performance. No tests other than the RCS motor tests are funded to further define this uncertainty.

A test to determine coating properties of Spacecraft 012 will be conducted during mission AS-204. This test should provide reasonable substantiation of our predicted lunar constraints.

3.0 MISSION CONSTRAINTS

The mission constraints which result from the coating degradation will now be discussed in detail. Figure 14 summarizes the predicted coating properties after boost for those components which are most sensitive.

We predict the ECS radiator coating to degrade to an $\alpha_s = 0.5$, while the emittance remains 0.92. An analysis of radiator performance with and without this degradation was conducted. Power requirements, heat leaks, metabolic rates, etc., for the lunar mission were also determined. A heat load and water generation profile resulted. Combination of the above data results in the predicted radiator mission constraint as shown on Figures 15 and 16. These plots of available spacecraft water as a function of time show that the present water tankage is inadequate for abort from lunar orbit after failure of the primary cooling system. The system would be adequate if no failure occurred in the primary cooling loop or if the radiator coating were protected to prevent boost contamination.

The maximum power capability of the Electrical Power System is shown on Figure 17 as a function of radiator solar absorptance for the design condition of two fuel cell operation in lunar orbit. The predicted coating degradation will limit fuel cell power level to approximately 2,450 watts for periods greater than one hour in lunar orbit. Since our mission study indicates no requirements in excess of this level, this poses no mission constraint.

The coating requirements for the Command Module heat shield are shown in Figure 18. Also noted are the expected properties following the boost contamination. Figure 19 summarizes the effect of the contamination. The reduction in absolute minimum temperature means that either tighter constraints must be placed upon spacecraft attitude or the temperature requirement relaxed. The specification minimum heat shield temperature of -150°F is based on microscopic surface cracking of the ablator and gaps in the heat shield joints resulting from thermal strains The actual occurence of cracking is unlikely in the thinest sections of the heat shield which are the coldest. The thermal strain analysis is not considered of sufficient accuracy to predict that gaps will not occur at -150°F and will occur at -170°F. The heat shield thermal deformations will be verified in the MSC Space Environment Simulation Laboratory (SESL). An increase in predicted entry temperature requires that the heat shield be cold-soaked back to +150°F prior to entry for a worst case entry trajectory. Cabin environment temperature extremes are changed by less than 5°F as the result of coating degradation.

Although other coatings are used on the spacecraft, their requirements are sufficiently loose so that no problem results from the boost contamination.

We reviewed the available window contamination data previously. It is expected that jettisonable covers will be necessary to assure the visibility required for docking and scientific observation. The data from the AS-202 mission should confirm this. Such covers are in design at the present time.

MSC is also designing covers which can be used for protection of the Service Module radiators if desired. Effectivity of these covers will be determined upon conclusion of the present studies.

4.0 CONCLUSIONS

Conclusions resulting from these studies are as follows:

1. Execution of the lunar mission requires addition of ECS cooling water and tankage or boost protective covers for the ECS radiator coating.

2. EPS radiator performance is acceptable with predicted coating degradation.

3. Command Module heat shield and cabin temperatures slightly exceed specification limits for predicted Command Module coating degradation. These off-limits conditions are acceptable.

4. Predicted degradation was obtained by extrapolation of limited data; therefore, some uncertainty exists. They are "best engineering judgment" predictions. Flight data from AS-204 and -205 is required for confirmation.

5. The spacecraft windows will be degraded. The extent and acceptability will be investigated further and verified by flight data.

6. No spacecraft changes will be implemented pending results of boost cover design studies and further flight data.

NASA-S-66-5219 JUN

THERMAL ATTITUDE CRITERIA

EARTH PARKING ORBIT

- X-AXIS PARALLEL TO VELOCITY VECTOR ± 20°
- 1.5 TO 4.5 HOURS DURATION
- S C ROLLED 180° 1 HOUR AFTER START OF THIS PHASE
- CONSIDER WORST CASE ROLL WITHIN ABOVE LIMITS

TRANSLUNAR INJECTION THRU TRANSPOSITION

- 0.5 TO 2.0 HOURS DURATION
- CONSIDER ATTITUDE RANDOM (i e, WORST CASE)

Figure 1

NASA-S-66-5221 JUN

THERMAL ATTITUDE CRITERIA (CONT)

TRANSLUNAR-TRANSEARTH COAST

- X-AXIS NORMAL TO INCIDENT SUN ± 20°
- ROLL AT 1.0-2.5 REV/HOUR
- 60 TO 110 HOURS DURATION
- ARBITRARY (WORST CASE) ATTITUDE PERIODS OF 3 HOURS MAX AT ANY TIME
- ROLL MODE STABILIZED TEMPERATURES TO INITIATE ALL ARBITRARY ATTITUDE PERIODS

Figure 1 (cont)

NASA-S-66-5220 JUN

THERMAL ATTITUDE CRITERIA (CONT)

LUNAR ORBIT

- X-AXIS APEX DOWN ± 20° OF LOCAL VERTICAL
- ARBITRARY ATTITUDE FOR LM ASCENT & DESCENT
- ARBITRARY ATTITUDE FOR PERIODS OF 3 ORBITS WITH TOTAL NOT TO EXCEED 8 ORBITS
- ASSUME STABILIZATION TO START ARBITRARY PERIOD

LUNAR LANDING

- PRIOR KNOWLEDGE OF DAY OR NIGHT LANDING
- LANDING ORIENTATION CONSTRAINED IN TILT ANGLE & SEPARATION DISTANCE

Figure 1 (cont)

NASA-S-66-5227 JUN

CM HEATSHIELD - THERMAL RESPONSE

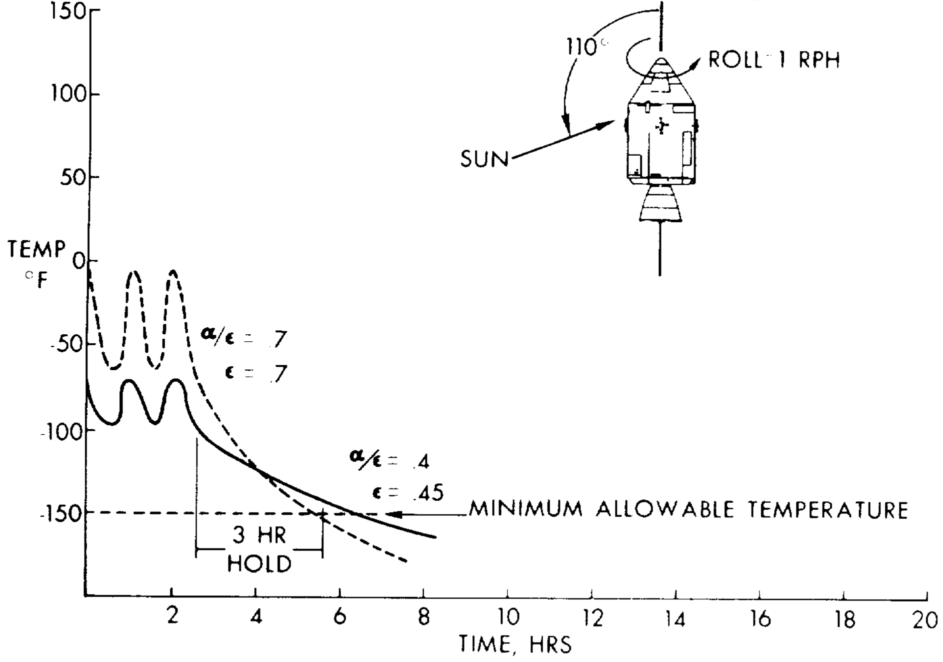


Figure [illegible]

NASA-S-66-6126 JUN

MOST SENSITIVE COMPONENTS

CSM

- CM HEAT SHIELD
- CM RCS ENGINE
- SM RCS ENGINE
- SPS ENGINE AND FEED LINES
- ECS COOLANT LOOP
- EPS COOLANT LOOP
- CREW COMPARTMENT
- THERMAL COATINGS
- EXPENDABLE STORAGE

Figure 3

NASA-S-66-6428 JUN

MOST SENSITIVE COMPONENTS (CONT)

- LM
  - LANDING GEAR
  - EXPENDABLE STORAGE
  - HIGH GAIN, LANDING AND RENDEZVOUS RADAR ANTENNA
  - A/S RCS ENGINE
  - DESENT ENGINE
  - ASCENT ENGINE
  - THERMAL COATINGS
  - COOLANT LOOP
  - CREW COMPARTMENT

Figure 3 (cont)

NASA-S-66-6472 JUN

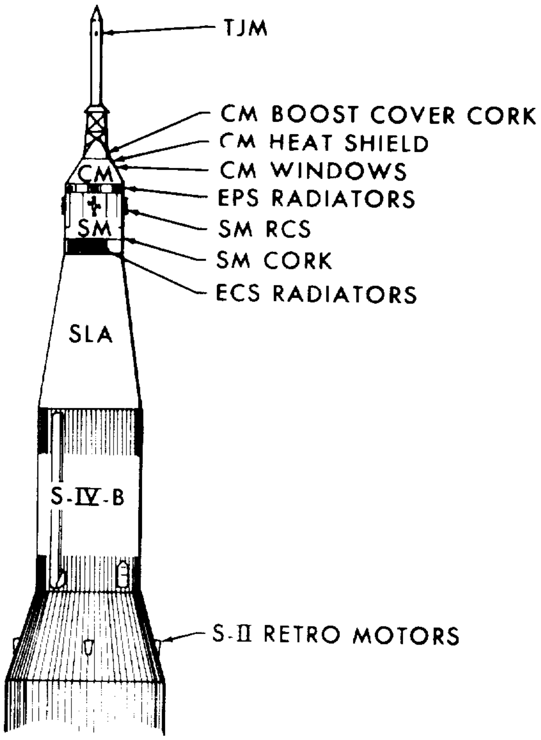
THERMAL DESIGN

- DESIGN IS BASED UPON CONTROLLED TEMPERATURE TRANSIENTS
- STEADY-STATE DESIGN RESULTS IN WEIGHT AND COMPLEXITY PENALTIES
- LIMITED CAPABILITY FOR WORST CASE MISSION CONDITIONS
- THESE CONDITIONS ESTABLISH DESIGN ENVELOPE
- THIS CAPABILITY ALLOWS A WIDE RANGE OF MISSIONS TO BE EXECUTED
- PARAMETRIC DATA TO DEFINE BOUNDS OF CAPABILITY IS IN WORK

Figure 4

NASA-S-66-6040 MAY

SOURCE OF DEPOSITS


TJM
CM BOOST COVER CORK
CM HEAT SHIELD
CM WINDOWS
EPS RADIATORS
SM RCS
SM CORK
ECS RADIATORS
CM
SM
SLA
S-IV-B
S-II RETRO MOTORS

- LAUNCH VEHICLE RETRO ROCKETS
- TOWER JETTISON MOTOR OPERATION
- ABLATING CORK

Figure 5

NASA-S-66-5035 JUNE 1

CONTAMINATION DATA

FLIGHT TEST DATA

(S-13 WHITE PAINT - USED ON S-IV)
SATURN 8,9,AND 10 (PEGASUS) INFLIGHT DATA - MSFC

	EXPOSED SMA EXTERNAL	SHROUDED CALORIMETERS
INITIAL α/ϵ	0.25	0.25
FINAL α/ϵ	0.5 TO 0.6	0.25

Figure 6

NASA-S-66-5216 JUN

SOLID ROCKET PLUME GEOMETRY COMPARISON

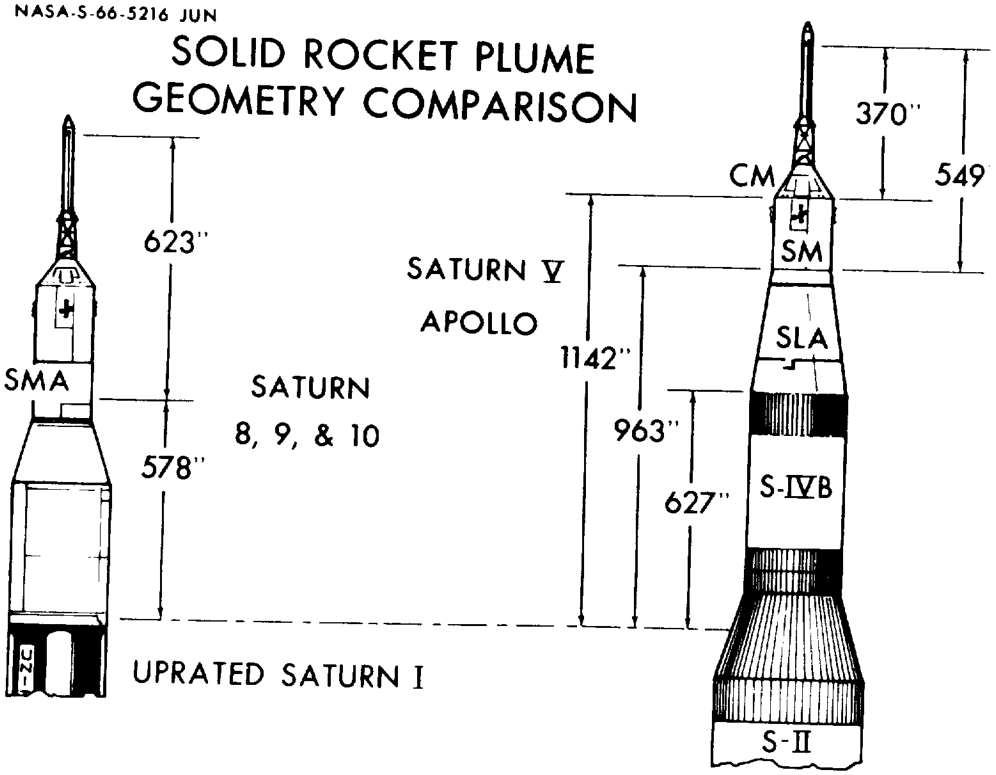

623"
578"
SMA
SATURN 8, 9, & 10
UPRATED SATURN I
SATURN V APOLLO
370"
549
CM
SM
SLA
1142"
963"
627"
S-IVB
S-II

Figure 7

NASA-S-66-5224 JUN

CENTAUR/IB CONTAMINATION TEST AT AEDC

J-4 FACILITY
TEST ALTITUDE: 120,000 FEET
MOTOR: TX-143

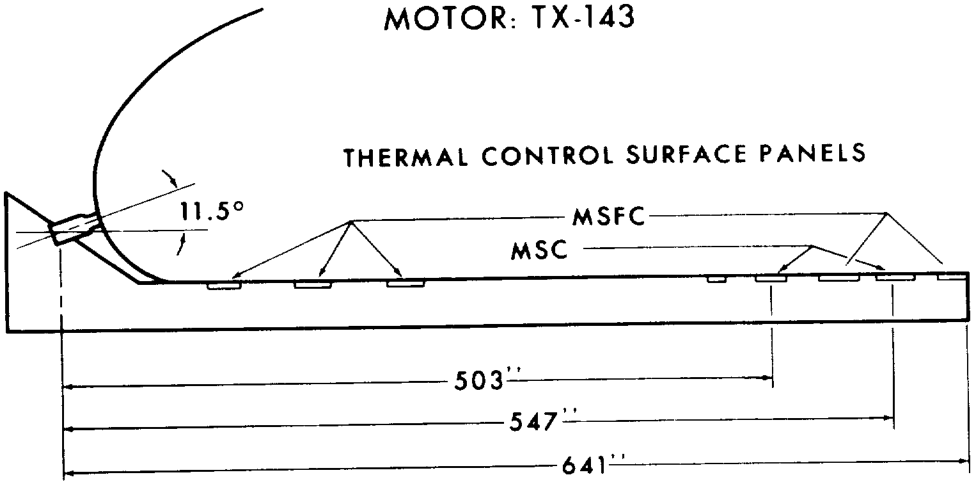

THERMAL CONTROL SURFACE PANELS
11.5°
MSFC
MSC
503''
547''
641''

Figure 8

NASA-S-66-5036 JUNE 1

CONTAMINATION DATA (CONT)

GROUND TEST DATA

AEDC

APOLLO RADIATOR COATING	α/ϵ	α	ϵ
INITIAL (PRE-FIRE)	.196	.18	.92
FINAL (POST-FIRE)			
AVERAGE	.35	.33	.93
RANGE	.28-.45	.26-.42	.92-.94
APOLLO SM SKIN COATING	α/ϵ	α	ϵ
INITIAL (PRE-FIRE)	1.0	.25	.25
FINAL (POST-FIRE)			
AVERAGE	1.14	.50	.43
RANGE	.96-1.43	.38-.63	.28-.66

Figure

NASA S-66-5406 MAY 31

TJM AND S-II RETRO PARTICLE IMPINGEMENT

	TOTAL INCIDENT	COATING PROPERTIES			
	FLUX LB_m/FT^2	INITIAL		FINAL	
		α	ϵ	α	ϵ
AEDC TEST 528 INCHES FROM NOZZLE	1.64×10^{-4}	.18	.90	.42	.90
AEDC TEST 569 INCHES FROM NOZZLE	1.32×10^{-4}	.18	.90	.35	.90
SA 8, 9, 10 - SMA	2.02×10^{-3}	.22	.90	.45-.54	.90
APOLLO - CM	8.3×10^{-3}	.16	.40		
APOLLO - SM	2.9×10^{-3}	.18	.90	PREDICTED .50	.90

Figure 10

NASA-S-66-5033 JUNE 1

EMISSION SPECTROGRAPHIC ANALYSIS

	SPACECRAFT 009	SPACECRAFT 002	AEDC TESTS
ALUMINUM	MAJOR**	LIGHT MINOR	*
ANTIMONY	TRACE	NONE	NONE
BARIUM	TRACE	NONE	NONE
BISMUTH	NONE	TRACE	NONE
BORON	LIGHT MINOR	LIGHT MINOR	LIGHT MINOR
CADMIUM	BASE	NONE	BASE
CALCIUM	HEAVY MINOR	BASE	TRACE
CHROMIUM	LIGHT MAJOR	TRACE	TRACE
COPPER	MINOR	BASE	LIGHT MINOR
IRON	MAJOR	MAJOR	LIGHT MAJOR
LEAD	HEAVY MINOR	HEAVY MAJOR	LIGHT MAJOR
MAGNESIUM	HEAVY MINOR	BASE	BASE

Figure 11

NASA-S-66-5034 JUNE 1

EMISSION SPECTROGRAPHIC ANALYSIS (CONT)

	SPACECRAFT 009	SPACECRAFT 002	AEDC TESTS
MANGANESE	LIGHT MINOR	TRACE	TRACE
MOLYBDENUM	TRACE	NONE	NONE
NICKEL	TRACE	TRACE	TRACE
POTASSIUM	NONE	NONE	*
SILICON	HEAVY MAJOR	MINOR	*
SILVER	TRACE	NONE	TRACE
SODIUM	HEAVY MAJOR	MAJOR	NONE
TIN	BASE	BASE	BASE
TITANIUM	LIGHT MAJOR	LIGHT MAJOR	NONE
VANADIUM	BASE	NONE	NONE
ZINC	LIGHT MAJOR	HEAVY MINOR	*
ZIRCONIUM	TRACE	NONE	NONE

Figure 11 (cont)

NASA-S-66-5007 JUN

SOLAR ABSORPTIVITY VS CONTAMINATION FOR APOLLO T/C COATING

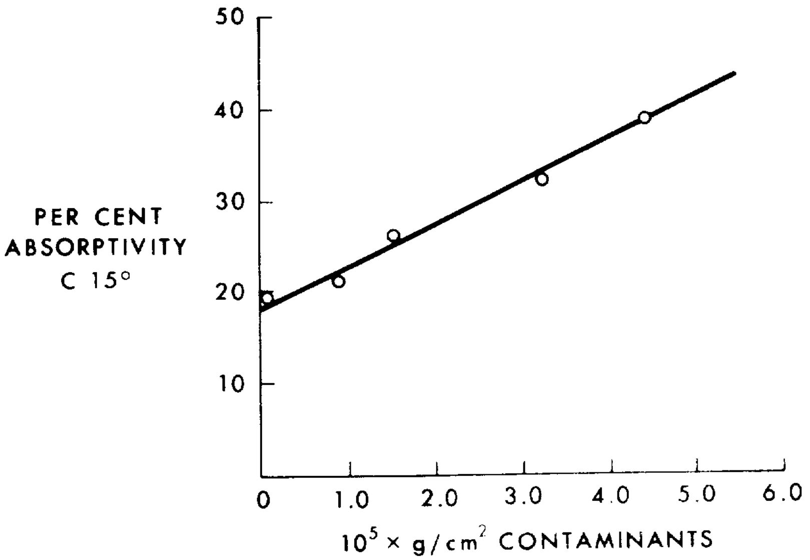


Figure 12

NASA-S-66-5223 JUN

C/M WINDOWS

AFRM 009 TRANSMITTANCE, %

	TOTAL	DIRECT	DIFFUSE
CLEAN	90 - 97	90 - 97	—
DEGRADED	68	29	39

- COMMENTS
  - NO DETERMINATION OF RELATIVE EFFECT OF BOOST, ENTRY, AND RECOVERY
  - CAMERA WILL BE FLOWN ON AS 202
  - MORE QUANTITATIVE ANALYSIS IS IN PROGRESS

Figure 13

NASA-S-66-5032 JUNE 1

PRESENT THERMAL COATINGS

	α		ϵ	
	DESIGN	DEGRADED	DESIGN	DEGRADED
ECS RADIATOR	0.2	.5	0.92	.92
EPS RADIATOR	0.2	.5	0.92	.92
CM HEAT SHIELD	0.16	.49	0.4	.7

Figure 14

NASA-S-66-6820 JUN

MISSION WATER BALANCE

POTABLE & WASTE TANK STATUS

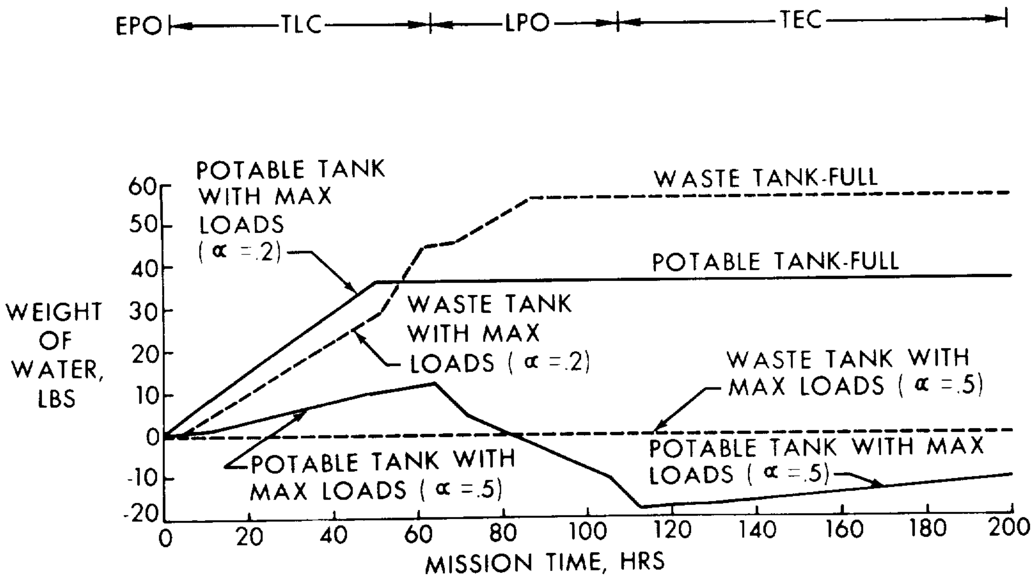


Figure 1[illegible]

NASA-S-66-5030 JUNE 1

DESIGN MISSION WATER HISTORY ABORT CASE

RADIATOR SOLAR ABSORPTIVITY = 0.5

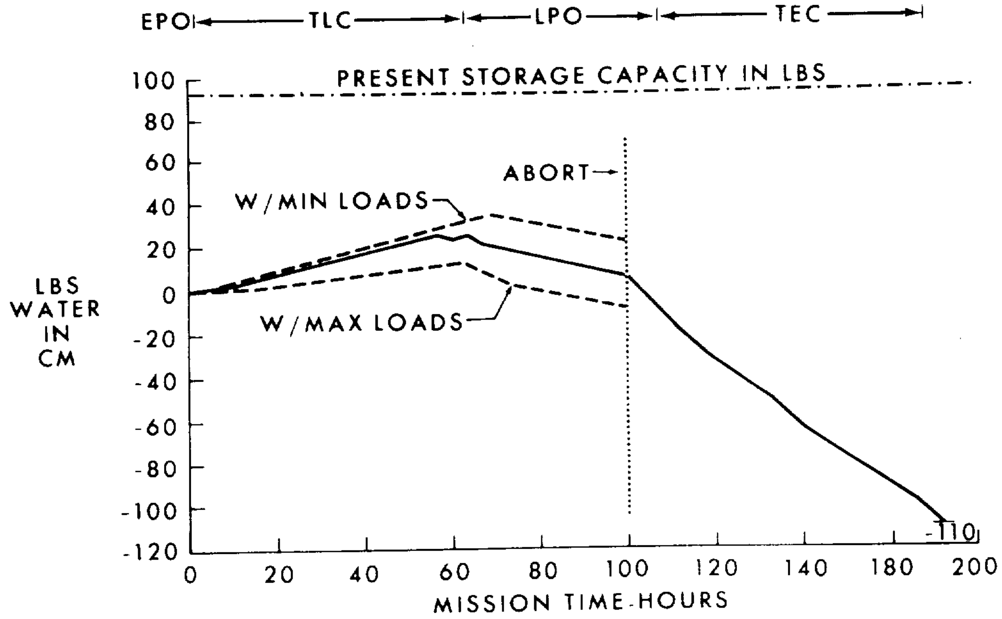


Figure [illegible]

NASA-S-66-6809 JUN

BLOCK II EPS RADIATOR CAPACITY FOR TWO FUEL CELLS IN LUNAR ORBIT

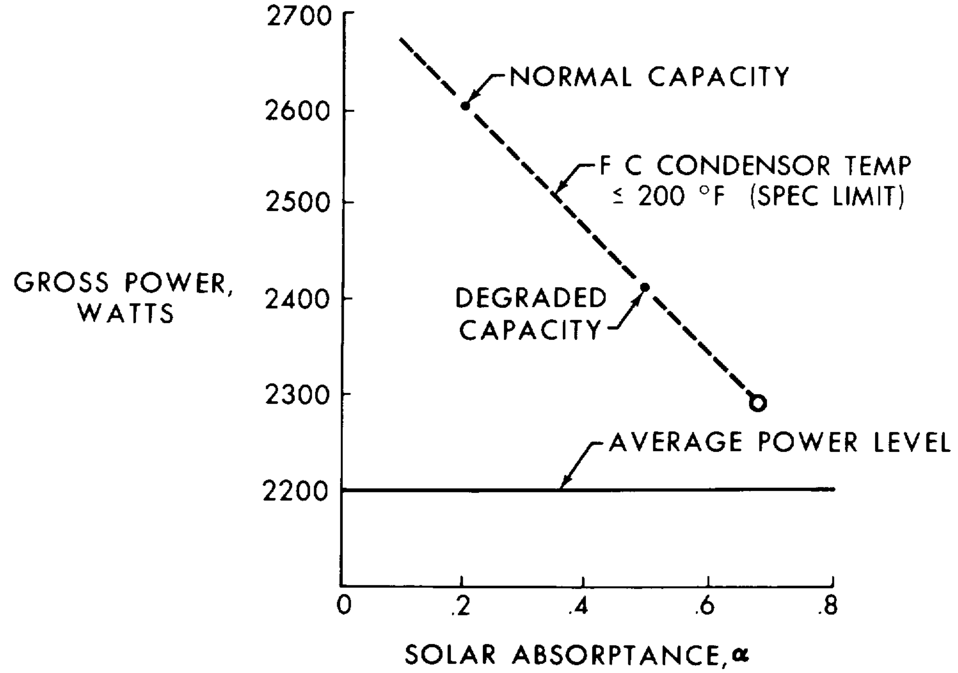


Figure 17

NASA-S-66-6450 JUN

CM COATING PERFORMANCE*

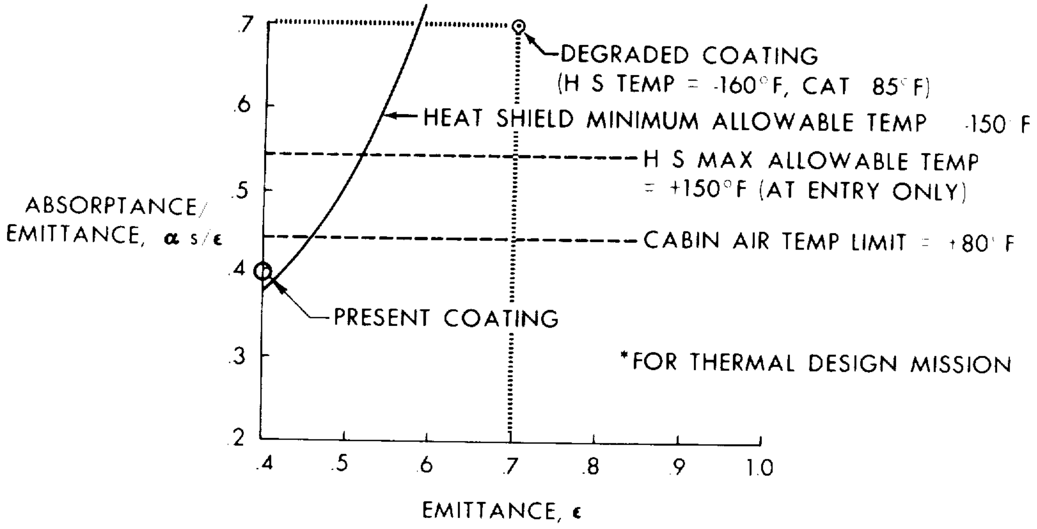


Figure 18

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RESULTING TEMPERATURE CONDITIONS - BLOCK II

HEAT SHIELD - MINIMUM ABLATOR TEMP CRITERIA

$\alpha_{s/\epsilon}=0.4$ $\epsilon=0.4$	-150
$\alpha_{s/\epsilon}=0.7$ $\epsilon=0.65$	-170°F

HEAT SHIELD - MAXIMUM ABLATOR TEMP (PRE-ENTRY)

$\alpha_{s/\epsilon}=0.4$ (BLOCK II TCC)	+108°F
$\alpha_{s/\epsilon}=0.54$	+150°F
$\alpha_{s/\epsilon}=0.7$	+190°F

ECS

$\alpha_{s/\epsilon}=0.4$	CAT ≤ 80°F
$\alpha_{s/\epsilon}=0.7$	CAT ≤ 85°F

Figure 19

SERVICE MODULE
REACTION CONTROL SYSTEM
PROPELLANT MANAGEMENT

BY

OWEN E. MAYNARD

SERVICE MODULE REACTION CONTROL SYSTEM PROPELLANT MANAGEMENT

This discussion of the Service Module RCS propellant management philosophy was not presented during the symposium due to lack of time. It is included in this report to illustrate how the maneuver requirements are being planned to provide maximum propellant reserves to accommodate contingencies.

The Service Module RCS system is used for spacecraft attitude control and for small velocity changes, such as providing main engine ullage.

Figure 1 shows the key points of the SM-RCS propellant usage philosophy. An austere baseline was developed in which only those maneuvers absolutely required were allowed. In addition, the maneuvers allowed are performed with minimum practical rates with maximum utilization of roll maneuvers instead of pitch or yaw to take advantage of the lower inertia. Planning to fly the mission in this way will assure maximum propellants for both expected and unexpected contingencies.

Expected contingencies, that is, contingencies for which RCS propellants have been specifically budgeted are loss of one quad, LM rescue and failure of the MSFN navigation loop. If no contingencies occur prior to the LM rejoining the CSM, then the propellant margin can be used for non-essential maneuvers to further enhance the mission accomplishments.

Figures 2 and 3 show the key features of the austere baseline.

Maneuver requirements during earth orbit are provided by the S-IVB stage reaction control system; hence, these requirements are not included on these figures. Navigation sightings can be made in earth orbit using the S-IVB RCS should this prove to be desirable. The SM RCS is first used for transposition and docking, as described in the session on the general mission description.

MSFN is the prime source of navigation data and only two mid-course corrections are expected during the translunar and transearth phases: one near each end of the phase. The majority of the transit time is spent in a thermal roll mode in which the spacecraft is rolled about its longitudinal axis which is maintained within ± 20° of normal to the vehicle-sun line. In lunar orbit, MSFN is again the prime source of navigation data; however, some sightings will be taken on the landing area for altitude refinement and on a few other sites for confidence. Other maneuvers required during the lunar orbit phase are as shown on the figures. Since each of these maneuvers is described in the general mission description, they are not described in this paper.

Figure 4 shows the austere maneuver plan used for IMU alignment to illustrate how SM RCS propellants can be conserved if care is taken to preplan the maneuvers for maximum efficiency.

Figures 5 and 6 show the additional maneuvers required to accomplish onboard navigation should the MSFN navigation loop fail.

Figure 7 describes the three LM rescue contingency situations considered. The first two are associated with an on-time launch, and the third one with an anytime launch.

Characteristics of these situations will be discussed in another paper, but the point to be made here is that the LM rescue represents 180 to 300 pounds of the total 790 pounds available, which is 22% to 37% of the total even if the LM performs the docking. The anytime launch situation in which the CSM must perform the docking involves about 45% of the total. Therefore, it is vital to understand the detail requirements leading to such a contingency. These are currently under investigation.

Figure 8 presents the factors and calculations related to an RCS quad failure. The consequences of a quad failure are shown under "consequences". Propellants available after a quad failure are determined based on the equations shown under "calculations". (The second one infers good management to balance quads at all times.)

The items under "checks" indicate that translation demands after a quad failure are significant, and has led to investigation of no ullage starts for the SPS with the LM rescue contingency.

Figure 9 shows the non-scheduled activities still under review.

Growth factors, of course, would be very significant. Additional safety factor considerations would only be identified as experience grows. Unevaluated factors are considered to be small.

In the case of the RCS activities, a thorough understanding of the maneuver requirements are required before they are admitted into the budget, and, of course, we must understand by simulation and flight test that the system does perform as estimated. A detailed understanding of situations which lead to anytime launch and subsequent LM rescue and finally the ability of the SPS to perform with no ullage starts at low propellant levels must be understood.

This philosophy results in a 36 lb. reserve for the worst case anytime launch (with LM performing docking) with one RCS quad out and with having to do onboard navigation from there on. For the more probable LM rescue contingency associated with on-time launch, the reserve would increase to about 50 pounds or 12% of two good quads.

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SM-RCS PROPELLANT USAGE

PHILOSOPHY

- AUSTERE BASELINE
- PLANNED CONTINGENCY RESERVE
  - ONE QUAD OUT
  - LM RESCUE
  - MSFN OUT
- EXERCISE FRUGAL MANAGEMENT OF PROPELLANT PRIOR TO LM REJOINING CSM

FIGURE 1

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BASIS FOR AUSTERE PROFILE BASELINE TRANSLUNAR AND TRANSEARTH

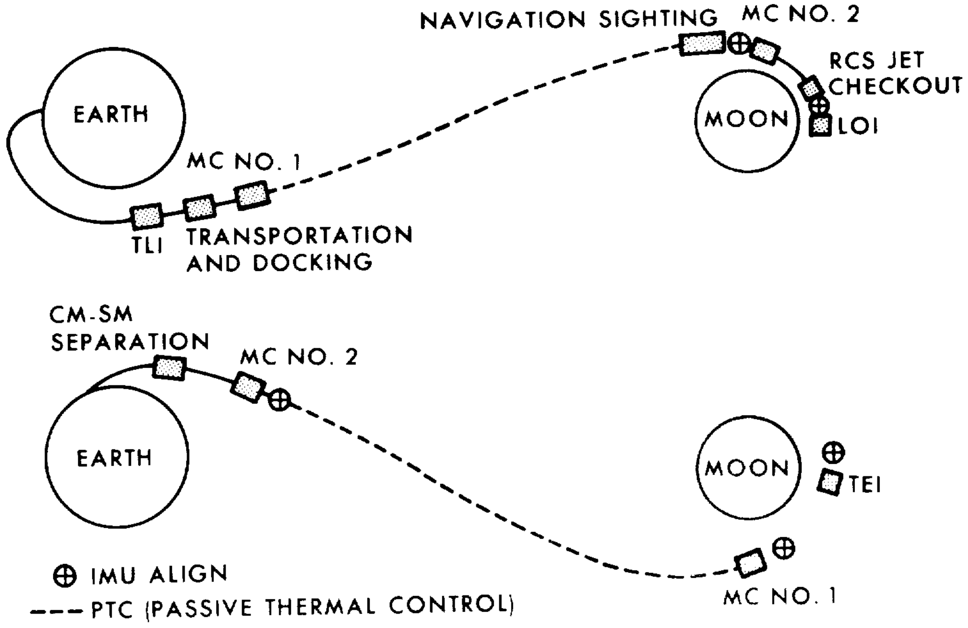
NAVIGATION SIGHTING
MC NO. 2
RCS JET CHECKOUT
EARTH
MOON
LOI
MC NO. 1
TLI
TRANSPORTATION AND DOCKING
CM-SM SEPARATION
MC NO. 2
EARTH
MOON
TEI
MC NO. 1
⊕ IMU ALIGN
--- PTC (PASSIVE THERMAL CONTROL)

FIGURE 2

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LUNAR ORBIT

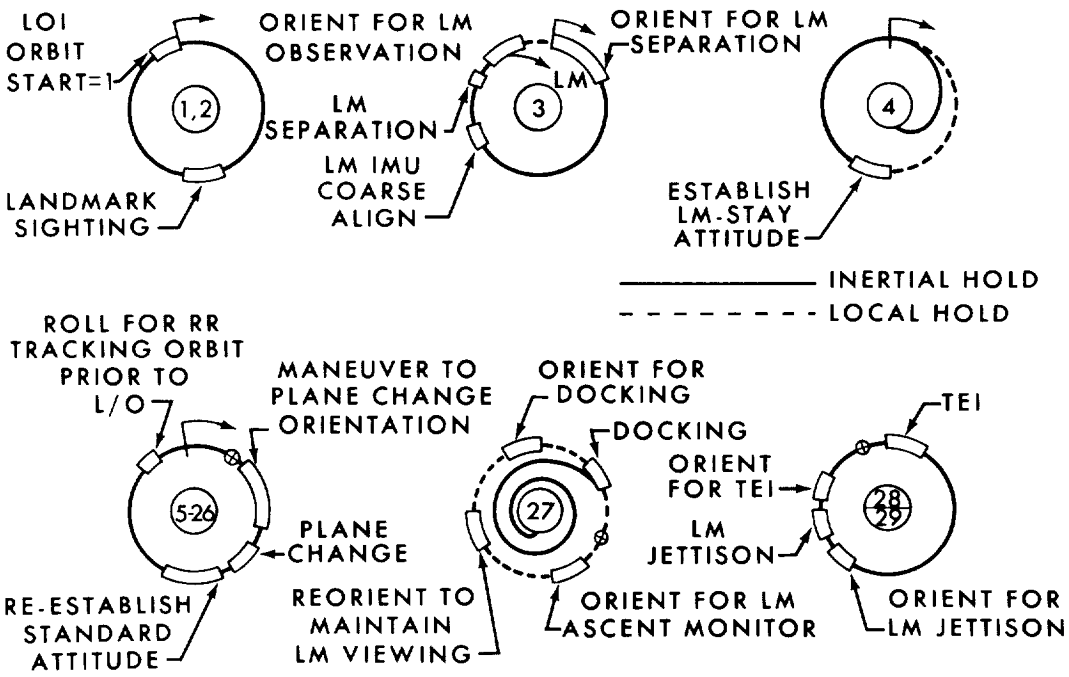
LOI ORBIT START=1
ORIENT FOR LM OBSERVATION
LM
ORIENT FOR LM SEPARATION
1,2
3
4
LM SEPARATION
LM IMU COARSE ALIGN
LANDMARK SIGHTING
ESTABLISH LM-STAY ATTITUDE
INERTIAL HOLD
LOCAL HOLD
ROLL FOR RR TRACKING ORBIT PRIOR TO L/O
MANEUVER TO PLANE CHANGE ORIENTATION
ORIENT FOR DOCKING
DOCKING
TEI
ORIENT FOR TEI
5-26
27
28
29
PLANE CHANGE
LM JETTISON
RE-ESTABLISH STANDARD ATTITUDE
REORIENT TO MAINTAIN LM VIEWING
ORIENT FOR LM ASCENT MONITOR
ORIENT FOR LM JETTISON

FIGURE 3

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EXAMPLES OF EFFICIENT MANEUVERS FOR AUSTERE PROFILE BASELINE

MANUAL ROLL MANEUVER ONLY REQUIRED FOR SCT ACQUISITION OF 2 STARS WITH LM OFF

MANUAL PITCH-ROLL ONLY REQUIRED FOR SCT ACQUISITION OF 2 STARS WITH LM ON

ONE G & N ROLL MANEUVER ONLY REQUIRED FOR SXT ACQUISITION OF 2 STARS

TWO JET ULLAGE

[illegible]

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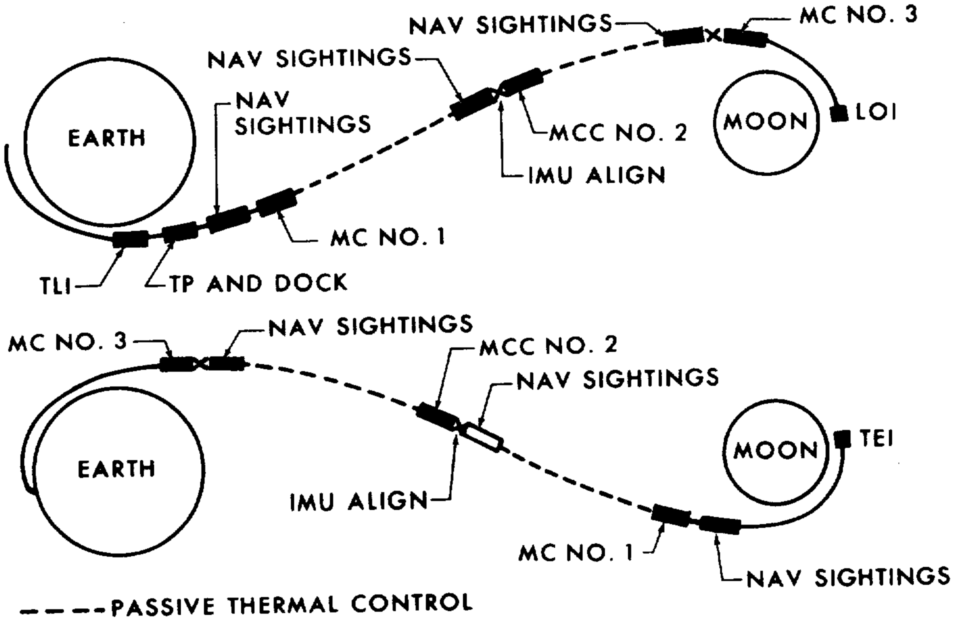
MSFN OUT



NAV SIGHTINGS
MC NO. 3
NAV SIGHTINGS
NAV SIGHTINGS
EARTH
MCC NO. 2
MOON
LOI
IMU ALIGN
MC NO. 1
TLI
TP AND DOCK
MC NO. 3
NAV SIGHTINGS
MCC NO. 2
NAV SIGHTINGS
MOON
TEI
EARTH
IMU ALIGN
MC NO. 1
NAV SIGHTINGS
PASSIVE THERMAL CONTROL

FIGURE 5

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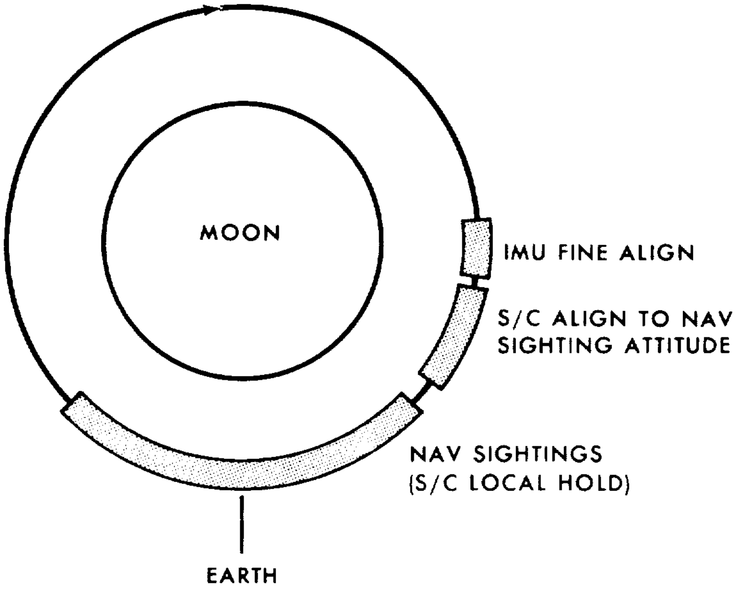
MSFN OUT-LUNAR ORBIT



MOON
IMU FINE ALIGN
S/C ALIGN TO NAV SIGHTING ATTITUDE
NAV SIGHTINGS (S/C LOCAL HOLD)
EARTH

FIGURE 6

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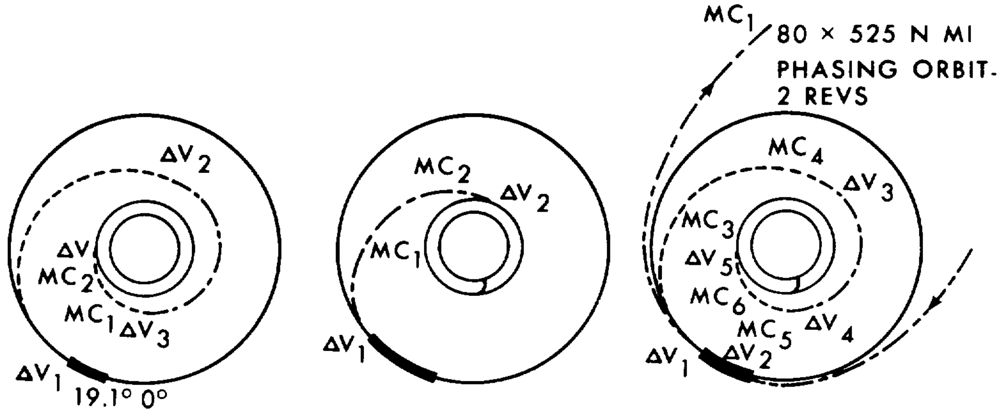
LM RESCUE CASES


MC_1
80 × 525 N MI
PHASING ORBIT-
2 REVS
ΔV_2
ΔV
MC_2
MC_1 ΔV_3
ΔV_1
19.1° 0°
MC_2
ΔV_2
MC_1
ΔV_1
MC_4
ΔV_3
MC_3
ΔV_5
MC_6
MC_5
ΔV_4
ΔV_1
ΔV_2

CSM LEAD
4 BURN COELLIPTIC

2 BURN-DIRECT

5 BURN-COELLIPTIC
ANYTIME LM LAUNCH

CSM RCS PROPELLANT FOR LM RESCUE

FULL CSM RESCUE	289 LB	272 LB	395 LB
LM-ACTIVE DOCKING	194	178	301

FIGURE 7

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TECHNIQUE FOR EVALUATING QUAD-OUT PERFORMANCE

- CONSEQUENCES
  - LOSS OF REMAINING PROPELLANT IN FAILED QUAD
  - REQUIRES ADDITIONAL PROPELLANT FOR QUAD MANAGEMENT OF REMAINING QUADS
  - TRANSLATION DEMAND MUST COME FROM REMAINING QUAD PAIR
  - ALL SUBSEQUENT ROTATIONAL MANEUVERS MUST BE MANUAL
- CALCULATIONS
  - POSTULATE EQUAL QUAD USAGE PRIOR TO FAILURE
  - PROPELLANT LOSS=PROPELLANT/QUAD=¼ (TOTAL AVAILABLE-AMOUNT USED)
  - PROPELLANT TOTAL RESERVE =3 × PROPELLANT/QUAD
  - PROPELLANT TRANSLATION RESERVE =2 × PROPELLANT/QUAD
- CHECKS
  - TRANSLATION DEMAND POST FAILURE VS TRANSLATION RESERVE
  - TOTAL DEMAND POST FAILURE VS TOTAL RESERVE

FIGURE 8

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FLEXIBILITY ACTIVITY/RESERVE

- GROWTH FACTORS
  - LATER MISSIONS

- SAFETY FACTORS
  - UNDEFINED MANUAL MANEUVERS
  - UNPREDICTABLE OPERATIONAL DISPERSIONS
  - CALCULATION APPROXIMATIONS
  - INPUT DATA EXTRAPOLATION ERRORS
  - MISSION UNCERTAINTIES
- UNEVALUATED FACTORS
  - FUEL SLOSHING
  - DISTURBANCE TORQUES
  - BODY BENDING
  - ORIENT AGAINST SOLAR FLARES
  - REPEAT NAV SIGHTINGS
  - TANK TEMPERATURE EFFECTS
  - CROSS COUPLING

FIGURE 9